A METHOD FOR PREDICTING THE STATIC AERODYNAMIC CHARACTERISTICS OF TYPICAL MISSILE CONFIGURATIONS FOR ANGLES OF ATTACK TO 180 DEGREES

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SUMMARY

A method for predicting the static, longitudinal aerodynamic characteristics of typical missile configurations at zero roll angle (i.e., in a plus configuration) has been developed and programmed for use on the IBM 7090 digital computer. It can be applied throughout the subsonic, transonic, and supersonic speed regimes to slender bodies of revolution or to nose-cylinder body combinations with low aspect-ratio lifting surfaces. The aerodynamic characteristics can be computed for missile configurations operating at angles of attack up to 180 degrees. The effect of control surface deflections for all modes of aerodynamic control are taken into account by this method. The method is based on well-known linear, nonlinear crossflow and slender body theories with empirical modifications to provide the high angle of attack capability. Comparisons of the theory with experimental data are presented to demonstrate the accuracy of the method.
# TABLE OF CONTENTS

<table>
<thead>
<tr>
<th>Section</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>INTRODUCTION</td>
<td>1</td>
</tr>
<tr>
<td>LIFT CHARACTERISTICS</td>
<td>2</td>
</tr>
<tr>
<td>DRAG CHARACTERISTICS</td>
<td>8</td>
</tr>
<tr>
<td>PITCHING MOMENT CHARACTERISTICS</td>
<td>17</td>
</tr>
<tr>
<td>COMPUTER PROGRAM DESCRIPTION</td>
<td>21</td>
</tr>
<tr>
<td>COMPARISON OF THEORY WITH EXPERIMENTAL DATA</td>
<td>22</td>
</tr>
<tr>
<td>CONCLUSIONS</td>
<td>23</td>
</tr>
<tr>
<td>REFERENCES</td>
<td>24</td>
</tr>
</tbody>
</table>

## LIST OF FIGURES

<table>
<thead>
<tr>
<th>Figure</th>
<th>Description</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Typical Missile Configurations</td>
<td>26</td>
</tr>
<tr>
<td>2</td>
<td>General Geometric Characteristics</td>
<td>27</td>
</tr>
<tr>
<td>3</td>
<td>Parameters Used to Compute Body Normal Force and Pitching Moment (from Reference 3)</td>
<td>28</td>
</tr>
<tr>
<td>4</td>
<td>Crossflow Drag Coefficient as a Function of Mach Number (from Reference 3)</td>
<td>29</td>
</tr>
<tr>
<td>5</td>
<td>Linear Lift Interference Factors (from Reference 7)</td>
<td>30</td>
</tr>
<tr>
<td>6</td>
<td>Lift Curve Slope for Wings and Tails (from Reference 8)</td>
<td>33</td>
</tr>
<tr>
<td>7</td>
<td>Crossflow Drag Coefficient for Wings and Tails as a Function of Aspect Ratio and Taper Ratio</td>
<td>35</td>
</tr>
<tr>
<td>8</td>
<td>Vortex Model Used to Determine the Lift Loss Due to Downwash (from Reference 1)</td>
<td>36</td>
</tr>
<tr>
<td>9</td>
<td>Incompressible Skin Friction Coefficient (from Reference 9)</td>
<td>37</td>
</tr>
<tr>
<td>10</td>
<td>Compressibility Effect on Turbulent Skin Friction (from Reference 9)</td>
<td>38</td>
</tr>
<tr>
<td>11</td>
<td>Transonic Wave Drag for Ogival and Blunted Conical Forebodies</td>
<td>39</td>
</tr>
<tr>
<td>12</td>
<td>External Wave Drag of Blunt Forebodies (from Reference 11)</td>
<td>40</td>
</tr>
<tr>
<td>13</td>
<td>Transonic Zero-Lift Wing Wave Drag for Unswept Wings (from Reference 9)</td>
<td>41</td>
</tr>
<tr>
<td>14</td>
<td>Ratio of Wave Drag for Noses of Various Fineness Ratios to the Wave Drag for a Hemispherical Nose</td>
<td>42</td>
</tr>
<tr>
<td>15</td>
<td>Wave Drag of a Pointed Conical Nose (from Reference 6)</td>
<td>43</td>
</tr>
</tbody>
</table>
Figure 16 - Drag Coefficient for a Flat Plate Normal to the Flow

Figure 17 - Lifting Surface Center of Pressure as a Function of Effective Aspect Ratio (from Reference 1)

Figure 18 - Subsonic Center of Pressure Location of Lift on the Body in the Pressure of Wings or Tails (from Reference 1)

Figure 19 - Supersonic Center of Pressure Location of Lift on the Body in the Pressure of Wings or Tails for $\text{BAR} \ (1 + \lambda) \left(1 + \frac{1}{m^2}\right) \leq 4.0.$ (from Reference 1)

Figure 20 - Supersonic Center of Pressure Location of Lift on the Body in the Presence of Wings or Tails for $\text{BAR} \ (1 + \lambda) \left(1 + \frac{1}{m^2}\right) > 4.0.$ (from Reference 1)

Figure 21 - Missile Axis Systems

Figure 22 - Configurations Used to Compare Theory with Experiment

Figure 23 - Comparison of Experimental Data with Theoretical Results for Configuration 1

Figure 24 - Comparison of Experimental Data with Theoretical Results for Configuration 2

Figure 25 - Comparison of Experimental Data with Theoretical Results for Configuration 3

Figure 26 - Comparison of Experimental Data with Theoretical Results for Configuration 4

LIST OF TABLES

Table 1 - Computer Program Listing

Table 2 - Input Nomenclature

Table 3 - Program Input Format

Table 4 - Output Nomenclature
<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>AR</td>
<td>exposed aspect ratio</td>
</tr>
<tr>
<td>b</td>
<td>semispan of an aerodynamic surface including the body radius, feet</td>
</tr>
<tr>
<td>$C_D$</td>
<td>total drag coefficient</td>
</tr>
<tr>
<td>$C_{D_b}$</td>
<td>base drag coefficient</td>
</tr>
<tr>
<td>$C_{D_c}$</td>
<td>crossflow drag coefficient</td>
</tr>
<tr>
<td>$C_{D_f}$</td>
<td>friction drag coefficient</td>
</tr>
<tr>
<td>$C_{D_i}$</td>
<td>induced drag coefficient</td>
</tr>
<tr>
<td>$C_{D_o}$</td>
<td>total zero-lift drag coefficient</td>
</tr>
<tr>
<td>$C_{D_p}$</td>
<td>pressure drag coefficient</td>
</tr>
<tr>
<td>$C_{D_v}$</td>
<td>wave drag coefficient</td>
</tr>
<tr>
<td>$C_f$</td>
<td>incompressible skin-friction coefficient</td>
</tr>
<tr>
<td>$C_{f_c}$</td>
<td>compressible skin-friction coefficient</td>
</tr>
<tr>
<td>$C_{D_{FP}}$</td>
<td>drag of a flat plate normal to the flow</td>
</tr>
<tr>
<td>$C_L$</td>
<td>total lift coefficient</td>
</tr>
<tr>
<td>$C_{L_{\alpha}}$</td>
<td>lift curve slope, per radian</td>
</tr>
<tr>
<td>$C_m$</td>
<td>total longitudinal pitching moment coefficient</td>
</tr>
<tr>
<td>$C_r$</td>
<td>root-chord of an aerodynamic surface, feet</td>
</tr>
<tr>
<td>$C_t$</td>
<td>tip-chord of an aerodynamic surface, feet</td>
</tr>
<tr>
<td>d</td>
<td>diameter of the body at any station, feet</td>
</tr>
<tr>
<td>$d_B$</td>
<td>base diameter of the body, feet</td>
</tr>
<tr>
<td>$d_N$</td>
<td>diameter of the nose at the nose-body juncture, feet</td>
</tr>
<tr>
<td>f</td>
<td>spanwise location of the vortex which emanates from the forward surface, feet</td>
</tr>
<tr>
<td>$h_A$</td>
<td>height of the trailing vortex above the body centerline at the aft surface center of pressure, feet</td>
</tr>
<tr>
<td>i</td>
<td>downwash interference constant</td>
</tr>
</tbody>
</table>
NOMENCLATURE
(continued)

\( k_2 - k_1 \) apparent mass factor

K linear lift interference factor due to angle of attack

\( K' \) linear lift interference factor due to control surface deflection

\( l_B \) total length of the body, feet

\( l_N \) length of the nose, feet

\( l_{\text{REF}} \) arbitrary reference length, usually the maximum body diameter, feet

\( l_T \) distance from the tip of the nose to the intersection of the tail leading edge with the body, feet

\( l_W \) distance from the tip of the nose to the intersection of the wing leading edge with the body, feet

M free-stream Mach number

m cotangent of the leading edge sweep angle

r radius of the body at any station, feet

Re Reynolds number

\( S_B \) base area of the body, \((\text{feet})^2\)

\( S_F \) exposed planform area of one pair of forward lifting surfaces, \((\text{feet})^2\)

\( S_N \) body cross-sectional area at the nose, body juncture, \((\text{feet})^2\)

\( S_P \) planform area of the body, \((\text{feet})^2\)

\( S_S \) surface area of the body, \((\text{feet})^2\)

\( S_T \) exposed planform area of one pair of tail surfaces, \((\text{feet})^2\)

\( S_T' \) planform area of one pair of tail surfaces as obtained by extending the leading and trailing edges to the centerline of the body, \((\text{feet})^2\). See Figure 2.
NOMENCLATURE
(continued)

$S_W$ exposed planform area of one pair of wings, $(\text{feet})^2$

$S_W'$ planform area of one pair of wings as obtained by extending the leading and trailing edges to the centerline of the body, $(\text{feet})^2$. See Figure 2.

$V_B$ volume of the body, $(\text{feet})^3$

$X_{CG}$ distance from the nose to the missile center of gravity, $(\text{feet})^2$

$X$ distance to the surface center of pressure as measured from the intersection of the leading edge of the aerodynamic surface with the body, feet

$X_{CP}$ distance from the nose to the center of pressure location, feet

$X_h$ distance from the intersection of the panel leading edge and the body to the hinge line, feet

$X_p$ distance from the nose to the centroid of the body planform area, feet

$\alpha$ missile angle of attack, degrees

$\beta$ compressibility factor, $\sqrt{M^2-1}$

$\delta$ control surface deflection, degrees (See Figure 1 for sign conventions)

$\Delta C_D$ component of the induced drag coefficient

$\Delta C_{D_o}$ increment of wave drag for the transonic speed regime

$n$ ratio of the drag coefficient of a circular cylinder of finite length to that of infinite length

$\theta_N$ conical nose semi-vortex angle, degrees

$\lambda$ lifting surface taper ratio, $C_t/C_r$

$\Lambda$ leading edge sweep angle, degrees

$\Lambda_{c/4}$ sweep angle of the quarter chord line, degrees
NOMENCLATURE  
(continued)  

SUBSCRIPTS  

A  
aft lifting surface, alone  
B  
body alone  
BT  
body in the presence of the tail  
BT - α  
body in the presence of the tail due to angle of attack  
BT - δ  
body in the presence of the tail due to control surface deflection  
BW  
body in the presence of the wing  
BW - α  
body in the presence of the wing due to angle of attack  
F  
forward surface alone  
FB  
forward surface in the presence of the body  
N  
nose  
T  
tail alone  
T - α  
tail alone due to angle of attack  
T - δ  
tail alone due to control surface deflection  
TB  
tail in the presence of the body  
TB - α  
tail in the presence of the body due to angle of attack  
TB - δ  
tail in the presence of the body due to control surface deflection  
TV  
tail, nonlinear component  
W  
wing alone  
WB  
wing in the presence of the body  
WB - α  
wing in the presence of the body due to angle of attack  
WV  
wing, nonlinear component  

The control surface is defined as the tail regardless of the mode of control; the fixed surface is defined as the wing (see Figure 1).  

viii
INTRODUCTION

Increasing maneuverability requirements of missiles indicated a need for predicting the aerodynamic characteristics, including lift, drag, and pitching moment, of missile configurations to angles of attack of 90 degrees and higher. A study showed that existing methods for computing these aerodynamic characteristics are based on a number of different theories all of which are applicable only to small angles of attack. To fulfill the high angle of attack requirements, a method for determining the aerodynamic characteristics of low aspect-ratio configurations at zero roll angles operating at angles of attack up to 180 degrees has been developed. The method is applicable throughout the subsonic, transonic, and supersonic speed regimes up to $\delta_{\text{AR}} \leq 10.0$, and accounts for control surface deflections.

The method is composed of well-known linear, nonlinear crossflow, and slender body theories which have been modified to provide the required high angle of attack capability. These theories can be applied to slender bodies of revolution or nose-cylinder bodies with canard, wing, or tail controls (Figure 1).

This report describes the methods developed and the computer program which has been written for use on the IBM 7090 digital computer. The description of the method is divided into three parts: lift, drag, and pitching moment. For the sake of clarity, the description of the method is kept to a minimum, without lengthy justification and description of the techniques employed. The reader is referred to the references for detailed descriptions of the various theories. The description of the computer program consists of a brief discussion of the main program and subroutines and complete instructions required for use of the program. Comparisons of theoretical results with experimental data are presented for angles of attack up to 90 degrees over the entire speed range to demonstrate the accuracy of the theories. Some data for a missile configuration at 180 degrees angle of attack is available and is compared with the theoretical results.
LIFT CHARACTERISTICS

The total lift on the missile is the sum of the body lift, the lift due to the aerodynamic surfaces, and the interference lift between the forward and aft surfaces. The lift on the body and aerodynamic surfaces is composed of two components: linear lift including the effects of the body-lifting surface interaction and nonlinear crossflow lift. In general, the crossflow lift component is caused by flow separation which occurs at angle of attack, while the interference component is the lift-loss on the aft lifting surface due to downwash from the forward surface (Reference 1).

Allen, References 2 and 3, developed a method for predicting the total lift on bodies of revolution at angles of attack. This method includes the linear or potential flow component and two nonlinear components: the viscous crossflow force and the viscous axial force. Because the contribution of the axial force component to the body lift is small, it is usually neglected. Allen’s expression for the body lift is

\[ C_{L_B} = (k_2 - k_1) \left( \frac{S_B}{S_{REF}} \right) \sin 2\alpha \cos \frac{\alpha}{2} + \eta C_{dc} \left( \frac{S_P}{S_{REF}} \right) \sin^2 \alpha \cos \alpha - C_D \cos^2 \alpha \sin \alpha \]

where the first term is the linear contribution and the second term is the nonlinear contribution. The apparent mass factor, \( k_2 - k_1 \), and the drag ratio, \( \eta \), can be obtained from Figure 3, while the crossflow drag coefficient, \( C_{dc} \), is obtained from Figure 4. Comparisons of theory with experimental data for numerous bodies of revolution over a wide range of Mach numbers and angles of attack are presented in Reference 3. It should be noted that although this expression for the lift is independent of the nose shape, good agreement with experiment is indicated in Reference 4 for a body with an unusual shape.

The linear lift characteristics of low aspect-ratio lifting surfaces whose cross-sections are thin and symmetrical are generally a function of speed, planform area, and aspect-ratio. When the diameter of the missile body is of the same order of magnitude as the span of the
lifting surfaces, the effects of body-wing and body-tail interactions are significant. Hence, the linear lift of the aerodynamic surfaces is composed of two components: the lift on the surface in the presence of the body, and the added lift on the body due to the presence of a surface. Most low aspect-ratio missile configurations exhibit a nonlinear dependence of lift on angle of attack, especially at the higher angles. One primary cause of this is the crossflow lift component which is due to lateral flow separation and the formation of free vortices on the upper surface. This nonlinear dependence is analyzed in References 1 and 5 and summarized by Eaton in Reference 6.

The expression for the total wing lift based on an arbitrary reference area is

\[ C_{L_W} = C_{L_{WB-\alpha}} + C_{L_{BW-\alpha}} + C_{L_{WV}} \]  

(2)

In order to provide high angle of attack capability, it is necessary to modify both the linear and nonlinear theories. The lift on the wing in the presence of the body, as presented in Reference 2, is a linear function of angle of attack and can be expressed as

\[ C_{L_{WB-\alpha}} = k_{WB} C_{a_w} \left( \frac{S_W}{S_{REF}} \right)^\alpha \]  

(3)

Since the lift force does not vary linearly with angle of attack at high angles, Equation (3) is modified such that the linear lift becomes a function of \( \sin \alpha \) as shown below

\[ C_{L_{WB-\alpha}} = k_{WB} C_{a_w} \left( \frac{S_W}{S_{REF}} \right) \sin \alpha \]  

(4)
This component of the linear lift is modified further to satisfy the end condition of zero lift at 90 degrees angle of attack. The resulting expression for the linear wing lift in the pressure of a body is

$$C_{L_{WB=\alpha}} = K_{WB} C_{L_{\alpha}} \left( \frac{S_w}{S_{REF}} \right) \sin \alpha \cos \alpha$$

(5)

It is important to note that for small angles of attack the modified theory should be very close to the method of Reference 1 since $\sin \alpha \approx \alpha$ and $\cos \alpha \approx 1$. Similarly, the additional lift on the body due to the presence of the wing is

$$C_{L_{BW=\alpha}} = K_{BW} C_{L_{\alpha}} \left( \frac{S_w}{S_{REF}} \right) \sin \alpha \cos \alpha$$

(6)

The parameters, $K_{WB}$ and $K_{BW}$, are determined from Figure 5. $C_{L_{\alpha}}$ is obtained from Figure 6 by multiplying $C_{L_{\alpha}}$ by the aspect-ratio $\frac{C_{L_{\alpha}}}{AR}$ if $AR < 1.0$. When the aspect ratio is greater than one, the lift-curve slope is obtained from the following equation

$$C_{L_{\alpha}} = \left( \frac{1}{AR(AR - 1)/AR} \right) \left( \frac{C_{L_{\alpha}}}{AR} \right) AR$$

(7)

where $\left( \frac{C_{L_{\alpha}}}{AR} \right)$ is obtained from Figure 6. The first term of Equation (7) is an empirical modification of the lift curve slope for a lifting surface with aspect-ratio greater than 1.
The nonlinear wing lift from Reference 6 is

\[ C_{L_{WV}} = C_{dc} \sin^2 \alpha \left( \frac{S_W}{S_{REF}} \right) \]  

This expression is modified to satisfy the aforementioned end condition with the result being

\[ C_{L_{WV}} = C_{dc} \sin^2 \alpha \left( \frac{S_W}{S_{REF}} \right) \cos \alpha \]  

where \( C_{dc} \) is obtained from Figure 7. It should be noted that the cross-flow drag coefficient is not appreciably affected by Mach number, (References 6 and 7); hence, \( C_{dc} \) is presented independent of Mach number.

The total tail lift is computed basically the same as the wing lift except for the lift due to deflection of the control surfaces. The tail lift is expressed as

\[ C_{L_T} = C_{L_{TB-\alpha}} + C_{L_{BT-\alpha}} + C_{L_{TB-\delta}} + C_{L_{BT-\delta}} + C_{L_{TV}} \]  

where \( C_{L_{TB-\alpha}} \) and \( C_{L_{BT-\alpha}} \) are obtained by applying Equations (5) and (6) to the tail surface. The other three components are:

\[ C_{L_{TB-\delta}} = K_{TB} C_{\alpha_T} \sin \delta \left( \frac{S_T}{S_{REF}} \right) \cos (\alpha + \delta) \]  

\[ C_{L_{BT-\delta}} = K_{BT} C_{\alpha_T} \sin \delta \left( \frac{S_T}{S_{REF}} \right) \cos (\alpha + \delta) \]  

\[ C_{L_{TV}} = C_{dc} \sin^2 (\alpha + \delta) \left( \frac{S_T}{S_{REF}} \right) \cos (\alpha + \delta) \]
The parameters, $K_{BT}'$ and $K_{TB}'$, are obtained from Figure 5, while $C_{dc}$ and $C_L$ are obtained as specified for the wing. Notice that the nonlinear lift is based on the local angle of attack, $(\alpha + \delta)$, of the control surface.

The lift-loss on the aft surface due to downwash from the forward surface is obtained from the method presented in Reference 1 and discussed in Reference 7. Since the method for computing this component of the total lift on the missile is both complex and lengthy, only the equations necessary to compute the lift-loss are presented. The reader is referred to Reference 1 for a detailed discussion of the assumptions and technique used in deriving the method. It is noted that the nomenclature used to describe the lifting surfaces is changed from wing and tail to forward and aft surfaces. This is necessary because the control surface, whether it be wing, canard, or tail type of control, is designated the tail and because the aft surface, regardless of the mode of control, is the one which is affected by downwash.

This method is valid for the entire speed range. The lift-loss due to downwash is

$$C_{L_i} = \frac{C_L}{\alpha_F} \frac{C_L}{\alpha_A} \left[ K_{FB} \sin \alpha + K_{FB}' \sin \delta_F \right] i(b - r)_A \frac{S_F}{S_{REF}}$$

This equation is obtained from line-vortex theory assuming only one trailing vortex per forward panel exists (see Figure 8). The lateral location, $f_F$, and the vertical location, $h_A$, of the vortex are required for use in Equation (14) and to compute the interference factor, $i$. The lateral location of the vortex on the forward surface expressed as a fraction of the exposed semispan of this surface is

$$\left(\frac{f - r}{b - r}/_F\right) = \frac{\pi - \frac{\pi}{4}(r/b)_F^2 - (r/b)_F^2 + \left[ \frac{1 + (r/b)_F^2}{2 \left[ 1 - (r/b)_F^2 \right]} \right]}{2 \left[ 1 - (r/b)_F^2 \right]} \sin^{-1} \left[ \frac{1 - (r/b)_F^2}{1 + (r/b)_F^2} \right]$$

$$2 \left[ 1 - (r/b)_F^2 \right]$$

(15)
For convenience, the right hand side of this equation is defined as $A'$. Isolating $f_F$ results in the following expression

$$f_F = A'(b - r)_F + r_F$$

where $f_F$ is the spanwise location of the vortex at the forward surface. Since the lateral location of the vortex with respect to the body axis is unchanged, the subscript may be dropped and Equation (16) may be written as

$$f = A'(b - r)_F + r_F$$

The vertical location of the vortex, $h_A$, is measured normal to the body axis at the center of pressure of the aft surface. The expression for $h_A$ is

$$h_A = (C_r - X_h) F sin \delta_F + \left[ l_A + (\bar{X}_{CP} A)_F - l_F - (C_r)_F \right] sin \alpha \quad (18)$$

Note that the vertical location of the vortex is a function of both angle of attack and the deflection angle of the forward surface. The interference factor, $i$, is given by

$$i = \left( \frac{2}{1+\lambda} \right) \left[ L \left( \frac{\lambda, r, f, h}{b, b, b} \right) - L \left( \frac{\lambda, r, f, h}{b, b, b} \right) - L \left( \frac{\lambda, r, f, h}{b, b, b} \right) + L \left( \frac{\lambda, r, f, h}{b, b, b} \right) \right] \quad (19)$$

where

$$L \left( \frac{\lambda, r, f, h}{b, b, b} \right) = \left\{ \frac{(b - r\lambda) - f(1 - \lambda)}{2(b - r)} \ln \left( \frac{h^2 + (f - b)^2}{b^2 + (f - r)^2} \right) \left( \frac{1 - \lambda}{b - r} \right) \left( b - r \right) + \right.$$  

$$\left. h\tan^{-1} \left( \frac{f - r}{h} \right) - h\tan^{-1} \left( \frac{f - r}{h} \right) \right\} \quad (20)$$

and

$$f_i = \frac{f x_i^2}{f^2 + h^2} \quad , \quad h_i = \frac{h r_i^2}{f^2 + h^2} \quad (21)$$
Hence, once the location of the vortex is determined and the position of the image vortices, \( f_1 \) and \( h_1 \), is determined from Equation (21), the interference factor can be computed from Equations (19) and (20). The lift-loss due to interference can then be calculated.

The total lift based on an arbitrary reference area is obtained by adding the components

\[
C_L = C_{LB} + C_{LW} + C_{LT} + C_{LI}
\]  

(22)

where \( C_{LI} \), a lift-loss, will be a negative quantity. The other three components include both linear and nonlinear contributions.

**DRAG CHARACTERISTICS**

The total aerodynamic drag acting on a missile is the sum of the zero-lift drag, the induced drag due to angle of attack and/or control surface deflection, and the base pressure drag. It is well-known that the selection of the proper technique for computing the zero-lift drag is determined by the operating speed of the missile. Hence, the methods employed to compute the zero-lift drag of a missile are described for three speed regimes. These speed regimes and their associated limits are defined as follows:

1. **Subsonic** -- \( M < 0.8 \)
2. **Transonic** -- \( 0.8 \leq M \leq 1.2 \)
3. **Supersonic** -- \( M > 1.2 \)

The description of \( C_{D0} \) calculations for these speed regimes is followed by a description of the method used to compute the induced drag due to angle of attack and/or tail deflection. The last section presents the total drag.
1. Subsonic Region

The zero-lift drag for subsonic speeds can be expressed as

\[ C_{D_0} = C_{D_{o_B}} + C_{D_{o_W}} + C_{D_{o_T}} \] (23)

where each of the components is composed of skin-friction drag and pressure-drag. The pressure drag at subsonic speeds is usually small compared to the drag due to skin friction. Since the flow around high speed missiles of the type being considered here is primarily turbulent, the methods employed are developed assuming the existence of fully turbulent boundary layers.

The zero-lift drag of the body based on an arbitrary reference area is obtained from Section 4.2.3.1 of Reference 9 and can be expressed as

\[ C_{D_{o_B}} = 1.02 \, C_f B \left[ 1 + \frac{1.5}{(l_B/d_B)^{3/2}} + \frac{7}{(l_B/d_B)^3} \right] \frac{S_s}{S_{REF}} \] (24)

where \( C_f \), the skin-friction coefficient, is determined using Figure 9.

The wing zero-lift drag as presented in Section 4.1.5.1 of Reference 9 is

\[ C_{D_{o_W}} = 8.0 \, C_f W \left[ 1 + 2(t/c) + 100(t/c)^{1/4} \right] \frac{S_w}{S_{REF}} \] (25)

The factor of 8.0 is included to account for the total wetted area of the surfaces and for the existence of four wings. The tail zero-lift drag, \( C_{D_{o_T}} \), can be obtained by using the tail thickness-to-chord ratio and the total tail area, \( S_T' \), in Equation (25).

These three components, based on the same arbitrary reference area, are added as indicated by Equation (23) to give the total subsonic zero-lift drag.
2. Transonic Region

The body drag in the transonic speed regime is composed of compressible skin-friction drag, subsonic pressure drag, and transonic wave drag and is obtained using the techniques presented in Section 4.2.3.1 of Reference 9. The compressible skin-friction drag is obtained using the following equation:

\[ C_{Df_B} = 1.02 \frac{C_{f_c} S_T}{S_{REF}} \]  \hspace{1cm} (26)

where \( C_{f_c} \) is a function of both Reynolds number and Mach number and can be determined from Figures 9 and 10. The subsonic pressure drag as extracted from Equation (24) is

\[ C_{Dp_B} = 1.02 \frac{C_{f_B} \left[ \frac{1.5}{(\frac{L}{d})^{3/2}} + \frac{7}{(\frac{L}{d})^3} \right] S_T}{S_{REF}} \]  \hspace{1cm} (27)

The transonic pressure drag is computed using Equation (27) up to a Mach number of 1.0 and then it is decreased linearly from its value at 1.0 to zero at \( M=1.2 \).

The transonic wave drag of the body is obtained from Figure 11 which presents the wave drag as a function of the nose fineness ratio and Mach number. This figure was constructed from experimental data presented in Reference 10 and by using the curves of wave drag for bodies of revolution as shown in Reference 11 and reproduced here in Figure 12.

The total transonic body zero-lift drag is obtained form the following expression:

\[ C_{D_{OB}} = C_{Df_B} + C_{Dp_B} + C_{Dv_B} \frac{S_N}{S_{REF}} \]  \hspace{1cm} (28)
Experimental results show little increase in the viscous drag of the aerodynamic surfaces from the subsonic to the transonic regime and therefore, the skin-friction drag for the subsonic region is also used in the transonic regime. It may be expressed as follows:

\[
C_{D_{\text{fW}}} = 8.5 C_{f_{W}} \left[ 1 + 2 \left( \frac{t}{c} \right) \right] \tag{29}
\]

To this wing transonic skin-friction drag is added a drag increment, \( \Delta C_{D_{\text{w}}} \), which is the transonic wave drag of the wing surfaces. Figure 13 expresses this component as a function of thickness-to-chord ratio, \( \frac{t}{c} \), aspect-ratio, and Mach number for rectangular surfaces. For surfaces having swept leading edges, \( \Delta C_{D_{\text{w}}} \) is obtained as for rectangular surfaces and then adjusted to account for the sweep angle using the following equation:

\[
\Delta C_{D_{\text{w}}} = \Delta C_{D_{\text{w}}}^{'} \left[ \cos \frac{\Lambda}{c/4} \right]^{2.5} \tag{30}
\]

The Mach number used in Figure 13 to obtain \( \Delta C_{D_{\text{w}}}^{'} \) for swept lifting surfaces is

\[
M' = M \left[ \cos \frac{\Lambda}{c/4} \right]^{1/2} \tag{31}
\]

This component of the drag due to the aerodynamic surfaces must be computed for both the wings and tails. The tail contribution is determined in the same manner as the wing contribution above using tail parameters.
The total transonic zero-lift drag is the sum of these components as shown below:

\[
C_{D_0} = \left(C_{D_{0W}} + \Delta C_{D_{0W}}\right) \frac{S_{W}'}{S_{REF}} + \left(\Delta C_{D_{0T}} + C_{D_{0T}}\right) \frac{S_{T}'}{S_{REF}} + C_{D_{0B}} \quad (32)
\]

3. Supersonic Region

A simple empirical method for computing the zero-lift drag for the supersonic speed regime has been developed by assuming a parabolic variation of \(C_{D_0}\) with Mach number between 1.2 and 3.0. The resulting equation which is used to compute the zero-lift drag of a missile for Mach numbers greater than 1.2 is

\[
C_{D_0} = \left[C_{D_0}'' - C_{D_0}' \right] \sqrt{M} + \left[C_{D_0}'' - C_{D_0}' \right] \frac{1}{1 - \sqrt{3/1.2}} + C_{D_0}' \quad (33)
\]

where \(C_{D_0}'\) and \(C_{D_0}''\) are the values of the total zero-lift drag at Mach numbers 1.2 and 3.0 respectively. It should be noted that although \(C_{D_0}\) for Mach numbers greater than 3.0 can be determined from Equation (33), existing hypersonic flow theories would probably provide a more accurate estimate of the zero lift drag.

In order to utilize Equation (33) for determining the variation of \(C_{D_0}\) with Mach number, \(C_{D_0}'\) and \(C_{D_0}''\) must be specified. \(C_{D_0}'\) is determined by using the techniques described in the previous section for the transonic flow regime. Since the magnitude of the supersonic wave drag is heavily dependent on the nose shape of the missile, \(C_{D_0}''\) is determined using one of two methods; the selection of the proper method depends on the missile forebody shape.
In the first method, which is for blunted ogives, pointed ogives, and blunted cones, the supersonic zero-lift drag is a function of Mach number and nose fineness ratio. Since \( C_D \) is the zero-lift drag at \( M=3.0 \), it remains to define its variation with \( \) nose fineness ratio. Hoerner, Reference 12, indicates the zero-lift drag for body-fin configurations with slender (high fineness ratio) nose shapes generally peaks at a Mach number of 1.0 to 1.2 and then decreases to approximately its subsonic value plus the transonic wave drag of the lifting surfaces at \( M=3.0 \). Similar configurations with blunted nose shapes of low fineness ratio reach a peak at about the same Mach number, but decrease very little as the Mach number is increased. Using these trends as an indication of the effect of fineness ratio on the variation of zero-lift drag with Mach number for the supersonic speed regime, \( C_D \) for the aforementioned nose shapes is specified as follows:

a) For \((1/d)_N < 0.5\), \( C_D'' = C_D' \)

b) For \((1/d)_N > 8.0\), \( C_D'' = C_D'/M=0.8 + \left( \Delta C_D_{W} \frac{S_W}{S_{\text{REF}}} + \Delta C_D_{T} \frac{S_T}{S_{\text{REF}}} \right) \)

where \( C_D'' \) at \( M=0.8 \) is determined utilizing subsonic flow theory and \( \Delta C_D_{W} \) and \( \Delta C_D_{T} \) are the transonic wing and tail wave drag. The variation of the forebody wave drag as a function of Mach number is presented in Figure 12 and was used to construct Figure 14 which is utilized with the following equation:

\[
C_D'' = K_1 C_D'
\]

(34)

to compute \( C_D'' \) for nose fineness ratios between 0.5 and 8.0.
In the second method, which is for pointed conical noses, $C_{D_0}''$ is determined from the following equation:

$$
C_{D_0}'' = C_{D_0}/M=0.8 + \left( \Delta C_{D_0} \frac{S_W}{S_{REF}} + \Delta C_{D_0} \frac{S_T}{S_{REF}} \right) + C_{D_0}'' \quad (35)
$$

where the first two terms are obtained in the same manner described above. The forebody wave drag, which for this type of nose is a function of Mach number and the cone semivertex angle, is obtained from Figure 15.

Once $C_{D_0}''$ is determined, Equation (30) can be used to compute the zero-lift drag at any Mach number between 1.2 and 3.0. The complete zero-lift drag curve of a missile can now be determined.

4. Induced Drag

The induced drag due to angle of attack and/or tail surface deflection is composed of four drag increments as shown below:

$$
C_{D_1} = \Delta C_{D_{B-a}} + \Delta C_{D_{W-a}} + \Delta C_{D_{T-A}} + \Delta C_{D_{T-\delta}} \quad (36)
$$

The drag increment for the body is obtained using a method presented in Reference 3, while the induced drag due to the wing and tail is obtained from the drag of an equivalent flat plate normal to the flow. The induced drag on a body of revolution at angle of attack can be expressed as follows:

$$
\Delta C_{D_{B-a}} = (k_2 - k_1) \frac{S_B}{S_{REF}} \sin^2 \alpha + n_{dc} \frac{S_p}{S_{REF}} \sin^3 \alpha \quad (37)
$$
The wing induced drag based on an arbitrary reference area is obtained from

\[ \Delta C_{D_{W-\alpha}} = C_{D_{FP}} \left( \frac{S_W \sin \alpha}{S_{REF}} \right) \]

(38)

where \( C_{D_{FP}} \) is the drag of a flat plate normal to the flow field, and is determined from Figure 16. The drag curve shown in Figure 16 has been constructed using the three-dimensional subsonic drag coefficient for a flat plate normal to the flow (Reference 13) and the variation of the drag coefficient with Mach number for the two-dimensional flat plate (Reference 12). The equivalent flat plate area of the wing is taken as the projection of the wing area on the normal plane, \( S_W \sin \alpha \). Similarly, the drag increment of the tail surface at angle of attack is

\[ \Delta C_{D_{T-\alpha}} = C_{D_{FP}} \left( \frac{S_T \sin \alpha}{S_{REF}} \right) \]

(39)

The drag increment due to deflection of the tail surface cannot be obtained directly since the drag increment is not a linear function of local angle of attack. The total drag increment for the tail surface can be expressed as

\[ \Delta C_{D_{T-\alpha}} + \Delta C_{D_{T-\delta}} = C_{D_{FP}} \left[ \frac{S_T \sin(\alpha + \delta)}{S_{REF}} \right] \]

(40)

where \((\alpha + \delta)\) is the local tail angle of attack. It now becomes a simple matter to obtain the drag increment due to the deflection of the control surface by taking the difference between Equations (39) and (40). Thus

\[ \Delta C_{D_{T-\delta}} = \left( \Delta C_{D_{T-\alpha}} + \Delta C_{D_{T-\delta}} \right) - \Delta C_{D_{T-\alpha}} \]

(41)
6. Total Drag

The total drag can be expressed as follows:

\[ C_D = C_{D_0} + C_{D_1} \]  \hspace{1cm} (42)

where \( C_{D_0} \) is determined using Equation (36) and \( C_{D_1} \) is obtained from Equation (23), (32), or (33) depending on the speed of the missile. Base pressure drag is not included in Equation (42).
PITCHING MOMENT CHARACTERISTICS

The total pitching moment acting on the missile is the sum of the moments due to the lift and drag forces acting on the body, wings, and tails. Most methods for computing the pitching moment (References 1, 6, and 9) consider only the moment due to lift. This is valid only for small angles of attack. If large angles of attack are to be considered, the moment must include the drag contribution. In general, the body longitudinal pitching moment is determined directly; while the other components are determined only after the centers of pressure of the wing and tail surfaces are specified.

The body-alone pitching moment about its center of gravity is obtained from the method of Allen, References 2 and 3. The expression is

\[ C_{m_B} = (k_2 - k_1) \left( \frac{V_B - S_B \left( \frac{1_B - X_{CG}}{l_{REF}} \right)}{S_{REF} l_{REF}} \right) \sin 2\alpha \cos \frac{\alpha}{2} \]

\[ + \eta C_{dc} \left( \frac{S_P}{S_{REF}} \right) \left( \frac{X_{CG} - X_P}{l_{REF}} \right) \sin^2 \alpha \]

(43)

where \((k_2 - k_1), C_{dc}\), and \(\eta\) are determined from Figures 3 and 4. This moment coefficient, \(C_{m_B}\), is based on an arbitrary reference length and area.

As noted above, the center of pressure locations for the wing and tail surfaces must be specified before their pitching moments can be determined. It must be remembered that the linear lift of the aerodynamic surfaces is composed of two components: the lift on the surface in the presence of the missile body and the additional lift on the body due to the presence of a lifting surface. This means that in order to determine the linear pitching moment caused by the lifting surfaces, it is necessary to specify the center of pressure location for each of these linear components.

The center of pressure of the lift on the wing in the presence of the body as measured from the junction of the wing leading edge and the
body is obtained using Figure 17. The reference point is transferred to the nose by using the following equation

\[ X_{WB} = \left( \frac{X}{C_r} \right)_{WB} \left( C_r \right)_W + l_W \]  

(44)

where \( \left( \frac{X}{C_r} \right)_{WB} \) is obtained from Figure 17a for subsonic speeds and from Figure 17b for supersonic speeds.

The center of pressure of the additional lift on the body in the presence of the wing is obtained using Figure 18 if the flow is subsonic, and either Figure 19 or 20 if the flow is supersonic. For the case of supersonic flow, Figure 19 is used if

\[ \text{BAR} \left( 1 + \lambda \right) \left( 1 + \frac{1}{m^B} \right) \leq 4.0 \]

and Figure 20 is used if the above quantity is greater than 4.0. The center of pressure, \( X_{BW} \), as obtained from the aforementioned figures, is referred to the nose by using the following equation.

\[ \left( X_{CP} \right)_{BW} = \left( \frac{X}{C_r} \right)_{BW} \left( C_r \right)_W + l_W \]  

(45)

The location of both centers of pressure for the tails, \( X_{CP} \)_{TB} and \( X_{CP} \)_{BT}, can also be obtained from the above procedure.

The centers of pressure for a given lifting surface are combined to obtain a single average center of pressure location for each set of aerodynamic surfaces. For example, the average center of pressure of the wings is obtained by computing the total pitching moment due to the wings and dividing by the wing normal force. The pitching moment about the
nose of the body due to the wing is

\[
C_{m_w} = \left[ \left( C_{L_{WB}} + C_{L_{W}} + C_{L_i} \right) \cos \alpha + \Delta C_{D_{w-a}} \sin \alpha \right] \left[ (X_{CP})_{WB} \right]_{1 \text{REF}} \\
+ \left[ C_{L_{BW}} \cos \alpha \right] \left[ (X_{CP})_{BW} \right]_{1 \text{REF}}
\]

(46)

where all of the above terms have been previously defined. The average wing center of pressure as measured from the nose of the body can now be defined as

\[
(X_{CP})_w = \frac{C_{m_w \cdot 1 \text{REF}}}{\left( C_{L_w} + C_{L_i} \right) \cos \alpha + \Delta C_{D_{w-a}} \sin \alpha}
\]

(47)

Similarly, the tail pitching moment and center of pressure including the effect of surface deflection can be expressed as

\[
C_{m_T} = \left[ \left( C_{L_{TB-a}} + C_{L_{TV}} + C_{L_i} \right) \cos \alpha + C_{L_{TB-\delta}} + \\
+ \left( \Delta C_{D_{T-a}} + \Delta C_{D_{T-\delta}} \right) \sin \alpha \right] \left[ (X_{CP})_T \right]_{1 \text{REF}} + \left[ C_{L_{BT-a}} \cos \alpha \right] (X_{CP})_{BT} \left[ (X_{CP})_T \right]_{1 \text{REF}}
\]

\[
(X_{CP})_T = \frac{C_{m_T \cdot 1 \text{REF}}}{\left( C_{L_{T-a}} + C_{L_i} \right) \cos \alpha + C_{L_{T-\delta}} + \left( \Delta C_{D_{T-a}} + \Delta C_{D_{T-\delta}} \right) \sin \alpha}
\]

(49)

where

\[
C_{L_{T-a}} = C_{L_{TB-a}} + C_{L_{BT-a}} + C_{L_{TV}}
\]

and

\[
C_{L_{T-\delta}} = C_{L_{TB-\delta}} + C_{L_{BT-\delta}}
\]
The total longitudinal pitching moment of the missile about the missile's center of gravity may be expressed as

\[
c_m = c_{m_B} + c_{m_w} \left[ \frac{x_{CG} - (x_{CP})_W}{(x_{CP})_W} \right] + c_{m_t} \left[ \frac{x_{CG} - (x_{CP})_T}{(x_{CP})_T} \right] \tag{50}
\]
COMPUTER PROGRAM DESCRIPTION

The method presented herein for obtaining the static aerodynamic characteristics of a missile has been programmed for use on the IBM 7090 computer and other compatible digital computers. The program, Table 1, is written in Fortran II and requires only the geometric characteristics of the missile and its flight conditions as inputs. The output consists of the static longitudinal aerodynamic characteristics in coefficient form, the center-of-pressure location for the body, wings, and tails, the lift-curve slope for each set of lifting surfaces, and the components of the lift and normal force coefficients. The force components referred to are the body, wing, and tail lift and normal force coefficients and the coefficient representing the lift-loss due to downwash. The aerodynamic characteristics are output in both the stability and body axis systems (Figure 21). Provision has been made for a third lifting surface to account for the possibility of using strakes in combination with two other sets of lifting surfaces (Figure 1).

The program itself consists of a main program and three subroutines—GEOSUB, CLASUB, and CATSUB. The first subroutine performs some initial geometric computations, determines the nose wave drag constant (Figure 14), and obtains the Reynolds number per foot based on the altitude input to the program. Subroutine CLASUB determines the lift-curve slope of the lifting surfaces from curve fits employed to represent the curves presented in Figure 6. The last subroutine, CATSUB, obtains the body-wing and body-tail interference factors, computes the center of pressure location as a function of the root chord of the lifting surfaces, and determines the crossflow drag coefficient for the lifting surfaces.

The program computes the static force and moment coefficients for typical missile configurations at specified angles of attack, control surface deflection angles and Mach numbers. The angle of attack range is -180° to +180°, and any control surface deflection within this range may be used. The Mach number is limited to 3.0 only because the drag prediction methods are valid up to this particular Mach number. The program can be used for configurations at M > 3.0; however, the drag predictions above this limit should be used with caution. The computer
program can be used to obtain build-up information; that is, the aero­
dynamic characteristics of the missile body alone, the body-wing configu­
ration, and the body-tail configuration. This information may be ob­
tained by simply setting the appropriate parameters to zero.

The inputs to the computer program with their FORTRAN symbols are
presented in Table 2. The format for preparing the input cards is pre­
sented in Table 3. It should be noted that if a configuration does not
have a control surface, e.g. a body alone configuration, the number of
control surface deflection angles should be set at one and the deflec­
tion angle, itself, would be 0.0 degrees. Cards 11 and 12 are used
only when the number of angles of attack require their use. The output
variables are defined in Table 4. There is no limit to the number of
data decks which may be stacked together and run at the same time.

COMPARISON OF THEORY WITH EXPERIMENTAL DATA

Numerous comparisons of theory with experiment have been made in
order to establish and verify the accuracy of the method. Figure 22
presents the configurations used for comparison. Configuration 1 is a
strake-tail configuration. Figure 23 presents the comparison between
the experimental data and the theoretical results obtained from the
method described herein. The theoretical lift and drag are within
15 percent of the experimental data for $M = 0.7$. The deviation for
$M = 1.1$ does increase to about 25 percent. However, it should be noted
that the accuracy of the method to 30 degrees is good. The method also
predicts the center of pressure location very satisfactorily.

Configuration 2 is a wing-controlled vehicle. Comparisons are
presented in Figure 24 for Mach numbers of 1.12 and 2.16 and for wing
deflection angles of $0$ and $-10$ degrees. Although high angle of attack
data is not available for this configuration, it appears that the com­
paratively high theoretical lift shown by Configuration 1 may be more
pronounced for Configuration 2. This trend is seen by the decrease in
the slope of the experimental lift data at 25 to 30 degrees angle of
attack, Figure 24. The comparison with the wing deflected down 10 de­
grees appeared to be good.
Data for Configuration 3 is presented in Figure 25. Configuration 4 was tested by the Cornell Aeronautical Laboratory to 180 degrees angle of attack, Reference 14. Comparison of theory with experimental data is presented in Figure 26.

CONCLUSIONS

A method for predicting the static, longitudinal aerodynamic characteristics of low aspect-ratio missiles operating at angles of attack to 180 degrees has been developed. The method is valid for a wide speed range and considers control surface deflections. A computer program, written to facilitate use of the method, has been described. Results obtained using the method have been compared with wind tunnel data and acceptable agreement has been demonstrated.
REFERENCES


Figure 1a - Canard Control

Figure 1b - Wing Control

Figure 1c - Tail Control

Figure 1d - Three Surfaces - Any Mode of Control

Figure 1 - Typical Missile Configurations
Figure 2 - General Geometric Characteristics
Figure 3 - Parameters Used to Compute Body Normal Force and Pitching Moment (from Reference 3)
Figure 4 — Crossflow Drag Coefficient as a Function of Mach Number (from Reference 3)
Figure 5 - Linear Lift Interference Factors (from Reference 7)

Figure 5a - \((\beta AR) (1 + \lambda) (1/m\beta + 1) \leq 4\)
Figure 5b - \((\beta \Delta R)(1 + \lambda)(1/m\beta + 1) > 4\); with Afterbody
Figure 5c: $(\beta A_R)(1 + \lambda)(1/m\beta + 1) > 4; \text{ No Afterbody}$
Figure 6 - Lift Curve Slope for Wings and Tails (from Reference 8)

Figure 6a - Unswept Trailing Edge
Figure 6b - Unswept Mid Chord
Figure 7 - Crossflow Drag Coefficient for Wings and Tails as a Function of Aspect Ratio and Taper Ratio
Figure 8 - Vortex Model Used to Determine the Lift Loss due to Downwash (from Reference 1)
Figure 9 - Incompressible Skin Friction Coefficient (from Reference 9)
Figure 10 — Compressibility Effect on Turbulent Skin Friction (from Reference 9)
Figure 11 - Transonic Wave Drag for Ogival and Blunted Conical Forebodies
Figure 12 – External Wave Drag of Blunt Forebodies (from Reference 11)
Figure 13: Transonic Zero-Lift Wing Wave Drag for Unswept Wings (from Reference 9)
Figure 14 - Ratio of Wave Drag for Noses of Various Fineness Ratios to the Wave Drag for a Hemispherical Nose
Figure 15 - Wave Drag of a Pointed Conical Nose (from Reference 6)
Figure 16 – Drag Coefficient for a Flat Plate Normal to the Flow
Figure 17 - Lifting Surface Center of Pressure as a Function of Effective Aspect Ratio
(from Reference 1)

Figure 17a - $M \leq 1.0$
Figure 17b - $M > 1.0$
Figure 18 – Subsonic Center of Pressure Location of the Lift on the Body in the Presence of Wings or Tails (from Reference 1)

Figure 18a – No Mid Chord Sweep
Figure 18b – No Trailing Edge Sweep
Figure 19 - Supersonic Center of Pressure Location of Lift on the Body in the Presence of Wings or Tails for $\beta\bar{A}(1 + \lambda)(1 + 1/m\beta) \leq 4.0$ (from Reference 1)

Figure 19a - No Mid Chord Sweep
Figure 19b — No Trailing Edge Sweep
Figure 20 – Supersonic Center of Pressure Location of Lift on the Body in the Presence of Wings or Tails for $\beta AR(1 + \lambda)(1 + 1/m\beta) > 4.0$ (from Reference 1)
Figure 21a - Stability Axes

Figure 21b - Body Axes

Figure 21 - Missile Axis Systems
Figure 22 — Configurations Used to Compare Theory with Experiment
Figure 23 – Comparison of Experimental Data with Theoretical Results for Configuration 1

**Lift Coefficient**

- **Maximum Lift Coefficient**: 14.0
- **Minimum Lift Coefficient**: 0.0

**Drag Coefficient**

- **Maximum Drag Coefficient**: 24.0
- **Minimum Drag Coefficient**: 0.0

**Mach No. = 0.7**

- **Angle of Attack (Degrees)**: 0.0 to 90.0

---

**Theoretical Curve**

- **Experimental Points**

---

**Angle of Attack (Degrees)**

- **0.0**
- **10**
- **20**
- **30**
- **40**
- **50**
- **60**
- **70**
- **80**
- **90**
Figure 23 (Continued)

CONFIGURATION 1
MACH NO. = 0.7

CENTER OF PRESSURE LOCATION AS MEASURED FROM THE NOSE OF THE MISSILE (FEET)

ANGLE OF ATTACK (DEGREES)
Figure 23 (Continued)
Figure 23 (Continued)
Figure 24 – Comparison of Experimental Data with Theoretical Results for Configuration 2
Figure 24 (Continued)
Figure 24 (Continued)
Figure 24 (Continued)
Figure 25 — Comparison of Experimental Data with Theoretical Results for Configuration 3
Figure 25 (Continued)
Figure 26 — Comparison of Experimental Data with Theoretical Results for Configuration 4
Table 1

COMPUTER PROGRAM LISTING

DIMENSION XVMX(16),XDT(16),XAL(48)
COMMON XVMX,XDT,XAL
COMMON CN,CNB,CNW,CNT,CMW,CLW,CLTT,CLW2,CLI,CLWB,CLVISW,CLTV,
1 CLW/CNT,CAB,XCP2,XYG2,XCPB,XCPY,XCPW,XCPW2,XCG,XCPWTV,
2 XLAMT4
COMMON LLK,ISWP,ISWPT,ISWP2,IAFBW,IAFBT,IAFBW2,IL,LLL,ILL,II,1
COMMON XLAMW,XLAMTX,XLAMW2,VMACW,VMACXT,VMACW2,CLAMW,CLAMT,CLAMW2,
1 BWBT,BW,CR8BT,CR8BTT,CR8WT,CR8W2,SW,ST,SW2,XWING,XTAIL,XWING2
2 XL,D,D1,XML,AREA,ARF,SSUBS,XMLB,ZFT,ART,ARAW2,XMLN3SE
COMMON CBLAM,BCBLAM,CREBT,R1,CLAM1,XLAM1,BAR,RAT1,BXTB,XTB0
1 XKBW,XKKB,XML,CRM,BKKB,XMLB,XMLB2,XXKBB,XXKBT,XXKBBW,XXKBT2,
2 XCPBW,XCPBT,XCPW2,XCPB2,XXKBBW,XXKBT,XXKBB2,XXKBT2,XB,XXCW,BDCT
3 BCCH2,CLAW,CLALT,CLALT2,REFT,BETA,AL,STNST,STNST,HT,KXTB1,
4 VXM,VXMR,DELT,XXCRW,XXCRW2,XXMAC,XXMAC2,XXMACW,XXMAC2
5 XLAM4,T0VC,TVCT,TVCC,TVCCW,TVCCW2,TVCCW2,TVCCW2,TVCCW2,TVCCW2,
6 CD9,CD9W,CD9T,CD98W,CD9PW,CD9PW2,CD9PW2,CD9PW2,CD9PW2
7 CLDB,CLDBT,CLVIST,CLVIST2,CDBW,CDBW2,CDBW2
COMMON CDALZ
IL = C
304C FORMAT(215F1.5)
305C FORMAT(615)
306C FORMAT(7F10.3)
307C FORMAT(10A6)
311C FORMAT(1H1,10A6)
312C FORMAT(6X,2HHT, 9X,2HD, 8X,2HXL, 6X,6HXL9SE, 5X,3HXC9, 6X,4HAREA, 6X,
14XWREF)
313C FORMAT(//,5X,5HT8VCW,5X,5HT8VCW2,5X,5HT8VCW)
314C FORMAT(3X,15, 5X, 15, 4X, 7F15.6, //)
315C FORMAT(4X,15, 5X,15,5X,15,5X,15,6X,15,5X,15)
324C FORMAT(6X,5ISHPW, 5X,5HIAFBW, 10X,5HXLAMW, 10X,5HCLAMW,
1 10X,5HTV
2 9X,6HCR8BT,10X,5H8TV/10X,5HXLAMW,10X,5HCLAMW)
324C FORMAT(6X,6ISHPW, 4X,5HIAFBW, 9X,6HXLAMW, 9X,6HCLAMW2,
1 10X,5HTV
2 9X,6HCR8BT,10X,5H8TV/9X,6HXLAMW2,9X,6HCLAMW2)
324C FORMAT(6X,5ISHPW, 9X,6HIAFBW, 9X,6HXLAMW, 9X,6HCLAMW,
1 10X,5HTV
2 9X,6HCR8BT,10X,5H8TV/9X,6HXLAMW,9X,6HCLAMW)
3248 FORMAT(6X,5ISHPW, 5X,5HIAFBW, 10X,5HXLAMT, 10X,5HCLAMT,
1 10X,5HTV
2 9X,6HCR8BT,10X,5H8TV/10X,5HXLAMT,10X,5HCLAMT)
3241 FORMAT(//,6X,5ICSC, 5X,5HIN9SE, 5X,5HND85, 5X,5H8TV/5X,5HNLAPH,
1 15X,5HND85)
3333 READ 3010, TITL1,TITL2,TITL3,TITL4,TITL5,TITL6,TITL7,TITL8,TITL9,
1 TITL0
READ 305C, ICSC,IN9SE, IDC, IDT, IDT, IALES, IN8D
READ 304C, ISWP, IAFBW, XLAMT, CLAMT, BW, CR8BT, BW, W, XMACW, XWIN
READ 304C, ISWP2, IAFBW2, XLAMW2, CLAMW2, BW2, CR8BT2, BW2, XMACW2, XWIN
1G2
READ 304C, ISWP, IAFBW, XLAMT, CLAMT, BT, CR8BT, ST, XMACX, XTAIL
READ 302C, HT, CX, XLN9SE, XCG, AREA, XREF
READ 3020, TVCC, TVCC2, TVCT
PRINT 3110, TITL1, TITL2, TITL3, TITL4, TITL5, TITL6, TITL7, TITL8, TITL9, TITL0
PRINT 3241
PRINT 3150, ICSC, IN8E, IDT, IM, IAL, NBBDY
PRINT 3240
PRINT 3140, ISwPW1, IAFBT, XLAMW, CLAMW, BW, CR00TW, SW, XMACW, XWING
PRINT 3244
PRINT 3140, ISwPW2, IAFBT2, XLAMW2, CLAMW2, BW2, CR00W2, SW2, XMACW2, XWING2
PRINT 3248
PRINT 3140, ISwPT, IAFBT, XLAMT, CLAMT, BT, CR00TT, ST, XMCT, XTAL
PRINT 3120
PRINT 3020, HT, D, XL, XL, "SE, XCG, AREA, XREF
PRINT 3021
PRINT 3020, TENVW, TENVW2, TVCT
IL = 1 + IL
LLKK = 0
LLLL = 0
XCG2 = XCG
IZZY = 0
CALL GSUB
READ 4000, [XDT[M], M = 1, IDT]
READ 4000, [XVXM[N], N = 1, IM]
READ 4000, [XAL[NA], NA = 1, [AL]
RE = REF'T * VXM
VXM = XVXM[1]
DS 60C2 [I = 1, I = 1]
DELT A = XDT [1]
DS 60C1 [I = 1, I = 1]
ALPHA = XAL [1]
4CCC FORMAT(16F5.1)
5CCC FORMAT([1H1, 4HVXM = F5.2, 2X, 4HDELTA = F6.2, ]
5CC1 FORMAT([1H1, 4HVXM = F5.2, 2X, 4HDELTA = F6.2, ]
8200 PRINT 5000, VXV, DELTA1
PRINT 50C1
DELTA = DELTA1 / 57.29578 * 100000001
DB 60C0 J = 1, I = 1
AL = ALPHA / 57.29578 * 10000001
1 VXMR1 = VXV
IZZY = IZZY + 1
IF [IZZY = 4] 6666, 6666, 1111
6666 VXV = * 6
1111 CALL CLASUB
IF [LL = 1] 900, 942, 980
9CC IF [IZZY = 4] 66C9, 60C9, 925
925 CALL CATSUB
1[.5[B1-C]]
RE = REFT * VXM * XMAC
IF [RE = 1.E06] 6010, 6020, 6020
6C1C  AA=*CB35
     XNN=*211
     GO TO 6070
6C2C  IF [RE=1*E07] 6030,6040,6040
6C3C  AA=*C52
     XNN=*177
     GO TO 6070
6C4C  IF [RE=1*E08] 6050,6060,6040
6C5C  AA=*C333
     XNN=*1488
     GO TO 6070
6C6C  AA=*C221
     XNN=*127
6C7C  CF=AA*RE*XNN
     CD1=4*CF*[(1+2*TRVC+100*TRVC**4)]
     EXS=([DC]/(8*CBLAM])
     IF [ISWP1=1] 6C80,6080,6090
6C8C  EXS*2=*EXS
6C9C  EXS=[CREAT*D/2.+EXS]*2.
     IF [IZZY=4] 6091,6091,6092
6C91  IF [IZZY=3] 6093,6094,6095
6C92  IF [AL] 2401,2402,2402
     2401 CDD=*6DC
     2402 LLKK=LLKK+1
     IF [LLKK=2] 2403,2404,2420
     2403 IF [SW] 2410,2410,2420
     2410 LLKK=LLKK+1
     2404 IF [SW] 2411,2411,2420
     2411 LLKK=LLKK+1
     2420 IF [LLKK=2] 93C,943,95C
     93C XKBW=XKBW
     XCPW=XING+XBCRW*CR9AT
     XCPW=XING+XBCRWW*CR9AT
     PCC=6DC
86C  C5W=SIN*[AL]*XKBW*XKBW*CLALW*SW*CBS[AL]/AREA
     CLW=SIN*[AL]*XKBW*CLALW*SW*CBS[AL]/AREA
     CLW=SW[AL]*SIN[AL]*SW*CBS[AL]/AREA*6DCW
     C5W=CLW+CLVISW
6C93  CD9=CD9*[SW+EXS]/AREA
     XLAMW4=XLAM14
     T0VCW=T0VC
     SWT9=SW+EXS
     IZZY=IZZY+1
     IF [SW] 542,942,511
     511 CBLAM=CBR[CLAM*2]/SIN[CLAM*2]
     BECLAM=ETA*CBLAM.
     CR9BT=CREBWW2
     B1=BMW2
     IAFB=IAFBW2
     CLAL1=CLALW2
     XLAM1=XLAMW2
     T0VCW=T0VCW2
     XMAC=XMACW2
     ISWP1=ISWPW2

67
BAR=BETA*ARW2
RATIA=CR88T/[BETA*D]
IF [IZZY=4] 6009,6009,925
943 XKBW2=XKWB
XKBW2=XKBW
XCP82=XWING2*XBRCRW2*CR88T
XCP82=XWING2*XBRCRW2*CR88T
6DCW2=6DDC
944 CLW2=SIN(AL) *(XKWBW2+XXBW2)*CLALW2*SW2*COS(AL)/AREA
CLB2=SIN(AL) *(XKWBW2+XXBW2)*CLALW2*SW2*COS(AL)/AREA
CLW2=CLW2+CLW2
CLW2=CLW2+CLW2
CLW2=CLW2+CLW2
6094 CBW2=CD8 *(SW2+EXS)/AREA
XLAMA=XLAM14
XLAMA=XLAM14
SW2*ST=SW2*EXS
542 IZZY=IZZY+1
LLK=LLK+2
IF [ST] 980,980,940
94C CBLAM=COS [CLAMT]/SIN [CLAMT]
ART=(BT=D)+2/ST
RCHLAM=BETA*CBLAM
CRBT=CR88TT
B1=BT
PAR=BETA*BT
CLALT=CLALT
IAFB=IAFBT
XMAC=XMAC
TBVC=TBVCT
ISX1=ISXPT
XLAM=XLAMT
RATIA=CR88T/[BETA*D]
IF [IZZY=4] 60C9,60C9,925
95C XKBW=XKWB
XKBW=XKBW
XCPBT=XYAIL+XBCRW2*CR88T
6DCT=6DC
951 CLBT=[(XKWB+XXKBT)*SIN(AL)]*CLALT*ST*COS(AL)/AREA
CLBT=SIN(AL)*XXKBT*CLALT*ST*COS(AL)/AREA
CLBT=CLBT
CLTB=[XKTB*CLALT*SIN(DELTA)]*ST*COS(AL+DELTA)/AREA
CLTB=[XKTB*XXKBT]*CLALT*SIN(DELTA)*ST*COS(AL+DELTA)/AREA
CLBT=CLBT
CLBT=CLBT
CLVIST=[SIN(AL+DELTA)*SIN(AL+DELTA)]*ST*COS(AL+DELTA)/AREA
6095 CD8=[ST+EXS]/AREA
STBT=ST+EXS
XLAMA=XLAMA
IF [IZZY=4] 1610,1610,6098
6098 XCPBT=XYAIL+[(XKWB+XKBT)*SIN(AL)+XBCRW2*XKTB+SIN(DELTA)]*XBCRW2
IF [XKTB*ST]+[XKTB*SIN(DELTA)]>CR88TT
98C IF [IZZY=4] 1610,1610,1710
171C XLBA=XLBD
ZXM=VXMSQRT(1.5129*1.5129*ZXM*ZXM)
IF [ZXM<8] 1310,1350,1350
131C CXX=2.4*SQRT(1.5129*1.5129*ZXM*ZXM)
68 TO 1391
135C IF ($X^2>1.15$) 1380, 1370, 1370
138C $CDC^1=1.6+SQRT[.344+($X^2-975)^2]$.
      GS TO 1391
137C IF ($X^2>3.$) 1360, 1381, 1381
136C $CDC^1=1.9-SQRT[.361-09*($X^2-3)^2]$.
      GS TO 1391
1381 $CDC^1=1.3$
1391 ETA=($0.0000475*XLB^3+[0.00173*XLB^2]*[0.0298*XLB]+0.5146$)
      IF ($VX^2>5.$) 1395, 1395, 1392
1392 IF ($VX^2>1.4$) 1393, 1394, 1394
1393 ETA$ETA=(1+ETA)*[($VX^2-5.)*1111111111111111111]
      GS TO 1395
1394 ETA=1.
1395 IF ($XLB=10.$) 1380, 1330, 1340
138C $YK2K1=0.0054*($XLB^3+0.104*XLB+C^4.37$)
      GS TO 1600
1330 $YK2K1=0.939$
      GS TO 1600
134C $YK2K1=0.939+[0.001525*XLB^10.00])$
1600 $ALP=AL$
      IF ($VX$) 1600, 1601, 1601
162C $CDS=CDC$
160C $CDS=CDS$.[2.8ALP*CBS(ALP/2)]$+3.14159*0.0000475*XLB^3+[0.00173*XLB^2]*[0.0298*XLB]+0.5146$)
      IF ($VX^2>5$) 1395, 1395, 1392
1392 IF ($VX^2>1.4$) 1393, 1394, 1394
1393 ETA=ETA+(1+ETA)*[$VX^2-5.]*1111111111111111111$
      GS TO 1395
1394 ETA=1.
1395 IF ($XLB=10.$) 1380, 1330, 1340
138C $YK2K1=0.0054*($XLB^3+0.104*XLB+C^4.37$)
      GS TO 1600
1330 $YK2K1=0.939$
      GS TO 1600
134C $YK2K1=0.939+[0.001525*XLB^10.00])$
1600 $ALP=AL$
      IF ($VX$) 1600, 1601, 1601
162C $CDS=CDC$
160C $CDS=CDS$.[2.8ALP*CBS(ALP/2)]$+3.14159*0.0000475*XLB^3+[0.00173*XLB^2]*[0.0298*XLB]+0.5146$)
      IF ($VX^2>5$) 1395, 1395, 1392
1392 IF ($VX^2>1.4$) 1393, 1394, 1394
1393 ETA=ETA+(1+ETA)*[$VX^2-5.]*1111111111111111111$
      GS TO 1395
1394 ETA=1.
1395 IF ($XLB=10.$) 1380, 1330, 1340
138C $YK2K1=0.0054*($XLB^3+0.104*XLB+C^4.37$)
      GS TO 1600
1330 $YK2K1=0.939$
      GS TO 1600
134C $YK2K1=0.939+[0.001525*XLB^10.00])$
1600 $ALP=AL$
      IF ($VX$) 1600, 1601, 1601
162C $CDS=CDC$
160C $CDS=CDS$.[2.8ALP*CBS(ALP/2)]$+3.14159*0.0000475*XLB^3+[0.00173*XLB^2]*[0.0298*XLB]+0.5146$)
      IF ($VX^2>5$) 1395, 1395, 1392
1392 IF ($VX^2>1.4$) 1393, 1394, 1394
1393 ETA=ETA+(1+ETA)*[$VX^2-5.]*1111111111111111111$
      GS TO 1395
1394 ETA=1.
1395 IF ($XLB=10.$) 1380, 1330, 1340
138C $YK2K1=0.0054*($XLB^3+0.104*XLB+C^4.37$)
      GS TO 1600
1330 $YK2K1=0.939$
      GS TO 1600
134C $YK2K1=0.939+[0.001525*XLB^10.00])$
1600 $ALP=AL$
      IF ($VX$) 1600, 1601, 1601
162C $CDS=CDC$
160C $CDS=CDS$.[2.8ALP*CBS(ALP/2)]$+3.14159*0.0000475*XLB^3+[0.00173*XLB^2]*[0.0298*XLB]+0.5146$)
      IF ($VX^2>5$) 1395, 1395, 1392
1392 IF ($VX^2>1.4$) 1393, 1394, 1394
1393 ETA=ETA+(1+ETA)*[$VX^2-5.]*1111111111111111111$
      GS TO 1395
1394 ETA=1.
1395 IF ($XLB=10.$) 1380, 1330, 1340
138C $YK2K1=0.0054*($XLB^3+0.104*XLB+C^4.37$)
      GS TO 1600
1330 $YK2K1=0.939$
      GS TO 1600
134C $YK2K1=0.939+[0.001525*XLB^10.00])$
1640 IF [TV*] = 164, 1642
1641 IF [TV*] = 1643, 1644
1647 FUNCTION 3 = 47912 + 42798 * SGMTIC = 324 + 324 * SGMTIC
GO TO 1650
1648 FUNCTION 1 = 5 = ATC
GO TO 1650
1649 FUNCTION 5 = 47202 * SGMTIC = 08631 * SQMTIC = SGMTIC
GO TO 1650
1650 IF [12T = 1] 1651 + 3333 * SQMTIC = 071429 * SGMTIC = SQMTIC
1652 IF [12T = 2] 1652, 1653, 1654
1653 DCDBSW = FUNCTION [TBCVC = 1 + 66667] = [CBS [XLAMW]] ** 2.5
1702 IF [1ST = 2] 1703, 1704, 1705
1703 DCDBSW = 0
SSTB = 0
CDDBWT = 0
GO TO 1705
1704 12T = 2
XXV = VXMC SQRT [CBS [XLAM24]]
1705 IF [TST = 1] 1706, 1707
1706 DCDBST = 0
STTB = 0
CDDBT = 0
GO TO 1708
1707 12T = 3
XXV = VXMC S Q R T [CBS [XLAM24]]
SGMTIC = SGRT [ABS ([XXV * XXV] = 1.1)] / TBCVCT = 0.33333
ATC * AR * [TBCVC = 0.33333]
GO TO 1640
1654 DCDST*FUNCTION[T0VCT*1.6667]*[COS[XLAMT4]]*2.5
DCDST=DCDST
1708 CVX = 1.31213*0.36333*VXM*0.6038*VXM*2*0.0601*VXM*3*0.00275*
1VXM**4
CDPTR=1.02*CFBBD*CBVC*SSUBS/AREA
CDPTR=CDDB*1.02*CFBBD*SSUBS/AREA
IF (VXM=1.0) 1709, 1709, 1711
1711 CDPPTR=[CDPPTR/0.2]*[1.2*VXM]
1709 FR=XLN90/E
CDWN1=0.000407*[FR**8] = 0.0102*[FR**7] + 0.018*[FR**6] = 0.016*[FR**5]
CDWN2=0.00172*[FR**8] = 0.00453*[FR**7] + 0.050*[FR**6] + 0.364*[FR**5]
1+1.96*[FR**4] = 2.406*[FR**3] + 3.160*[FR**2] + 2.391*FR + 0.000
CDWN3=0.00125*[FR**8] = 0.00370*[FR**7] + 0.0447*[FR**6] + 0.288*[FR**5]
IF (VXM=0.8) 1664, 1661, 1662
1661 CDPPTR=CDWN1
GO TO 1658
1662 IF (VXM=1.0) 1655, 1655, 1666
1665 CDPPTR=CDWN2
GO TO 1658
1655 CDPPTR=[(CDWN2=CDWN1)/0.2]*[VXM=0.8]+CDWN1
GO TO 1658
1666 IF (VXM=1.0) 1667, 1668, 1664
1668 CDPPTR=CDWN3
GO TO 1658
1667 CDPPTR=[(CDWN3=CDWN2)/0.2]*[VXM=1.0]+CDWN2
GO TO 1658
1664 CDPPTR=0.0
1658 CDDB=CDPPTR+CDPPTR+CDPPTR*VXM*3*14159*D=D/[4.*AREA]
CDALZ=CDDB
CDGT=1.1*[CDDBST*[STTB/AREA]*CDGT]
CDWX=1.1*[CDDBSW*[SWTB/AREA]*CDGW]
CDW2=1.1*[CDW2*DCDBS2*[SWTB/AREA]]
CDWBT=1.1*[CDW4+DCDSt*SWTB/AREA]+CDW2+DCDSt*[SWTB/AREA]
1+CDGT+CDDBST*[STTB/AREA]+CDDB
IF ([FFZ]=4) 1659, 1659, 1663
1655 T9NST=CDW9BT
\HONST=HONST+ZF*[T9NST=H9NST]
VXM=VXM*1
IZZY = IZZY + 1
GO TO 1
1632 CDW9BT=([H9NST=T9NST]/[SQRT[3]*SQRT[1*2]])*SQRT[VXM]+T9NST
1+(H9NST=T9NST)/[1+SQRT[3]*SQRT[1*2]]*SQRT[VXM]+T9NST
1+CDGT=1.1*[CDGT*[STTB/AREA]+CDGT]
CDGW=1.1*[CDGW*[SWTB/AREA]+CDGW]
CDW2=1.1*[CDW2*[SWTB/AREA]]
CDW2=CDW2+CDW2*[SWTB/AREA]
CDWBT=CDW2+CDW2*[SWTB/AREA]
CDALZ=CDDB
1663 CDSS=XXK2K1*SIN[2*AL]*SIN[AL/2]*3*14159*D/D/[4.*AREA]+ETA*CDC*AL
1+CDSS=CDSS*3/AREA
CAL=XXK2K1*SIN[2*AL]*COS[AL/2]*3*14159*D/D/[4.*AREA]+ETA*CDC*AL
1+CDSS=CDSS*2/AREA-CDALZ*COS[AL]*COS[AL]*SIN[AL]
CDSS=CDSS
IF [AL] 1603, 1604, 1603
1604 XCPB=0.0
90 T9 1605
1603 XCPB=[XCG/XREF]#[CMB/CNB]*XREF
1605 IF [SW] 1607,1607,1606
1606 IF [ST] 2973,2973,1608
1608 R=D/2.
1971 XB1=BT/2*
   XB2=BT/2*
   TT=0/6*
   HX=2*CR80TT*ABS(SIN[DE[TA]]+[XCPWB=XTAIL*CR80TT]ABS(SIN[AL])
   XLM=1*XLAMW
98 T9 1969
1970 XB1=BT/2*
   XB2=BT/2*
   TT=0/6*
   HX=2*[XCPWB+XWING*CR80TT]*ABS(SIN[AL])
   XLM=1*XLAMT
1965 FRT=[(XB2*R)/(2.0**[1+TT])]*[(3.04159/4.0)**[1+TT**2]]**TT+
   1**[1+TT**2]FRT=ARB[[(1+TT**2)/(1+TT**2)]]
   FWS=FRT+R
   F1=2.0**[FWS]**2+[HW1**2]
   H1=2.0**[FWS]**2+[HW1**2]
   ZC=FK
   ZD=HW1
   ZLT=0.0
   D0 1800 1=1
   ZL1=[(XB1-XLM1*R)]-(ZC*[1+XLAM1J]/[2.0*(XB1-R)])
   ZL2=ALB8C[(ZD**2)+(ZC+XB1)**2]/[(ZD**2)+(ZC+R)**2])
   ZL3=[(1+XLAM1J)/(XB1-R)]*[(XB1-R)+(ZD*ATAN[(ZC+XB1)/ZD]-ATAN[(ZC+
   1R)/ZD]J)
   ZL=[ZL1+ZL2+ZL3]
   JFT[1+2] 1810,1820,1820
1810 ZC=ZC
98 T9 1850
1820 IF[1+3] 1830,1840,1850
1830 ZL=ZL
   ZC=F11
   ZD=H11
   GB T9 1850
1840 ZL=ZL
   ZC=F11
1850 ZLT=ZLT+ZL
1860 CONTINUE
   IF [ICSC = 1] 297C,297C,2971
297C ART=[&T-F]**2/ST
   XBT=BT/2*
   CLI=[CLALW+CLALT+XXWB1+SW]*2.*ZLT*[XBT*R]/[2.0*3.04159*ART*
   FRT*AREA*[1+XLAMT]
   CLI=CL1*CBT(CAL)
   CLT=CLT+CLI
   CLI=CLT
   CLI=0.
   XCPVT=XCPB
98 T9 2972

72
2971 ARW*[BW*D]**2/SW
  XBW=BW/2.
   CI=[CLALW*CLALT*{XKWB1*SIN(AL)+XKTB1*SIN(DELTA)}*ST=2.*ZLT*(XBW=1R)]/2*3.14159*ARW*FRTT*AREA*[1.1*XLAMW]
   CL1=CL1*COS(AL)
   CLW=CLW+CLW
   C2=CLT+CLT
   XCPW=XCPW
   GO TO 2972

2973 CLT=0.
   CLALT=0.
   CLTD=0.
   CTD=0.
   CLBT=0.
   CLTB=0.
   CLTDB=0.
   CLVIST=0.
   CL1T=0.
   CL1W=0.
   CL1=0.
   GO TO 2972

2974 CLW=C.
   CLALW=0.
   CLW=0.
   CLWB=0.
   CLVSW=0.
   CL1H=0.
   CL1T=0.
   CL1C=C.

2972 IF [SW2] 700, 70C, 701
70C CLW=0.
   CLALW=0.
   CLW=0.
   CLWB=0.
   CLVSW=0.
   CL1=0.

701 ALPHA=AL
   IF[VXM=0.5] 1975, 1976, 1976

1975 XX=1.17
   GO TO 1973


1977 XX=2.0*SQRT[0.764*[VXM=0.126]**2]
   GO TO 1973


1979 XX=2.0*SQRT[1.298*[VXM=2.13]**2]
   GO TO 1973

1980 XX=0.87

1973 CDT=XX*ABS[SIN(AL)]*ST/AREA
   CDW=XX*ABS[SIN(AL)]*SW/AREA
   CDW2=XX*ABS[SIN(AL)]*SW2/AREA
   CDD=XX*ABS[SIN(AL+DELTA)]*ST/AREA
   CTD=CD=CDT
   CAT=CDT*COS(AL)=CLTD*SINT[AL]
   CNT=CLTD*COS[AL]+CDSD*SINT[AL]
   CAT=CL*COS[ALPHA]+CDT*SIN[ALPHA]
   CAT=COS[ALPHA]+CDT*SIN[ALPHA]
   CAW2=CDW2*COS[AL]=CLW2*SINT[AL]
\[ C = 49 \cdot 1 \cdot \text{SQRT}(T) \]
\[ P_S = P_S \cdot 70 \cdot 9 \]
\[ R_H = P_S / (1715 \cdot t) \]
\[ X_M = 2.270 \cdot [T \cdot 1.5] / [T \cdot 198.6] \cdot [10 \cdot 8] \]
\[ \text{REF} = [C \cdot R_H] / X_M \]
\[ \text{RETURN} \]
\[ \text{END} \]
SUBROUTINE CLASUB

COMMON XVXM, XDT, XL

COMMON CN, CA, CNB, CNXCNT, CNW2, CLW, CLT, CLW2, CLI, CLWB, CLVISW, CLIT,
1 CLIWCNDT, CAB, XCP2, XCG2, XCWP, XCP2, XCG2, XCPTV,
2 XLMAT4

COMMON LLK, ISKW, ISKW2, ISKPB, ISKPB2, IAFBW, IAFBW2, IL, LLLL, IJ, JJ,
1 INBODY, IZZY, ICSC, INBSE, NM, NMLK, IDT, IM, IAL, ISWP, IAFB

COMMON XLMAT, XLMAT2, XMACW, XMACWT, XMACW2, CLAMW, CLAMWT, CLAMW2,
1 XN, BT, B, CBREBT, CBRETT, CBREWT, CBREWT2, SW, ST, S, XWING, XTAIL, XWING2
2 XL, XL, XL, XL, XL, XL, XL, XL, XL, XL, XL, XN, AREA, XREF, SSUBS, XLBB, ZF, ART, ARW, ARW2, XLSNS

COMMON CLAM, BCBLAM, CBREBT, R1, CLAL1, CLAM1, BAR, RATI, XKTBI, XKTBI

1 XKWB, XKWB, XKWB, XKWB, XKWB, XKWB, XKWB, XKWB, XKWB, XKWB,
2 XCPB, XCPB, XCPB, XCPB, XCPB, XCPB, XCPB, XCPB, XCPB, XCPB,
3 XDCW2, CLALW, CLALW, CLALW, CLALW, CLALW, CLALW, CLALW, CLALW, CLALW,
4 XVM, XVM, XVM, XVM, XVM, XVM, XVM, XVM, XVM, XVM,
5 XLMAT, XLMAT, XLMAT, XLMAT, XLMAT, XLMAT, XLMAT, XLMAT, XLMAT, XLMAT,
6 CD, CD, CD, CD, CD, CD, CD, CD, CD, CD,
7 CLTB, CLTB, CLTB, CLTB, CLTB, CLTB, CLTB, CLTB, CLTB, CLTB,

COMMON CDBS, S, CD, S, CD, S, CD, S, CD, S, CD

COMMON CLAZ

IF (VXM - 1) 2, 3, 3
2 BETA = SQRT (1 - VXM**2)
  GO TO 4
3 BETA = SQRT (VXM**2 - 1) 2
4 IF (IZY = 4) 10, 20, 111
111 1 IF (VXM - 1) 4, 1, 1, 1
42 BETA = 0.0000001
41 KFIN = 0
  KFIN = KFIN + 1
  IF (SW) 504, 504, 411
411 ARW = [BH - D] * P/SW
  PAR = BETA * ARW
  ISWP = ISWP
  XLMAT = XLMAT
  ARW = ARW
505 IF (ISWP = 1) 5, 5, 200
5 IF (XLMAT = 251) 60, 10, 10
6C IF (VXM = 10) 20, 20, 30
2C CLAR = 1.8333 * BAR + 1.6
  GO TO 370
3C IF (BAR = 1) 50, 40, 40
4C CLAR = 1.508 * [1.26] * [2 - BAR]
  GO TO 370
5C CLAR = 3 * BAR + 1.6
  GO TO 370
6C IF (VXM = 10) 70, 70, 80
7C CLAR = 1.667 * BAR + 1.575
  GO TO 370
8C IF (BAR = 1) 90, 90, 100
9C CLAR = 1.667 * BAR + 1.575
  GO TO 370
10C IF (BAR = 2) 110, 110, 120
11C CLAR = 1.7417
  GO TO 370

78
12C CLAR=1.428*(1.22)**(2.*BAR)
  G0 TO 370
20C IF [XLM=.1] 210,260,260
21C IF [VXM=1.*] 220,220,230
22C CLAR=2.077*BAR+1.575
  G0 TO 370
23C IF [BAR=.25] 240,240,250
24C CLAR=2.077*BAR+1.575
  G0 TO 370
25C IF [BAR=4.*] 251,251,252
251 CLAR=1.668*BAR+1.667
  G0 TO 370
252 CLAR=1.587*(1.26)**(2.*BAR)
  G0 TO 370
26C IF [XLM=.3] 270,320,320
27C IF [VXM=1.*] 280,280,290
28C CLAR=2.065*BAR+1.6
  G0 TO 370
29C IF [BAR=0.75] 300,300,310
30C CLAR=2.065*BAR+1.6
  G0 TO 370
31C IF [BAR=2.5] 311,311,312
311 CLAR=2.17*BAR+2.293
  G0 TO 370
312 CLAR=1.543*(1.26)**(2.*BAR)
  G0 TO 370
32C IF [VXM=1.*] 330,330,340
33C CLAR=2.25*BAR+1.675
  G0 TO 370
34C IF [BAR=1.*] 350,350,360
35C CLAR=2.25*BAR+1.675
  G0 TO 370
36C CLAR=1.508*(1.26)**(2.*BAR)
37C IF [BAR=1.0] 800,800,810
80C ARR=1.0
  G0 TO 820
81C ARR=1./[ARR**([AR=1.0]/ARR)]
82C CLAL=CLAR*ARR**AR
  IF [KFIN=2] 500,501,502
50C CLALW=CLAL
  G0 TO 503
501 CLALW=CLAL
  G0 TO 503
502 CLALT=CLAL
503 IF [KFIN=2] 504,507,90C
504 KFIN=KFIN+1
  IF [SWP] 507,507,506
506 AR=[BAR=D]**2/8W
BAR=BETA*AR
8AR=8AR
SWP=SWP+1
ARW2=AR
XLM=XLMW2
  G0 TO 505
  set for wing 2 go back to start over

507  KFIN=KFIN+1
      IF (ST) 509, 509, 508
508  AR=(ET*D)**2/ST
      BAR=BETA*AR
      ART=AR
      ISWP=ISWPT
      XLM=XLMNT
      GB=T8 505
509  IF (SW) 5100, 5100, 900
      5100 IF (SW2) 5110, 5110, 900
      5110 IF (ST) 980, 980, 900
980  LLLL=2
      RETURN
981  LLKK=0
982  IZZY = IZZY + 1
      IF (SW) 510, 510, 901
983  IF (SW2) 942, 942, 511
984  LLLL=1
      RETURN
510  IZZY = IZZY + 1
      IF (SW2) 942, 942, 511
942  LLLL=1
      RETURN
511  CBLAM=COS(CLAMW)/[SIN(CLAMW)]
      ARW=(BW-D)**2/SW
      BC=CLAM*BETA*CBLAM
      CRBT=CRBTW
      B1=BW
      IAFB=IAFBW
      MAC=MACW
      T8VC=T8VCW
      CLAL=CLALW
      XLAM=XLAMN
      ISWP=ISWPT
      BAR=BETA*ARW
      RATR=CRBT/[BETA*D]
      LLLL=0
      RETURN
512  IZZY = IZZY + 1
      IF (SW2) 942, 942, 511
942  LLLL=1
      RETURN
513  CBLAM=COS(CLAMW2)/[SIN(CLAMW2)]
      B1=BW
      IAFB=IAFBW2
      CLAL=CLALW2
      XLAM=XLAMW2
      MAC=MACW2
      T8VC=T8VCW2
      ISWP=ISWPT2
      ARW=(BW2-D)**2/SW2
      BAR=BETA*ARW2
      RATR=CRBT/[BETA*D]
      LLLL=0
      RETURN
END
SUBROUTINE CATSUB
DIMENSION XVMX(16), XCT(16), XAL(48)
COMMON XVMX, XCT, XAL
COMMON CN, CA, CKB, CNW, CNT, CN2, CLW, CLT, CLW2, CLI, CLWB, CLVISW, CLIT,
1 CL14, CNTD, CKB, CP2, VC2, XCPB, XCPD, XCPW, XCPW2, XCG, XCPTV,
2 XLM4
COMMON LLKX, ISXW, ISXPT, ISXPW, IAFTB, IAFTB2, IAFBW, IAFBW2, IL1, LL1, LL2, ILJ,
1 IJN, B0DY, IZZY, ISGC, INSE, NM, MA, MLK, IDT, IM, IAL, ISWP1, IAFB
COMMON XLMW, XLAM, XLAMW, XMACW, XMCT, XMACW2, CLAMW, CLAMT, CLAMW2,
1 BX, BT, TN, CROBT, CROBT2, CRW, ST, SW, STW, SWING, XTAIL, XWING2
2 XL, DL, D1, XLN, AREA, XPEF, SSUBS, XLAB, ZF, ART, ARW, ARW2, XLNA
COMMON CBMM9, CBMM, CBMM2, CBMM3, CBMM4, CBMM5, CBMM6, CBMM7,
1 CBMM8, CBMM9, CBMM10, CBMM11, CBMM12, CBMM13, CBMM14, CBMM15,
2 CBMM16, CBMM17, CBMM18, CBMM19, CBMM20, CBMM21, CBMM22, CBMM23,
3 CBMM24, CBMM25, CBMM26, CBMM27, CBMM28, CBMM29, CBMM30, CBMM31,
4 CBMM32, CBMM33, CBMM34, CBMM35, CBMM36, CBMM37, CBMM38, CBMM39,
5 CBMM40, CBMM41, CBMM42, CBMM43, CBMM44, CBMM45, CBMM46, CBMM47,
6 CBMM48, CBMM49, CBMM50, CBMM51, CBMM52, CBMM53, CBMM54, CBMM55,
7 CBMM56, CBMM57, CBMM58, CBMM59, CBMM60, CBMM61, CBMM62, CBMM63,
8 CBMM64, CBMM65, CBMM66, CBMM67, CBMM68, CBMM69, CBMM70, CBMM71,
9 CBMM72, CBMM73, CBMM74, CBMM75, CBMM76, CBMM77, CBMM78, CBMM79,
10 CBMM80, CBMM81, CBMM82, CBMM83, CBMM84, CBMM85, CBMM86, CBMM87,
11 CBMM88, CBMM89, CBMM90, CBMM91, CBMM92, CBMM93, CBMM94, CBMM95,
12 CBMM96, CBMM97, CBMM98, CBMM99, CBMM100, CBMM101, CBMM102, CBMM103,
13 CBMM104, CBMM105, CBMM106, CBMM107, CBMM108, CBMM109, CBMM110,
14 CBMM111, CBMM112, CBMM113, CBMM114, CBMM115, CBMM116, CBMM117,
15 CBMM118, CBMM119, CBMM120, CBMM121, CBMM122, CBMM123, CBMM124,
16 CBMM125, CBMM126, CBMM127, CBMM128, CBMM129, CBMM130, CBMM131,
17 CBMM132, CBMM133, CBMM134, CBMM135, CBMM136, CBMM137, CBMM138,
18 CBMM139, CBMM140, CBMM141, CBMM142, CBMM143, CBMM144, CBMM145,
19 CBMM146, CBMM147, CBMM148, CBMM149, CBMM150, CBMM151, CBMM152,
20 CBMM153, CBMM154, CBMM155, CBMM156, CBMM157, CBMM158, CBMM159,
21 CBMM160, CBMM161, CBMM162, CBMM163, CBMM164, CBMM165, CBMM166,
22 CBMM167, CBMM168, CBMM169, CBMM170, CBMM171, CBMM172, CBMM173,
23 CBMM174, CBMM175, CBMM176, CBMM177, CBMM178, CBMM179, CBMM180,
24 CBMM181, CBMM182, CBMM183, CBMM184, CBMM185, CBMM186, CBMM187,
25 CBMM188, CBMM189, CBMM190, CBMM191, CBMM192, CBMM193, CBMM194,
26 CBMM195, CBMM196, CBMM197, CBMM198, CBMM199, CBMM200, CBMM201,
27 CBMM202, CBMM203, CBMM204, CBMM205, CBMM206, CBMM207, CBMM208,
28 CBMM209, CBMM210, CBMM211, CBMM212, CBMM213, CBMM214, CBMM215,
29 CBMM216, CBMM217, CBMM218, CBMM219, CBMM220, CBMM221, CBMM222,
30 CBMM223, CBMM224, CBMM225, CBMM226, CBMM227, CBMM228, CBMM229,
31 CBMM230, CBMM231, CBMM232, CBMM233, CBMM234, CBMM235, CBMM236,
IF (BAR=4.0) 2120, 2130, 2140
XBCRBW=0.25*[1.*XLAM1]+[32.125-SQR{T[1032.02*(BAR-4.0)**2]}+1*[1.*XLAM1]*AL8G[1.12+0.3*D/B1]+7.5*SQR{T[72.25-(BAR-4.0)**2]}]*G0 T0 2400
2130 XBCRBW=0.25*[1.*XLAM1]*AL8G[1.12+0.3*D/B1]*G0 T0 2400
2140 IF [(ISwP=1 ) 2140, 2140, 2150
2140 IF (BAR=3.0) 2160, 2170, 2170
2160 XBCRBW=[9.235*25**[1.*XLAM1]]+SQR{T[9.71+25**[1.*XLAM1]]}*G0 T0 2200
2170 XBCRBW=0.005*BAR+0.46*G0 T0 2200
2180 IF (BAR=3.0) 2180, 2190, 2190
2180 XBCRBW=0.675*XLAM1+[0.675+9.235-SQR{T[94.1-(BAR-3.0)**2]}]*G0 T0 2300
2190 XBCRBW=C.005*BAR+0.46+0.2*[1.*XLAM1]*G0 T0 2300
2200 RARLAM=BAR*[1.*XLAM1]*[1.+(1./BCALAM)]
IF (RARLAM=4.0) 2210, 2210, 2220
2210 IF (BAR=2.0) 2230, 2240, 2240
2230 XBMID1 = AL8G[1.32 -(0.32*XLAM1)]*XBMID2=4.0+[XBMID1**2]+[0.5+0.5139*[D/B1]+[1.17+XLAM1]*[1./(0.331+1.0*(D/B1))]]*XBMID
2240 XBCRBW=C.5+0.2569*[C/D/B1]*[0.5+0.5139*[D/B1]+[1.17+XLAM1]*[1./(0.331+1.0*(D/B1))]]*BAR*[1.*XLAM1]*G0 T0 2400
2250 XBCRBW=C.067*G0 T0 2400
2270 XBCRBW=2.32+SQR{T[8.94C1-([1./RAT18]+1.)**2]}*G0 T0 2400
2310 XBCRBW=C.429/RAT18+0.5*G0 T0 2400
2330 RARLAM=BAR*[1.*XLAM1]*[1.+(1./BCALAM)]
IF (RARLAM=4.0) 2310, 2310, 2220
2310 IF (BAR=2.0) 2320, 2240, 2240
2320 XBMID1 = AL8G[1.65+0.69*XLAM1]*G0 T0 2235
2400 C=D1
ARAT=BAR/BETA
IF (ARAT=3.0) 20, 20, 21
20 ARAT=3.0
20 MDC=2.0*[1.*XLAM1]**1.6*EXP(-4.*ARAT)*IF (ARAT=1.0) 22, 23, 23
23 MDC=MDC+0.25*[ARAT=1.0]
22 CONTINUE
RETURN END
TABLE 2

INPUT NOMENCLATURE*

<table>
<thead>
<tr>
<th>Variable</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>AREA</td>
<td>arbitrary reference area, ft²</td>
</tr>
<tr>
<td>BT</td>
<td>total tip-to-tip tail span including the missile body**</td>
</tr>
<tr>
<td>BW</td>
<td>total tip-to-tip wing span including the missile body</td>
</tr>
<tr>
<td>BN2</td>
<td>total tip-to-tip wing two span including the missile body</td>
</tr>
<tr>
<td>CLAMT</td>
<td>leading edge sweep angle of the tail***</td>
</tr>
<tr>
<td>CLAMW</td>
<td>leading edge sweep angle of the wing</td>
</tr>
<tr>
<td>CLAMW2</td>
<td>leading edge sweep angle of wing two</td>
</tr>
<tr>
<td>CROOTT</td>
<td>tail root chord</td>
</tr>
<tr>
<td>CROOTW</td>
<td>wing root chord</td>
</tr>
<tr>
<td>CROOTW2</td>
<td>wing two root chord</td>
</tr>
<tr>
<td>D</td>
<td>body diameter</td>
</tr>
<tr>
<td>HT</td>
<td>altitude in feet</td>
</tr>
<tr>
<td>IAFBT</td>
<td>afterbody constant for the tail</td>
</tr>
<tr>
<td></td>
<td>0 - no afterbody following the tail</td>
</tr>
<tr>
<td></td>
<td>1 - afterbody following the tail</td>
</tr>
<tr>
<td>IAFBW</td>
<td>afterbody constant for the wing</td>
</tr>
<tr>
<td></td>
<td>0 - no afterbody following the wing</td>
</tr>
<tr>
<td></td>
<td>1 - afterbody following the wing</td>
</tr>
<tr>
<td>IAFIW2</td>
<td>afterbody constant for wing two</td>
</tr>
<tr>
<td></td>
<td>0 - no afterbody following wing two</td>
</tr>
<tr>
<td></td>
<td>1 - afterbody following wing two</td>
</tr>
<tr>
<td>IAL</td>
<td>number of angles of attack</td>
</tr>
<tr>
<td>ICSC</td>
<td>control surface constant</td>
</tr>
<tr>
<td></td>
<td>1 - tail control</td>
</tr>
<tr>
<td></td>
<td>2 - wing control</td>
</tr>
<tr>
<td></td>
<td>3 - canard control</td>
</tr>
<tr>
<td>IDT</td>
<td>number of control surface deflection angles (must be at least one; if there is no control surface, IDT = 1, DELTA = 0.0)</td>
</tr>
<tr>
<td>IM</td>
<td>number of Mach numbers</td>
</tr>
</tbody>
</table>
TABLE 2
(continued)

<table>
<thead>
<tr>
<th>Variable</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>INOSE</td>
<td>nose constant&lt;br&gt;1 - blunted ogive or cone&lt;br&gt;2 - pointed ogive&lt;br&gt;3 - pointed cone</td>
</tr>
<tr>
<td>ISWPT</td>
<td>sweep constant of tail&lt;br&gt;1 - unswept mid-chord&lt;br&gt;2 - unswept trailing edge</td>
</tr>
<tr>
<td>ISWPW</td>
<td>sweep constant of wing&lt;br&gt;1 - unswept mid-chord&lt;br&gt;2 - unswept trailing edge</td>
</tr>
<tr>
<td>ISWPW2</td>
<td>sweep constant of wing two&lt;br&gt;1 - unswept mid-chord&lt;br&gt;2 - unswept trailing edge</td>
</tr>
<tr>
<td>WBODY</td>
<td>number of configurations being run (a configuration is one complete data deck)</td>
</tr>
<tr>
<td>ST</td>
<td>exposed planform area of one pair of tail panels</td>
</tr>
<tr>
<td>SW</td>
<td>exposed planform area of one pair of wing panels</td>
</tr>
<tr>
<td>SW2</td>
<td>exposed planform area of one pair of wing two panels</td>
</tr>
<tr>
<td>TOVCT</td>
<td>thickness-to-chord ratio of the tail</td>
</tr>
<tr>
<td>TOVCW</td>
<td>thickness-to-chord ratio of the wing</td>
</tr>
<tr>
<td>TOVCW2</td>
<td>thickness-to-chord ratio of wing two</td>
</tr>
<tr>
<td>XAL</td>
<td>missile angle of attack (degrees)</td>
</tr>
<tr>
<td>XCG</td>
<td>missile center of gravity location as measured from the nose</td>
</tr>
<tr>
<td>XDT</td>
<td>control surface deflection angle (degrees)</td>
</tr>
<tr>
<td>XL</td>
<td>missile length</td>
</tr>
<tr>
<td>XLAMT</td>
<td>tip-to-root-chord ratio of the tail</td>
</tr>
<tr>
<td>XLAMW</td>
<td>tip-to-root-chord ratio of the wing</td>
</tr>
<tr>
<td>XLAMW2</td>
<td>tip-to-root-chord ratio of wing two</td>
</tr>
<tr>
<td>XLNOSE</td>
<td>length of the nose</td>
</tr>
</tbody>
</table>
TABLE 2  
(continued)

<table>
<thead>
<tr>
<th>Abbreviation</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>XMECT</td>
<td>mean geometric chord of the tail</td>
</tr>
<tr>
<td>XMACW</td>
<td>mean geometric chord of the wing</td>
</tr>
<tr>
<td>XMACW2</td>
<td>mean geometric chord of wing two</td>
</tr>
<tr>
<td>XREF</td>
<td>arbitrary reference length</td>
</tr>
<tr>
<td>XTAIL</td>
<td>distance from the nose to the leading edge of the tail root chord</td>
</tr>
<tr>
<td>XVXM</td>
<td>missile flight Mach number</td>
</tr>
<tr>
<td>XWING</td>
<td>distance from the nose to the leading edge of the wing root chord</td>
</tr>
<tr>
<td>XWING2</td>
<td>distance from the nose to the leading edge of wing two root chord</td>
</tr>
</tbody>
</table>

* The control surface is defined as the tail regardless of the mode of control, and the fixed surface(s) is (are) always defined as the wing(s). See Figure 1.

** All linear dimensions are in feet.

*** All angular dimensions are in degrees.
## TABLE 3
### PROGRAM INPUT FORMAT

<table>
<thead>
<tr>
<th></th>
<th>Run Identification</th>
<th>Configuration Title and/or Number (1st 60 columns)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td></td>
<td></td>
</tr>
<tr>
<td>2</td>
<td>Control Constants</td>
<td>ICSC (15) INOSE (15) IDT (15) IM (15) IAL (15) NBODY (15)</td>
</tr>
<tr>
<td>3</td>
<td>Wing Inputs</td>
<td>ISWPW (15) IAFBW (15) XLAMW (F10.5) CLAMW (F10.5) BW (F10.5) CROOTW (F10.5) SW (F10.5) XMACW (F10.5) XWING (F10.5)</td>
</tr>
<tr>
<td>4</td>
<td>Wing 2 Inputs</td>
<td>ISWPW2 (15) IAFBW2 (15) XLAMW2 (F10.5) CLAMW2 (F10.5) BW2 (F10.5) CROOTW2 (F10.5) SW2 (F10.5) XMACW2 (F10.5) XWING2 (F10.5)</td>
</tr>
<tr>
<td>5</td>
<td>Tail Inputs</td>
<td>ISWPWT (15) IAFBT (15) XLMT (F10.5) CLMT (F10.5) BT (F10.5) CROOTT (F10.5) ST (F10.5) XMACT (F10.5) XTAIL (F10.5)</td>
</tr>
<tr>
<td>6</td>
<td>Miscellaneous Data</td>
<td>HT (F10.3) D (F10.3) XL (F10.3) XLNOSE (F10.3) XCG (F10.3) AREA (F10.3) XREF (F10.3)</td>
</tr>
<tr>
<td>7</td>
<td>Miscellaneous Data</td>
<td>TOVCW (F10.3) TOVCW2 (F10.3) TOVCW3 (F10.3)</td>
</tr>
<tr>
<td>8</td>
<td>Control Surface Deflection Angles</td>
<td>XDT (F5.1) Any number of deflection angles up to 16 may be input.</td>
</tr>
<tr>
<td>9</td>
<td>Missile Flight Mach Numbers</td>
<td>XVXM (F5.1) Any number of Mach numbers up to 16 may be input.</td>
</tr>
<tr>
<td>10</td>
<td>Missile Angles of Attack</td>
<td>XAL (F5.1) Any number of angles of attack up to 48 may be input.</td>
</tr>
<tr>
<td>11</td>
<td>Missile Angles of Attack</td>
<td>If this card is not required, leave out of data deck.</td>
</tr>
<tr>
<td>12</td>
<td>Missile Angles of Attack</td>
<td>If this card is not required, leave out of data deck.</td>
</tr>
<tr>
<td>Symbol</td>
<td>Description</td>
<td></td>
</tr>
<tr>
<td>--------</td>
<td>------------------------------------------------------------------------------</td>
<td></td>
</tr>
<tr>
<td>AL</td>
<td>missile angle of attack, degrees</td>
<td></td>
</tr>
<tr>
<td>CA</td>
<td>total axial force coefficient</td>
<td></td>
</tr>
<tr>
<td>CDTOT</td>
<td>total missile drag coefficient</td>
<td></td>
</tr>
<tr>
<td>CLALT</td>
<td>$C_L$ of the tail</td>
<td></td>
</tr>
<tr>
<td>CLALW</td>
<td>$C_L$ of the wing</td>
<td></td>
</tr>
<tr>
<td>CLALW2</td>
<td>$C_L$ of wing two</td>
<td></td>
</tr>
<tr>
<td>CLB</td>
<td>body lift coefficient</td>
<td></td>
</tr>
<tr>
<td>CLI</td>
<td>lift loss due to downwash</td>
<td></td>
</tr>
<tr>
<td>CLTOT</td>
<td>total missile lift coefficient</td>
<td></td>
</tr>
<tr>
<td>CLTT</td>
<td>tail lift coefficient</td>
<td></td>
</tr>
<tr>
<td>CLWT</td>
<td>wing lift coefficient</td>
<td></td>
</tr>
<tr>
<td>CM</td>
<td>total pitching moment coefficient about the missile center of gravity</td>
<td></td>
</tr>
<tr>
<td>CN</td>
<td>total normal force coefficient</td>
<td></td>
</tr>
<tr>
<td>CNB</td>
<td>body normal force coefficient</td>
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<tr>
<td>CNT</td>
<td>tail normal force coefficient</td>
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<tr>
<td>CNTD</td>
<td>tail normal force coefficient due to control surface deflection</td>
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<tr>
<td>CNW</td>
<td>wing normal force coefficient</td>
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<tr>
<td>DELTA</td>
<td>control surface deflection angle, degrees</td>
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<tr>
<td>VXM</td>
<td>missile flight Mach number</td>
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<tr>
<td>XCPB</td>
<td>body center of pressure location as measured from the nose, feet</td>
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</tr>
<tr>
<td>XCPT</td>
<td>tail center of pressure location as measured from the nose, feet</td>
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\[
CMB = CNB(XCG - XCPB)
\]
<table>
<thead>
<tr>
<th>XCPW</th>
<th>wing center of pressure location as measured from the nose, feet</th>
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</thead>
<tbody>
<tr>
<td>XCP2</td>
<td>total missile center of pressure location as measured from the nose, feet</td>
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</tbody>
</table>

\[
X_{CP} = X_{CE} - X_{CP2}.
\]
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12  DDC

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Hampton, Virginia  23365
A method for predicting the static, longitudinal aerodynamic characteristics of typical missile configurations at zero roll angle (i.e., in a plus configuration) has been developed and programmed for use on the IBM 7090 digital computer. It can be applied throughout the subsonic, transonic, and supersonic speed regimes to slender bodies of revolution or to nose-cylinder body combinations with low aspect-ratio lifting surfaces. The aerodynamic characteristics can be computed for missile configurations operating at angles of attack up to 180 degrees. The effect of control surface deflections for all modes of aerodynamic control are taken into account by this method. The method is based on well-known linear, nonlinear crossflow and slender body theories with empirical modifications to provide the high angle of attack capability. Comparisons of the theory with experimental data are presented to demonstrate the accuracy of the method.
<table>
<thead>
<tr>
<th>KEY WORDS</th>
<th>LINK A</th>
<th>LINK B</th>
<th>LINK C</th>
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<td>Air-to-Air Missiles</td>
<td>Missiles</td>
<td>Aerodynamics</td>
<td>Aerodynamics</td>
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<td>Aerodynamic Prediction Computer Program</td>
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Pitching Moment
Stability and Control
Static Aerodynamics

4. Aerodynamic Calculations