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A 3-D vortex lattice study of delta wings with and without dihedral and twist

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**A 3-D VORTEX LATTICE STUDY OF DELTA
WINGS WITH AND WITHOUT DIHEDRAL
AND TWIST**

A Thesis

Presented to

The Faculty of the Department of Mechanical and Aerospace Engineering

San Jose State University

In Partial Fulfillment

of the Requirements for the Degree

Master of Science

By

Akbar Sultan

December, 1997

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APPROVED FOR THE DEPARTMENT OF MECHANICAL
AND AEROSPACE ENGINEERING

Dr. Nikos Mourtos

Dr. Henry Pernicka

Dr. Richard Desautel

APPROVED FOR THE UNIVERSITY

ABSTRACT

A 3-D VORTEX LATTICE STUDY OF DELTA WINGS WITH AND WITHOUT DIHEDRAL AND TWIST

by Akbar Sultan

Current demand on fighter aircraft for higher range, speed, and maneuverability has led to highly swept, trapezoidal, and delta wings of various shapes. Furthermore, to enhance the supersonic performance of these aircraft, wing droops and twist are often added. Although highly swept and delta wings have exceptional performance at supersonic speeds, they do not perform as well at subsonic speeds. This thesis is a computational study of the subsonic aerodynamic characteristics of four single sweep delta wings with and without dihedral, and with and without twist using a Three Dimensional Vortex Lattice Method (VLM3D) computer code. It is found that the wing with no dihedral and twist has the best subsonic aerodynamic performance, whereas the wings with dihedral have the best maneuverability performance.

DEDICATION

This thesis is dedicated to my father Dr. Mohammad Anwar Sultan, the best father one could possibly wish for, and also called by his colleagues to be one of the best professors and nuclear physicists of our time. Without his unbridled dedication to the success and well being of his family, I would never have reached the position to write this thesis. His wisdom and presence will truly be missed by me, the rest of our family, and friends throughout the world.

Dr. Mohammad Anwar Sultan

May 5, 1940, - February 2, 1997

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INTRODUCTION

Recent demand on military aircraft for higher range, speed, and performance at supersonic speeds has led to the development of highly swept, trapezoidal, and delta wings like those shown in figure 1. Furthermore, in some cases, wing droops and twist are also added to these wings in an effort to enhance the supersonic aerodynamic and maneuvering performance of the aircraft. For example, figure 1(d) shows the XB-70 Valkyrie, with wing droops deployed at supersonic speeds for capturing the high pressure field generated by the shock waves at the leading edge of the wing. Figure 1(b) shows another example of such wings, the F-4 Phantom, where wing dihedral has been added to improve maneuvering performance. Although highly swept and delta wings have exceptional performance at supersonic speeds, they lack good aerodynamic performance at subsonic speeds. In this thesis, the three dimensional Vortex Lattice Method (ref. 1) is used to predict the aerodynamic characteristics of delta wings at subsonic speeds.

Why use the Vortex Lattice Method (VLM)? Classical theories, such as Prandtl's lifting-line theory, which give reasonable results for straight wings of moderate to high aspect ratio, fail to produce reliable results in predicting the aerodynamic characteristics for wings with low aspect ratio and large sweep angles. The Vortex Lattice Method on the other hand, has been shown to yield reliable results for cases where the classical methods fail. In the VLM approach, the wing surface is broken down into a number of lattices, where horseshoe vortices are placed. Then, a control point on each lattice is taken, where the combined effects of all the horseshoe vortices of all the lattices are calculated by imposing the tangency condition. From this, the circulation around the wing is determined, and then all of the other aerodynamic

(a) XB-70 Valkyrie during takeoff

(b) F-4 Phantom with wing dihedral

(c) Soviet MiG 31

(d) XB-70 with droops deployed

(e) Lockheed-Martin YF22

(f) McDonnell Douglas YF-23

Figure 1. Various aircraft using delta, highly swept, and trapezoidal wings, including wings with dihedral

characteristics are easily calculated.

Thus, using the VLMD3D code, a detailed analysis of four delta wings with the same overall dimensions was performed, and the results were compared. The first delta wing studied, shown in figure 2(a), is a fixed sweep wing with dihedral, twist, and camber. The second delta wing studied, shown in figure 2(b) has the same twist as in the first case, except that it has no dihedral. The third wing, shown in figure 2(a), has the same dihedral as in case 1, but it has no twist. The fourth wing, shown in figure 2(b), has neither twist or dihedral. A detailed description of each wing is presented in the results and appendices sections of this thesis. Some of the results compared are the lift curve, induced drag coefficient (C_{Di}) vs. lift coefficient (C_L), and the induced drag parameter (C_{Di}/C_L^2) vs. the lift coefficient (C_L). The variation of the coefficients of induced section drag (c_{di}), section leading-edge thrust (c_T), section leading-edge suction (c_s), and combination of basic and additional spanwise load distribution at coefficients of lift $C_L=1.0$ are compared. Furthermore, spanwise location of center of pressure, lift coefficient due to twist, pitching moment coefficient at zero lift (C_{M0}), total leading-edge thrust and suction coefficients at 1 radian angle of attack, lift curve slope(a_0), induced drag parameter based on near-field solution, and angle of attack for zero lift (α_{L0}) are also compared. Finally, damping-in-pitch parameter, damping-in-roll parameter, lift coefficient due to pitch rate, and the longitudinal stability parameter (dC_m/dC_L) are compared. It is concluded that from the aerodynamic standpoint, the wing with no twist and dihedral has the best characteristics, and from maneuverability standpoint the wing with both twist and dihedral has the best performance.

(a)

(b)

Figure 2. Plot of Wing Planforms Studied.

(a) Fixed sweep wing with dihedral and twist and camber, for case 1; fixed sweep wing with dihedral and no twist for case 3. The spanwise plane view shows the dihedral at the bottom of the plot. (b) Fixed sweep wing with twist and camber and no dihedral for case 2; fixed sweep wing without dihedral and twist for case 4.

SYMBOLS

A	aspect ratio; listed as AR in computer code output
B_k	element of boundary-condition matrix, $4\pi\alpha_k$
b	wing span, m (ft)
C_{Di}	induced drag coefficient, $\frac{\text{Induced drag}}{q_\infty S_{ref}}$
C_{Di}/C_L^2	induced drag parameter based on Munk's far-field solution
C_{Di}/C_L^2	induced drag parameter based on near-field solution
C_L	lift coefficient, $L/q_\infty S_{ref}$
$C_{L\tau}$	lift coefficient based on additional loading and actual planform area
C_{Lq}	lift coefficient due to pitch rate, $\frac{\partial C_L}{\partial \left(\frac{qc_{ref}}{2U} \right)}$, per radian
$C_{L\alpha}$	lift-curve slope, $\left(\frac{\partial C_L}{\partial \alpha} \right)_0$, per degree or per radian
C_l	rolling-moment coefficient, Rolling moment/($q_\infty S_{ref} b$)
C_{lp}	damping-in-roll parameter, $\frac{\partial C_l}{\partial \left(\frac{pb}{2U} \right)}$, per radian
C_m	pitching-moment coefficient about $\hat{Y}$ -axis, Pitching moment/($q_\infty S_{ref} c_{ref}$)
$\partial C_m / \partial C_L$	longitudinal stability parameter

C_{mq}	damping-in-pitch parameter, $\frac{\partial C_m}{\partial \left(\frac{qc_{ref}}{2U} \right)}$, per radian
C_n	element of circulation term matrix, Γ_n/U
ΔC_p	incremental pressure coefficient, $\frac{p_{lower} - p_{upper}}{q_\infty} = \frac{\Delta p}{q_\infty}$
C_s	leading-edge suction coefficient, $\frac{Suction}{q_\infty S_{ref}}$
C_T	leading-edge thrust coefficient, $\frac{Leading - edge \ thrust}{q_\infty S_{ref}}$
c	chord, m (ft)
c_{av}	average chord, S_r/b , m (ft)
c_c	chord along left trailing leg of elemental panel, m (ft)
$C_{d,ii}$	section induced drag coefficient based on near-field solution
C_l	section lift coefficient
c_{ref}	reference chord, m (ft)
c_s	section leading-edge suction coefficient
c_t	section leading-edge thrust coefficient
d_{ii}	section induced drag based on near-field solution, N/m (lb/ft)
F	influence function which geometrically relates influence of single horseshoe vortex to a quantity which is proportional to velocity induced at a point, m-l (ft-l)

$\bar{F}$	sum of influence function F at a control point on the wing caused by two symmetrically located horseshoe vortices, one on left half of the wing and one on the right half of the wing, m-l, (ft-l)
G_{nk}	element of influence function matrix, $\bar{F}_{w,n,k} - \bar{F}_{v,n,k} \tan \phi_n$
L	lift for entire wing, N (lb)
l	lift per unit length of span, N/m (lb/ft)
$\tilde{l}$	lift per unit length of vortex filament, N/m (lb/ft)
M_Y	pitching moment for entire wing about $\hat{Y}$ -axis, m-N (ft-lb)
M_∞	free-stream Mach number
m_Y	pitching moment about $\hat{Y}$ -axis due to lift developed on elemental panel, m-N (ft-lb)
N	maximum number of elemental panels on entire wing
$\bar{N}_c$	number of elemental panels in a chordwise row
$\bar{N}_s$	number of chordwise rows of elemental panels on wing semispan
p	roll rate, rad/sec; also, pressure, N/m ² (lb/ft ²)
q	pitch rate about $\hat{Y}$ -axis, rad/sec
q_∞	free-stream dynamic pressure, N/m ² (lb/ft ²)
S_{ref}	reference area m ² (ft ²)
S_r	actual planform area, m ² (ft ²)
s	horseshoe semiwidth in plane of horseshoe vortex, m (ft)

T	$= \frac{S_{ref}}{\left(2s_n \cos \phi \ c_{av} \right)}$
t	section leading-edge thrust per unit span, N/m (lb/ft)
U	free-stream velocity, m/sec (ft/sec)
u	backwash velocity, m/sec (ft/sec)
V	resultant velocity, m/sec (ft/sec)
v	sidewash velocity, m/sec (ft/sec)
w	downwash velocity, m/sec (ft/sec)
X, Y, Z	axis system of a given horseshoe vortex (see figure 4)
$\bar{X}, \bar{Y}, \bar{Z}$	body-axis system for planform (see figure 4)
$\hat{X}, \hat{Y}, \hat{Z}$	wind-axis system
x, y, z	distance along X-, Y-, and Z-axis, respectively, m (ft)
$\bar{x}, \bar{y}, \bar{z}$	distance along $\bar{X}$ -, $\bar{Y}$ -, and $\bar{Z}$ -axis, respectively, m (ft)
$\hat{x}, \hat{y}, \hat{z}$	distance along $\hat{X}$ -, $\hat{Y}$ -, and $\hat{Z}$ -axis, respectively, m (ft)
$\bar{x}_{c/4}$	midspan $\bar{x}$ -location of quarter-chord of elemental panel, m (ft)
$\bar{x}_{3c/4}$	midspan $\bar{x}$ -location of three-quarter-chord of elemental panel, m (ft)
x'	$= x/\beta$
y_{cp}	fractional spanwise distance from root chord to center of pressure on left wing panel
α	angle of attack, degrees
α_i	induced angle of attack, radians

β	Prandtl-Glauert correction factor to account for effect of compressibility in subsonic flow, $\sqrt{1 - M_\infty^2}$
Γ	vortex strength, m ² /sec (ft ² /sec)
γ	nondimensional lift, $\frac{\Gamma}{bU}$ or $\frac{c_l c}{2b}$
$\Delta\Gamma$	net vortex strength along left trailing leg of elemental panel, m ² /sec (ft ² /sec)
η	nondimensional spanwise coordinate, $\hat{y}/(b/2)$
ρ	density, kg/m ³ (slugs/ft ³)
ϕ	dihedral angle, in $\bar{Y} - \bar{Z}$ plane, deg
Λ	planform leading-edge sweep angle, in $\bar{X} - \bar{Y}$ plane, deg
ψ	quarter-chord sweep angle of elemental panel; because of the small angle assumption, also used as sweep angle of spanwise horseshoe vortex filament, in X-Y plane, deg
ψ'	$= \tan^{-1} \left(\frac{\tan \Psi}{\beta} \right)$

Subscripts:

a	additional; or angle of attack
B	twist and/or camber at $C_L=0$ for chordwise row of elemental panels
b	twist and/or camber at $C_L=0$ for elemental panel
d	desired
i	index for elemental panel in chordwise row

j	maximum number of elemental panels in chordwise row
k	index for control point
l	left half of wing
lower	lower surface
n	index for elemental panel on wing semispan
o	value taken at $C_L=0$
r	right half of wing
rad	per radian angle of attack
s	spanwise bound vortex element
t	chordwise bound vortex element
tc	twist and/or camber
u	backwash
upper	upper surface
v	sidewash
w	downwash

1. METHODS

This section discusses the axis orientation, wing geometries, and the methods used to solve for the circulation around the wing and all other aerodynamic characteristics. It contains all of the relevant equations used in the VLM3D FORTRAN program used to compute the desired results.

1-1 Wing Discretization:

First, the wing is divided up into a finite number of elemental panels or lattices as shown in figure 3, inside of which a finite number of bound vortices are placed. This vortex lattice is now used in place of the real wing to generate the same flow field as the wing and to determine the forces and moments acting on the real wing.

Figure 3. Discretization of the entire wing into a set of panels or lattices.

1-2 Vortex-Strength Computation:

The continuous distribution of bound vorticity over the wing surface is approximated by a finite number of discrete horseshoe vortices as shown below in figure 4. The individual horseshoe vortices are placed in trapezoidal panels laid out in the previous section 1-1. The bound vortex coincides with the quarter-chord line of the panel and is, therefore, aligned with local sweepback angle. The control point, on which the boundary condition of no flow through the wing is imposed, is located at the three-quarter chord line.

Figure 4. Vortex layout, and upwind view of wing.

In this thesis, it is assumed that when the trailing vortices leave the wing, they follow a path parallel to the axis of the vehicle or the free stream. In contrast to this linearized approach, a rigorous theoretical approach assumes that the lattice panels are located on the mean camber surface of the wing and, when the trailing vortices leave the wing, they follow a curved path. However, suitable accuracy can be obtained using the linearized approach, where the computation of the influence coefficients is much simpler, and the geometric coefficients do not change as the angle of attack is varied. Given the panels and vortices previously laid out, the boundary condition of no flow through the wing is imposed, which essentially requires that the flow be tangent to the real wing mean-camber surface. Applying this condition simultaneously to all control points, yields a set of simultaneous equations where the vortex strength is the unknown parameter. This is expressed conveniently in matrix form as,

$$\{C\} = [G]^{-1} \{B\} \quad (1)$$

The reason for choosing the three-quarter-chord line as the control point is seen by referring to figure 5.

Figure 5. (a) Planar airfoil indicating spanwise vortex filaments. (b) airfoil section indicating location of control point where flow is parallel to the surface.

According to figure 5(b), a vortex filament whose strength Γ represents the lifting character of the section is placed at the quarter-chord location. It induces a downwash velocity,

$$w = \frac{\Gamma}{2\pi r} \quad (2)$$

at the control point c , which is at distance r from the vortex filament. If the flow is parallel to the surface at the control point, the incidence of the surface relative to the freestream is given by

$$\alpha \approx \sin \alpha = \frac{U}{U_\infty} = \frac{\Gamma}{2\pi r U_\infty} \quad (3)$$

But, according to the thin airfoil theory,

$$l = \frac{1}{2} \rho_\infty U_\infty^2 c 2\pi\alpha = \rho_\infty U_\infty \Gamma \quad (4)$$

Combination of the above relations yields,

$$\pi \rho_\infty U_\infty^2 c \frac{\Gamma}{2\pi r U_\infty} = \rho_\infty U_\infty \Gamma \quad (5)$$

Thus, solving for r yields

$$r = \frac{c}{2} \quad (6)$$

Therefore, it is clear that the control point where the boundary condition of flow parallel to the airfoil is imposed, is located at the three-quarter-chord location.

Given a three dimensional wing with dihedral at an angle of attack, figure 6 shows the view of the wing chord plane looking from the wing tip towards the root of the wing. Application of the boundary condition of flow parallel to the wing surface at the control

Figure 6. Wing chord plane, when viewed from wing tip towards the wing root.

point in the $\bar{X} - \bar{Z}$ plane shown in figure 6 yields,

$$w \cos \alpha - U \sin \alpha = 0 \quad (7)$$

Referring to figure 7 for the view of the wing span looking upstream towards the trailing edges of the left and right wings, application of the boundary condition in the $\bar{Y} - \bar{Z}$ plane of the left wing shown in figure 7(a) yields,

$$w \cos \alpha \cos \phi_l - v \sin \phi_l - U \sin \alpha \cos \phi_l = 0 \quad (8)$$

Figure 7. Wing span plane viewed downstream towards the trailing edges of the wings

Likewise, for the right wing shown in figure 7(b)

$$w \cos \alpha \cos \phi_r + v \sin \phi_r - U \sin \alpha \cos \phi_r = 0 \quad (9)$$

In the geometry convention of this thesis,

$$\phi = \phi_l = -\phi_r \quad (10)$$

This relationship can be used to show that eqs. (8) and (9) are identical and have the form

$$w \cos \alpha \cos \phi - v \sin \alpha \cos \phi - U \sin \alpha \cos \phi = 0 \quad (11)$$

or, for small angles of attack,

$$w - v \tan \phi \approx U \alpha \quad (12)$$

The downwash velocity induced by a vortex filament of strength Γ_n and a length of dl is given by the law of Biot and Savart as,

$$w = \frac{\Gamma_n (d\vec{l} \times \vec{r})}{4\pi r^3} \quad (13)$$

which in the geometry and formulation of this thesis is written as,

$$w(x, y, z) = \frac{\Gamma}{4\pi} F_w(x', y, z, s, \psi', \phi) \quad (14)$$

Likewise, the sidewash velocity can be expressed as

$$v(x, y, z) = \frac{\Gamma}{4\pi} F_v(x', y, z, s, \psi', \phi) \quad (15)$$

The F_w and F_v influence coefficients are listed on the following page as:

The downwash influence coefficient of eq. (14) is

$$\begin{aligned}
 F_w(x', y, z, s, \psi', \phi) = & \frac{(y \tan \psi' - x') \cos \phi}{(x')^2 + (y \sin \phi)^2 + \cos^2 \phi (y^2 \tan^2 \psi' + z^2 \sec^2 \psi' - 2yx' \tan \psi') - 2z \cos \phi \sin \phi (y + x' \tan \psi')} \\
 & \times \left\{ \frac{(x' + s \cos \phi \tan \psi') \cos \phi \tan \psi' + (y + s \cos \phi) \cos \phi + (z + s \sin \phi) \sin \phi}{[(x' + s \cos \phi \tan \psi')^2 + (y + s \cos \phi)^2 + (z + s \sin \phi)^2]^{1/2}} \right. \\
 & - \frac{(x' - s \cos \phi \tan \psi') \cos \phi \tan \psi' + (y - s \cos \phi) \cos \phi + (z - s \sin \phi) \sin \phi}{[(x' - s \cos \phi \tan \psi')^2 + (y - s \cos \phi)^2 + (z - s \sin \phi)^2]^{1/2}} \Bigg\} \\
 & - \frac{y - s \cos \phi}{(y - s \cos \phi)^2 + (z - s \sin \phi)^2} \left\{ 1 - \frac{x' - s \cos \phi \tan \psi'}{[(x' - s \cos \phi \tan \psi')^2 + (y - s \cos \phi)^2 + (z - s \sin \phi)^2]^{1/2}} \right\} \\
 & + \frac{y + s \cos \phi}{(y + s \cos \phi)^2 + (z + s \sin \phi)^2} \left\{ 1 - \frac{x' + s \cos \phi \tan \psi'}{[(x' + s \cos \phi \tan \psi')^2 + (y + s \cos \phi)^2 + (z + s \sin \phi)^2]^{1/2}} \right\}
 \end{aligned} \tag{16}$$

Likewise, the sidewash influence coefficient of eq. (15) is

$$\begin{aligned}
 F_v(x', y, z, s, \psi', \phi) = & \frac{x' \sin \phi - z \cos \phi \tan \psi'}{(x')^2 + (y \sin \phi)^2 + \cos^2 \phi (y^2 \tan^2 \psi' + z^2 \sec^2 \psi' - 2yx' \tan \psi') - 2z \cos \phi \sin \phi (y + x' \tan \psi')} \\
 & \times \left\{ \frac{(x' + s \cos \phi \tan \psi') \cos \phi \tan \psi' + (y + s \cos \phi) \cos \phi + (z + s \sin \phi) \sin \phi}{[(x' + s \cos \phi \tan \psi')^2 + (y + s \cos \phi)^2 + (z + s \sin \phi)^2]^{1/2}} \right. \\
 & - \frac{(x' - s \cos \phi \tan \psi') \cos \phi \tan \psi' + (y - s \cos \phi) \cos \phi + (z - s \sin \phi) \sin \phi}{[(x' - s \cos \phi \tan \psi')^2 + (y - s \cos \phi)^2 + (z - s \sin \phi)^2]^{1/2}} \Bigg\} \\
 & - \frac{z - s \sin \phi}{(y - s \cos \phi)^2 + (z - s \sin \phi)^2} \left\{ 1 - \frac{x' - s \cos \phi \tan \psi'}{[(x' - s \cos \phi \tan \psi')^2 + (y - s \cos \phi)^2 + (z - s \sin \phi)^2]^{1/2}} \right\} \\
 & + \frac{z + s \sin \phi}{(y + s \cos \phi)^2 + (z + s \sin \phi)^2} \left\{ 1 - \frac{x' + s \cos \phi \tan \psi'}{[(x' + s \cos \phi \tan \psi')^2 + (y + s \cos \phi)^2 + (z + s \sin \phi)^2]^{1/2}} \right\}
 \end{aligned} \tag{17}$$

Then, by using eqs. (14) and (15), eq. (12) can be rewritten as

$$\frac{\Gamma}{4\pi} (F_w - F_v \tan \phi) = U\alpha \quad (18)$$

For a vortex lattice of N elements, eq. (18) can be expressed for a particular control point by

$$\sum_{n=1}^N (F_{w,n} - F_{v,n} \tan \phi_n) \frac{\Gamma_n}{U} = 4\pi \alpha \quad (19)$$

For symmetrical aerodynamic loading on each half of the wing, eq. (19) may be expressed as

$$\sum_{n=1}^{N/2} (\bar{F}_{w,n} - \bar{F}_{v,n} \tan \phi_n) \frac{\Gamma_n}{U} = 4\pi \alpha \quad (20)$$

where

$$\bar{F}_{w,n} = F_{w,n}(x', y, z, s, \psi', \phi)_{\text{left panel}} + F_{w,N+1-n}(x', y, z, s, \psi', \phi)_{\text{right panel}} \quad (21)$$

and

$$\bar{F}_{v,n} = F_{v,n}(x', y, z, s, \psi', \phi)_{\text{left panel}} + F_{v,N+1-n}(x', y, z, s, \psi', \phi)_{\text{right panel}} \quad (22)$$

Figure 3 shows the locations of the elemental panels n and (N+1-n). Finally, the matrix

which is solved is

$$[\bar{F}_{w,n,k} - \bar{F}_{v,n,k} \tan \phi_n] \left\{ \frac{\Gamma_n}{U} \right\} = 4\pi \{ \alpha_k \} \quad (23)$$

1-3 Lift, Pitching, and Rolling Moments For Wings With Dihedral:

The circulation terms Γ_n/U computed in part 1-2 are used in this part to compute the lift and pitching-moment data for planforms with dihedral, and for wing-tail planforms where the planforms are not at the same elevation using entire horseshoe vortices. A special treatment is needed for both types of planforms because there are local sidewash and backwash velocities in addition to the free-stream velocity. The interaction of these velocity components with the spanwise bound vortex provides an additional lift force. Furthermore, the interaction of the sidewash with the chordwise bound vortex which is the portion of the horseshoe vortex trailing leg ahead of the wing trailing edge, results in another lift force.

The Kutta-Joukowski theorem for lift per unit length of a vortex filament is used to compute lift for wings with dihedral, and is given by

$$\tilde{l} = \rho V \Gamma \quad (24)$$

The sidewash velocity described in the preceding paragraph is given by

$$\frac{v}{U} = \frac{1}{4\pi} \sum_{n=1}^{N/2} \frac{\Gamma_n}{U} \bar{F}_{v,n} \quad (25)$$

Horseshoe vortex filaments or their extensions which go through the point at which the velocity is being computed are eliminated from eq. (25), since a line vortex filament cannot induce a velocity on itself. The lift generated along an elemental length of chordwise bound vortex divided by free-stream dynamic pressure and reference wing area is given by

$$\frac{\hat{l}_t}{qS_{ref}} = \frac{2}{S_{ref}} \frac{\Delta\Gamma}{U} c_c \frac{v}{U} \quad (26)$$

where $\Delta\Gamma$ is the local value of circulation and c_c is the chord or elemental length of the chordwise bound vortex. No lift is calculated along the chordwise bound vortex at the root because the sidewash velocity is zero for symmetric loading and geometry.

The lift along the spanwise bound vortex depends on the values of free-stream, backwash, and sidewash velocities and on the circulation at the elemental panel. The sidewash velocity is given by eq. (25) and the backwash velocity is calculated from

$$\frac{u}{U} = \frac{1}{4\pi} \sum_{n=1}^{N/2} \frac{\Gamma_n}{U} \bar{F}_{u,n} \quad (27)$$

where

$$\bar{F}_{u,n} = F_{u,n}(x', y, z, s, \psi', \phi)_{\text{left panel}} + F_{u,N-1-n}(x', y, z, s, \psi', \phi)_{\text{right panel}} \quad (28)$$

and the backwash influence coefficient is

$$F_u(x', y, z, s, \psi', \phi) = \frac{z \cos \phi - y \sin \phi}{(x')^2 + (y \sin \phi)^2 + \cos^2 \phi (y^2 \tan^2 \psi' + z^2 \sec^2 \psi' - 2yx' \tan \psi') - 2z \cos \phi \sin \phi (y + x' \tan \psi')} \quad (29)$$

$$\times \left\{ \frac{(x' + s \cos \phi \tan \psi') \cos \phi \tan \psi' + (y + s \cos \phi) \cos \phi + (z + s \sin \phi) \sin \phi}{\left[(x' + s \cos \phi \tan \psi')^2 + (y + s \cos \phi)^2 + (z + s \sin \phi)^2 \right]^{1/2}} \right.$$

$$\left. - \frac{(x' - s \cos \phi \tan \psi') \cos \phi \tan \psi' + (y - s \cos \phi) \cos \phi + (z - s \sin \phi) \sin \phi}{\left[(x' - s \cos \phi \tan \psi')^2 + (y - s \cos \phi)^2 + (z - s \sin \phi)^2 \right]^{1/2}} \right\}$$

Equations (16), (17), and (29) represent an extension of the original formulation by Glauert (ref. 10) for rectangular horseshoe vortices, later formulation by Campbell (ref. 11) for a spanwise vorticity filament with sweep, and the recent formulation by Blackwell (ref. 12) for a rectangular horseshoe vortex with dihedral respectively. In contrast, the present equations represent a subset of the formulation by Rubbert (ref. 13) in that the trailing legs are constrained to the free-stream direction.

The lift generated by a spanwise bound vortex filament comes from two velocity interactions. One is the total axial velocity interacting with the component of the vortex filament parallel to the $\hat{Y}$ -axis ($2s \cos \phi$). The second is the sidewash velocity interacting with the component of the vortex filament parallel to the $\hat{X}$ -axis ($2s \tan \psi \cos \phi$). The expression for this lift divided by the free-stream dynamic pressure and reference area is

$$\frac{\hat{l}_s}{q_\infty S_{ref}} = \frac{2}{S_{ref}} \frac{\Gamma}{U} (2s) \left[\left(1 - \frac{u}{U} \right) + \frac{v}{U} \tan \psi \right] \cos \phi \quad (30)$$

The lift given by eqs. (26) and (30) is the lift in units of force developed over a span equal to the width of a horseshoe vortex.

The contribution of the lift of the elemental panel to the pitching moment is given by

$$\frac{m_Y}{q_\infty S_{ref} c_{ref}} = \frac{\hat{l}_s}{q_\infty S_{ref}} \frac{\hat{x}_s}{c_{ref}} + \frac{\hat{l}_t}{q_\infty S_{ref}} \frac{\hat{x}_t}{c_{ref}} \quad (31)$$

To get the total wing lift and pitching-moment coefficients, these terms are summed over all the elemental panels which represent the wing and result in the following:

$$C_L = \frac{L}{q_\infty S_{ref}} = 2 \sum_{n=1}^{N/2} \left(\frac{\hat{l}_s}{q_\infty S_{ref}} \right)_n + \left(\frac{\hat{l}_t}{q_\infty S_{ref}} \right)_n \quad (32)$$

$$C_m = \frac{M_Y}{q_\infty S_{ref} c_{ref}} = 2 \sum_{n=1}^{N/2} \left(\frac{m_Y}{q_\infty S_{ref} c_{ref}} \right)_n \quad (33)$$

Eq. (32) gives the values for the lift-curve slope per radian. To obtain the lift-curve slope per degree, the following expression is used

$$C_{L_\alpha} = \left(\frac{L}{q_\infty S_{ref}} \right)_\alpha / 57.29578 \quad (34)$$

The span-load coefficients are obtained from the lift along the spanwise and chordwise bound vortices of each horseshoe vortex. The lift per unit length of span is

$$l = \frac{\hat{l}}{2s \cos \phi} \quad (35)$$

The span-load coefficient for an elemental panel is as follows:

$$\frac{c_l c}{C_L c_{av}} = \frac{\left(\frac{l}{q_\infty c} \right) c}{C_L c_{av}} = \left(\frac{\hat{l}}{q_\infty S_{ref}} \right) \frac{S_{ref}}{C_L 2s_n \cos \phi c_{av}} \quad (36)$$

where

$$c_{av} = \frac{S_r}{b} \quad (37)$$

and

$$T = \frac{S_{ref}}{2s_n \cos \phi c_{av}} \quad (38)$$

so that

$$\frac{c_l c}{C_L c_{av}} = \frac{\hat{l}}{q_\infty S_{ref}} \frac{T}{C_L} \quad (39)$$

Thus, at a particular spanwise location, each of these lifts are summed chordwise and converted to span-load coefficients. The lift along the spanwise bound vortex filament is

$$\left(\frac{c_l c}{C_L c_{av}} \right)_s = T \sum_{i=1}^J \left(\frac{\hat{l}_s}{q_\infty S_{ref}} \right)_i \frac{1}{C_L} \quad (40)$$

and the lift along the chordwise bound vortex filament is

$$\left(\frac{c_l c}{C_L c_{av}} \right)_t = T \sum_{i=1}^J \left(\frac{\hat{l}_t}{q_\infty S_{ref}} \right)_i \frac{1}{C_L} \quad (41)$$

The longitudinal stability parameter about the origin of the X-axis for the wing is given by

$$\frac{\partial C_m}{\partial C_L} = \frac{\left(\frac{M_Y}{q_\infty S_{ref} c_{ref}} \right)_a}{\left(\frac{L}{q_\infty S_{ref}} \right)_a} \quad (42)$$

The pitching moment at zero lift is

$$C_{m_0} = \left(\frac{M_Y}{q_\infty S_{ref} c_{ref}} \right)_{tc} - \frac{\partial C_m}{\partial C_L} \left(\frac{L}{q_\infty S_{ref}} \right)_{tc} \quad (43)$$

In order to determine the damping-in-roll parameter of wings with dihedral, the lift distribution which results from the antisymmetrical span loading must be combined with the appropriate spanwise moment arm. This combination is expressed as

$$C_l = \frac{2}{q_\infty S_{ref} b} \left[\sum_{n=1}^{N/2} (\hat{l}_t \hat{y}_t)_n + \sum_{n=1}^{N/2} (\hat{l}_s \hat{y}_s)_n \right] \quad (44)$$

and, thus

$$C_{l_p} = \frac{\partial C_l}{\partial \left(\frac{pb}{2U} \right)} \approx \frac{C_l}{5\pi / 180} \quad (45)$$

The center of pressure in a spanwise direction is given by,

$$y_{cp} = \frac{\sum_{n=1}^{N/2} \left[\left(\frac{\hat{l}_s}{q_\infty S_{ref}} \right)_{a,n} \hat{y}_{s,n} + \left(\frac{\hat{l}_t}{q_\infty S_{ref}} \right)_{a,n} \hat{y}_{t,n} \right]}{\frac{1}{2} \left(\frac{L}{q_\infty S_{ref}} \right)_a \left(\frac{b}{2} \right)} \quad (46)$$

1-4 Lift, Pitching, and Rolling Moments For Wings With No Dihedral:

The lift, pitching, and rolling moments of wings with no dihedral over any portion of the wing are computed using only the spanwise filament of the horseshoe vortex. All the lift is generated by the free-stream velocity crossing the spanwise vortex filament since there are no sidewash or backwash velocities. For a single elemental panel, the lift per unit length of vorticity is given by

$$\tilde{l} = \rho U \Gamma \cos \psi \quad (47)$$

Since the length of vorticity is $2s/\cos \psi$, the resultant lift is

$$\hat{l} = \tilde{l} \frac{2s}{\cos \psi} \quad (48)$$

Then, the lift per unit span is given by

$$l = \frac{\hat{l}}{2s} = \rho U \Gamma \quad (49)$$

upon nondimensionalization, the following form is obtained

$$\frac{l}{q_{\infty} c_{av}} = \frac{2\Gamma}{c_{av} U} \quad (50)$$

For a chordwise row, application of eq. (50) leads to

$$\frac{c_l c}{c_{av}} = \sum_{i=1}^J \left(\frac{l}{q_{\infty} c_{av}} \right)_i \quad (51)$$

The total lift coefficient is obtained by integrating the lift over the span as given by

$$C_L = \frac{S_r}{S_{ref}} \int_0^1 \frac{c_l c}{c_{av}} d\left(\frac{\hat{y}}{b/2}\right) \quad (52)$$

which is approximated by

$$C_L = \frac{8}{S_{ref}} \sum_{n=1}^{N/2} \frac{\Gamma_n}{U} s_n \quad (53)$$

The lift-curve slope per radian is obtained from a lift coefficient based on the circulation term obtained at 1 radian angle of attack.

The longitudinal stability about $\hat{Y}$ -axis is given by

$$\frac{\partial C_m}{\partial C_L} = \frac{1}{c_{ref}} \frac{\sum_{n=1}^{N/2} \frac{\Gamma_{a,n}}{U} \hat{x}_{s,n} s_n}{\sum_{n=1}^{N/2} \frac{\Gamma_{a,n}}{U} s_n} \quad (54)$$

The pitching moment at zero lift is given by

$$C_{m_0} = \frac{8}{c_{ref} S_{ref}} \sum_{n=1}^{N/2} \frac{\Gamma_{tc,n}}{U} \hat{x}_{s,n} s_n - \frac{\partial C_m}{\partial C_L} C_{L,tc} \quad (55)$$

The center of pressure in a spanwise direction is

$$y_{cp} = \frac{1}{b/2} \frac{\sum_{n=1}^{N/2} \frac{\Gamma_{a,n}}{U} \hat{y}_{s,n} s_n}{\sum_{n=1}^{N/2} \frac{\Gamma_{a,n}}{U} s_n} \quad (56)$$

The span-load coefficient is

$$\frac{c_l c}{C_L c_{av}} = \frac{\frac{b}{2} \sum_{i=1}^J \frac{\Gamma_i}{U}}{2 \sum_{n=1}^{N/2} \frac{\Gamma_n}{U} s_n} \quad (57)$$

The same procedure used to compute the damping-in-roll parameter for wings with dihedral is used to compute C_{lp} for zero-dihedral wing planforms, except the contribution of the chordwise bound vortex is eliminated. Therefore, eq. (44) becomes,

$$C_l = \frac{2}{q_\infty S_{ref} b} \left[\sum_{n=1}^{N/2} 2 \left(\frac{\Gamma}{U} \right)_n \hat{y}_{s,n} 2s_n \right] \quad (58)$$

and likewise,

$$C_{lp} \approx \frac{C_l}{5\pi / 180} \quad (59)$$

1-5 Aerodynamic Characteristics of the Various Wings:

The output data include the damping-in-pitch parameter, the lift coefficient due to pitch rate, the induced drag parameter, the angle of attack for zero lift, the angle of attack for the desired lift coefficient, the basic span load distribution, and the additional span load distribution. The pitch derivatives are computed by using the vortex strengths obtained with the boundary condition values which represent a constant pitching motion. These vortex strengths are used to compute C_L and C_m which, in turn, are used to compute C_{mq} and C_{Lq} as follows:

$$C_{mq} = \frac{\partial C_m}{\partial \left(\frac{qc}{2U} \right)} \approx \frac{C_m}{\frac{5\pi}{180} \frac{c_{ref}}{2}} \quad (60)$$

and

$$C_{Lq} = \frac{\partial C_L}{\partial \left(\frac{qc}{2U} \right)} \approx \frac{C_L}{\frac{5\pi}{180} \frac{c_{ref}}{2}} \quad (61)$$

The far-field method of computing the induced drag parameter is based on the lifting-line concepts employed in the Trefftz plane by Munk and the induced drag parameter

thereby obtained is expressed mathematically as

$$\frac{C_{D,i}}{C_L^2} = \frac{b^2}{C_L^2 S_{ref}} \int_{-1}^1 \gamma \alpha_i d\eta \quad (62)$$

The above equation can give good results for wings without dihedral, but still should be used only as a guide for wings with dihedral, since no vertical displacement of the span loadings is taken into account. For wings with dihedral, other methods or the near-field method should be used to compute the induced drag. Eq. (62) is programmed into the code using a reformulation by Multhopp (ref. 14), where he uses a quadrature formula and the procedure used to represent the variation of γ with $\sin^{-1}\eta$ is a linear curve fit between consecutive pairs of data points. This curve fit requires that a sufficient number of data points be available near the wing tip where the gradient of the $\gamma - \sin^{-1}\eta$ curve is the greatest. Therefore, even for wings without dihedral, good results can only be expected for the far-field method when a large number of chordwise rows of horseshoe vortices are specified.

The near-field computation for the induced drag is based on combining for each elemental panel the lift and leading-edge thrust as follows:

$$\frac{d_u}{q_\infty} = \alpha \frac{l}{q_\infty} - \frac{t}{q_\infty} \quad (63)$$

where the lift per unit span l/q_∞ is computed by eq. (50) for planforms without dihedral and by eqs. (26) and (30) for planforms with dihedral. The leading-edge thrust per unit span is computed by using the Kutta-Joukowski theorem where the induced and free-stream velocity components parallel to the $\bar{Y} - \bar{Z}$ plane interact with the spanwise bound

vortex filament as follows:

$$\frac{t}{q_\infty} = -2 \left(\frac{w}{U} - \frac{v}{U} \tan \varphi - \alpha \right) \left(\frac{\Gamma}{U} \right)_{u,rad} \quad (64)$$

There is no contribution of the chordwise bound vortex filaments to the leading-edge thrust, and the above equation is evaluated at 1 radian angle of attack. Furthermore, the circulation used is the one due to additional loading only.

These results are then summed along each chordwise row to get the section leading-edge thrust,

$$\frac{c_l c}{2b} = \frac{1}{2b} \sum_{i=1}^J \left(\frac{t}{q_\infty} \right)_i \quad (65)$$

From eq. (65), the section suction coefficient is computed as,

$$\frac{c_s c}{2b} = \frac{\left(\frac{c_l c}{2b} \right)}{\cos \Lambda} \quad (66)$$

Then, the section induced drag for a chordwise row of horseshoe vortices is

$$\frac{c_{d,i} c}{2b} = \alpha \left(\frac{c_l c}{C_L c_{av}} \right) \frac{c_{av} S_{ref} (C_{L_{\alpha}})_{rad}}{2b S_\tau} - \frac{c_l c}{2b} \quad (67)$$

Finally, the near-field solution for the induced drag parameter is

$$\frac{C_{D,i}}{C_L^2} = \frac{4b}{S_{ref} (C_{L_{\alpha}})_{rad}^2} \sum_{k=1}^{\bar{N}_s} \left(\frac{c_{d,i} c}{2b} \right)_k 2s_k \cos \phi_k \quad (68)$$

In addition, the leading-edge thrust and suction coefficients are computed similarly as

$$C_T = \frac{2}{S_{ref}} \sum_{k=1}^{\bar{N}_s} \left(\frac{c_l c}{2b} \right)_k 2s_k \cos \phi_k \quad (69)$$

and

$$C_s = \frac{2}{S_{ref}} \sum_{k=1}^{\bar{N}_s} \left(\frac{c_s c}{2b} \right)_k 2s_k \cos \varphi_k \quad (70)$$

The angle of attack for zero lift is computed by

$$\alpha_o = -\frac{C_{L,tc}}{C_{L_\alpha}} \quad (71)$$

The angle of attack required for the additional loading and basic loading combined to produce the input value of the desired lift coefficient is

$$\alpha_d = \frac{C_{L,d}}{C_{L_\alpha}} + \alpha_o \quad (72)$$

The basic load due to twist and/or camber is the load on the wing when the lift coefficient is zero. This load is obtained from the values of $c_l c / c_{av}$ for each elemental panel as,

$$\left(\frac{l}{q_\infty c_{av}} \right)_b = \left(\frac{l}{q_\infty c_{av}} \right)_{tc} - \left(\frac{l}{q_\infty c_{av}} \right)_a \frac{C_{L,tc}}{C_{L,a}} \quad (73)$$

Eq. (73) is then summed for each chordwise row for the span load distribution of basic load to give,

$$\left(\frac{c_l c}{c_{av}} \right)_B = \sum_{i=1}^J \left(\frac{l}{q_\infty c_{av}} \right)_{i,b} \quad (74)$$

The span load distribution at the input value of the desired lift coefficient is

$$\left(\frac{c_l c}{c_{av}} \right)_d = \left(\frac{c_l c}{c_{av}} \right)_B + \sum_{i=1}^J \left(\frac{l}{q_\infty c_{av}} \right)_{i,a} \frac{C_{L,d}}{C_{L,a}} \quad (75)$$

The incremental pressure coefficient is defined as

$$\Delta C_{p,n} = \frac{(p_{lower} - p_{upper})_n}{q_\infty} \quad (76)$$

Since the pressure is assumed to be uniform over an elemental panel,

$$\Delta C_{p,n} = \frac{(l/c)_n}{q_\infty} \quad (77)$$

which is used in the program. For planforms without dihedral, eq. (77) can be expressed as

$$\Delta C_{p,n} = \frac{\rho U \Gamma_n / c_n}{q_\infty} = \frac{2}{c_n} \frac{\Gamma_n}{U} \quad (78)$$

2. RESULTS

2-1 Computer Program Modifications:

At the beginning of this thesis, the VLM3D computer program was not running properly. Upon examination of the code, it was discovered that all of the formatting statements for the data had been removed.

The input file for the program is formatted, and it is extremely important to have the correct number of spacing between the data, and the data in the proper decimal format. Otherwise, if the input data are not properly arranged, the program will not execute in the correct manner. It appears that someone had previously attempted to run the code, and may have had an improperly formatted input file. In order to correct for this mistake, instead of rewriting the input file into the proper format, all of the format statements in the program were removed in the hope of alleviating the execution problem of the program. However, this was not the proper fix, since the program uses the formatting statements to read in the corresponding correct data for a given variable, and to place the decimal point in the correct position in the data. Therefore, without the formatting statements, the program would not execute properly since it could not read in all of the required data for the given input variables, and whatever it did read was not in the proper decimal format.

To correct this problem, at the beginning of the this thesis all of the required format statements were put back in the proper positions in the code. The correct operation of the program was validated by running some sample cases which Margason and Lamar (ref. 1) had run, and then comparing the two results. Comparison of the results indicated that the errors in

the program had been corrected, and the program was executing properly. Furthermore, to avoid any future problem of a misformatted input file, a template (Appendix 2) was generated for the input data to assure that the data were spaced apart in the correct manner.

2-2 Comparison of VLM3D To Other Programs:

Some of the more common linear panel methods in use today are listed in table 1 below. Due to the fact that the linear VLM program does not include source distribution, it is incapable of including effects of thickness. However, the solutions of the matrix equations associated with the linear VLM generally converge, except for configurations with multiple planforms where special care must be taken when the trailing vortices shed from the trailing edges of one planform may intersect a second planform or its wake. In the methods using the doublet singularity distribution, PAN AIR is the most advanced method which uses a linear source distribution and a quadratic distribution on each panel. But, it has been found that numerical problems may arise in these methods if the continuity of the doublet strengths at the edges of matching panels is not ensured. Therefore, PAN AIR spline fits the doublet strength

Table 1. Common Linear Panel Methods In Use And Their Features

Method	Panel Geometry	Strength Distribution of Doublet or Vortex across each panel		Reference
		chordwise direction	spanwise direction	
VLM3D	Planar Quadrilateral	constant	constant	Margason et al. 1971
VSAERO	Planar Quadrilateral	constant	constant	Maskew 1982
QUADPAN	Planar Quadrilateral	constant	constant	Cooper-Smith 1983
MCAERO	Quadrilateral with 8 subpanels	quadratic	quadratic	Bristow 1983
PAN AIR	Quadrilateral with 8 subpanels	quadratic	quadratic	PAN AIR 1980

between neighboring panels to ensure complete doublet strength continuity. PAN AIR is the probably the most versatile method, since it can be used on very complex configurations with many options for boundary conditions such as simulations for engine intake and exhaust flows. However, due to its complexity, PAN AIR is most expensive in terms of computer resources and CPU time.

VLM, QUADPAN, and MCAERO use second order expansions to calculate the pressure relations at increasing Mach number, and according to Rom (ref. 4) their results are in good qualitative agreement with experimental data. Furthermore, VSAERO and PAN AIR use the isentropic pressure relation and under-predict the lift increase at increasing Mach number. This is because VLM, QUADPAN, and MCAERO use a better compressibility relation than VSAERO and PAN AIR which offsets the effect of the lower-order panel method than the latter two. From data presented by Rom (ref. 4), for subsonic flows there is no particular advantage to the high order linear panel methods for accuracy of the evaluation of the aerodynamic characteristics. Furthermore, according to Rom (ref. 4), when the matrix calculations for the linear panel methods converge, the panel loading values and the associated aerodynamic coefficients are about the same for all reasonable panel methods. The high order methods may have advantages in ensuring convergent solutions of the matrix equations and eliminating spurious jumps at panel interfaces that may be encountered in some panel methods in supersonic flows. Overall, the low order panels methods such as VLM3D run extremely fast on current workstation computers in contrast to higher order methods, and also yield quite reasonable results. Therefore, VLM3D is a very nice program to use for preliminary design, which the program authors had intended.

2-3 Wing Shapes For The Four Cases Studied:

A detailed study of four wing shapes was performed, and the results are discussed in this thesis. Tables 2 and 3 below, list the geometric data along with the number of rows and vortices used for the solution by VLM3D for all four cases. The first case studied is a fixed sweep wing with dihedral, twist and camber. On the following page, figure 8 shows a plot of the wing shape along with the locations of the control points where the boundary condition is imposed. Figure 9, shows the dihedral angle and spanwise view of the wing. The second case studied (fig. 10), is a fixed sweep wing with twist and camber with exactly the same dimensions except that it has no dihedral. The twist used for cases 1 and 2 is exactly the same, and for more detailed data on the nature of the twist, one should refer to appendix 2 and view the local angles of attack in the third section of the group two data for the VLM3D input files. The

Table 2. Geometric Data For The Four Cases Studied

Case	Dihedral	Twist	Average Chord (in)	True Wing Area (in ²)	True AR	B/2 (in)
1	yes	yes	60.703	6138.254	1.66582	50.6
2	no	yes	60.703	6138.254	1.66582	50.6
3	yes	no	60.703	6138.254	1.66582	50.6
4	no	no	60.703	6138.254	1.66582	50.6

Table 3. VLM3D Setup For The Four Cases

Case	Total Number of Horseshoe Vortices	Number of Chordwise rows of Elemental Panels Along the Span	Number of Horseshoes in Each Chordwise Row
1	100	10	10
2	110	11	10
3	100	10	10
4	110	11	10

Figure 8. Wing shape and control points for cases 1 and 3.

Figure 9. Dihedral angles for cases 1 and 3.

third case studied (figs. 8 & 9), is a fixed sweep wing with dihedral and no twist and camber. Finally, the fourth case studied (fig. 10), is a fixed sweep wing with no dihedral and no twist and camber. Other than the differences in dihedral and twist, all wings have the same planforms. Essentially, cases 3 and 4 are replicas of cases 1 and 2 respectively without any twist or camber.

Figure 10. Wing shape and control points for cases 2 and 4.

2-4 Lifting Principles of Delta Wings:

The dominant flow characteristics around a delta wing with a sharp leading edge at an angle of attack are the two vortex cores shown in figure 11, which are formed on top of the wing due to the pressure difference between the lower and upper surfaces of the wing. Due to the higher pressure on the bottom surface of the wing, the flow from the bottom surface of the wing tries to curl around the leading edge from the bottom to the top. Since the leading edge is sharp, this curling flow will separate along the entire length of the leading edge, and will form a shear free layer or the primary vortex on the upper surface of the wing just inboard of the leading edge. The reason that the flow separates along the sharp leading edge is due to the fact that

Figure 11. Subsonic flow over a delta wing at an angle of attack α .

inviscid flow theory predicts an infinite velocity at a sharp corner, when low-speed, subsonic flow passes over a sharp convex corner and nature copes with this situation by having the flow separate at the corner. Thus, as the primary vortex hits the upper surface of the wing, the flow reattaches at the primary attachment line shown on figure 11. There is a great deal of axial component of motion in this core of high vorticity, and the fluid spirals around and along the axis. Due to this, a spanwise outflow is induced on the upper surface beneath the primary vortex, and the flow separates again as it approaches the leading edge. This is the secondary vortex which is formed underneath the primary vortex. The size and strength of the primary and secondary vortices increase with increasing angle of attack and dominate the flow which remains steady. It is the formation of these vortices that causes the nonlinear behavior of aerodynamic characteristics over the angle-of-attack range.

The leading-edge vortices are extremely strong and stable, and being a source of high energy, and relatively high vorticity flow create a region of very low local static pressure in the vicinity of the vortices on the upper surface of the wing and along the leading edges. Over the middle section of the wing, the surface pressure on the upper surface of the wing is higher and rather constant, but still lower than freestream. On the bottom surface of the wing, the surface pressure is essentially constant and higher than the freestream. Therefore, due the presence of the higher than freestream pressure on the bottom surface of the wing and the extremely low local static pressure in the vicinity of the vortices and the leading edges, a “suction on the wing” effect occurs. Essentially, the leading-edge vortices are creating a strong suction on the top surface of the wing and along the leading edge, thus pulling the wing forward and up. The forward pulling effect of the vortices due to the “suction” along the leading edges is known as

the leading edge thrust, and its characteristics are studied in this thesis by calculating the leading edge thrust coefficient C_T . Likewise, the upward pulling effect of the vortices due to the “suction” in the vicinity of the vortices on the upper surface of the wing is known as the leading edge suction, and its characteristics are studied in this thesis by calculating the leading edge suction coefficient C_s .

2-5 The Pressure Coefficient Distributions:

The extremely strong and stable leading-edge vortices have a great deal of energy and a relatively high-vorticity flow which causes the local static pressure in the vicinity of the vortices to be small. Therefore, the surface pressure on the upper surface of the delta wing is reduced near the leading edge and is higher and reasonably constant over the middle of the wing. On the following pages, figures 12 a, b, c, and d show the distribution of the difference of the lower and upper pressures for case 1 at Mach numbers of 0.1, 0.4, 0.7, and 0.9 respectively. On pages 42 and 43, figures 13 a, b, c, and d show the distribution of ΔC_p between the lower and upper surfaces for case 2 at Mach numbers of 0.1, 0.4, 0.7, and 0.9 respectively. Likewise, on pages 44 and 45, figures 14 a, b, c, and d show the distribution of ΔC_p between the lower and upper surfaces for case 3 at Mach number of 0.1, 0.4, 0.7, and 0.9 respectively. Finally, on pages 46 and 47, figures 15 a, b, c, and d show the distribution of ΔC_p between the lower and upper surfaces for case 4 at Mach numbers of 0.1, 0.4, 0.7, and 0.9 respectively.

(a)

(b)

Figure 12. Plot of Pressure Coefficient Difference Between Lower and Upper Wing

Surfaces (ΔC_p) for case 1. (a) $M_\infty=0.1$ (b) $M_\infty=0.4$

(c)

(d)

Figure 12. Plot of Pressure Coefficient Difference Between Lower and Upper Wing

Surfaces (ΔC_p) for case 1. (c) $M_\infty=0.7$ (d) $M_\infty=0.9$

(a)

(b)

Figure 13. Plot of Pressure Coefficient Difference Between Lower and Upper Wing

Surfaces (ΔC_p) for case 2. (a) $M_\infty=0.1$ (b) $M_\infty=0.4$

(c)

(d)

Figure 13. Plot of Pressure Coefficient Difference Between Lower and Upper Wing

Surfaces (ΔC_p) for case 2. (c) $M_\infty=0.7$ (d) $M_\infty=0.9$

(a)

(b)

Figure 14. Plot of Pressure Coefficient Difference Between Lower and Upper Wing Surfaces (ΔC_p) for case 3. (a) $M_\infty=0.1$ (b) $M_\infty=0.4$

(c)

(d)

Figure 14. Plot of Pressure Coefficient Difference Between Lower and Upper Wing Surfaces (ΔC_p) for case 3. (c) $M_\infty=0.7$ (d) $M_\infty=0.9$

(a)

(b)

Figure 15. Plot of Pressure Coefficient Difference Between Lower and Upper Wing

Surfaces (ΔC_p) for case 4. (a) $M_\infty=0.1$ (b) $M_\infty=0.4$

(c)

(d)

Figure 15. Plot of Pressure Coefficient Difference Between Lower and Upper Wing

Surfaces (ΔC_p) for case 4. (c) $M_\infty=0.7$ (d) $M_\infty=0.9$

Examination of all plots validates the fact that the surface pressure is reduced near the leading edge and higher and reasonably constant over the middle of the wing due to the primary and secondary vortices. Furthermore, it is also seen that the pressure coefficient difference is highest at the root section of the wing near the mid-chord section of the wing. Figures 12 and 14 also show that for the cases of the wings with a dihedral angle, there is a large negative ΔC_p distribution at the inboard section of the wing span and it is maximum at the root section of the wing. These figures also show that the ΔC_p distribution becomes positive towards the tip section of the wing with a maximum at about quarter span. On the other hand, figures 13 and 15 for the cases without wing dihedral show a smooth and positive ΔC_p distribution throughout the span of the wing. This great variation in ΔC_p distribution between wings with dihedral and without dihedral is probably caused by the interference of the vortices on the upper surfaces of the wing caused by the presence of the first positive dihedral angle of 5 degrees. Since the wing section is slanted up a little from the first dihedral angle to the second negative dihedral angle, it may cause the vortices generated by this wing section to be forced inboard towards the vortices generated by the straight inboard section. This being the case, the vortices from the two sections interfere with each other, and there is no longer a strong, steady and stable vortex core on top of the inboard section, thus causing a rather turbulent and chaotic flow. This effect explains the cause for the negative distribution of ΔC_p along the inboard section of the wing span. Since cases 2 and 4 have no dihedral, then there is no such interference of vortices on the top surface of the wing, thus a strong, steady and stable vortex core would exit resulting in a smooth positive ΔC_p distribution, which is shown by figures 13 and 15.

Figure 16 shows the spanwise distance in fraction of semispan from root chord to center of pressure as a function of Mach number for all cases, where the vertical axis has been scaled up for clarity. Examination of this figure indicates, that with increasing Mach numbers the location of the center of pressure on wings with dihedral behaves in a way exactly opposite of the wings with no dihedral. Figure 16 shows that with increasing Mach number the center of pressure for wings with dihedral (cases 1 & 3) moves inboard towards the root section of the wing. In contrast, the center of pressure for the wings without dihedral (cases 2 & 4) moves outboard towards the tip of the wing. This effect is explained by the ΔC_p distribution plots, where it was seen that wings with dihedral have a large negative ΔC_p distribution at the root section of wing and then a positive distribution

Figure 16. Spanwise location of Center of Pressure vs. Mach Number

along the rest of the span. Comparison of figures 12 a, b, c, d and figures 14 a, b, c, and d for increasing Mach numbers show that for increasing Mach numbers the magnitude of the negative ΔC_p distribution at the root section increases more rapidly than the magnitude of the positive distribution along the rest of the span. Thus, with increasing Mach numbers, the presence of this large negative magnitude in ΔC_p causes the center of pressure to move inboard.

In contrast, cases 2 and 4 have no wing dihedral and figures 13 a, b, c, d and 15 a, b, c, and d show that for increasing Mach numbers, the magnitude of the ΔC_p distribution is positive throughout the wing span. Furthermore, these plots also show that for increasing Mach numbers, the magnitude of ΔC_p increases slightly faster at the outboard sections of the wing as compared to the root section. Thus, with increasing Mach numbers, this causes the center of pressure to move outboard towards the wing tips.

2-6 Load Distributions:

On the following pages, figures 17, 18, 19, and 20 show the distribution of the span load coefficient along the left wing span at $C_L=1.0$ for cases 1, 2, 3, and 4 respectively. This span load is the combination of the basic span-load-coefficient at $C_L=1.0$, and the additional span-load-coefficient required to produce zero lift when combined with lift due to twist and camber at $C_L=1.0$. Examination of figures 17 and 19 for cases 1 and 3 respectively, indicates that they are exactly identical, where the only physical difference between the two wings is that case 1 has twist and case 3 does not. Therefore, for wings with dihedral, it is clearly seen that twist does not affect the distribution of the span load

Figure 17. Case 1: Distribution of Combination of Basic Span Load and Additional Span Load Coefficients at $C_L=1.0$ along the left semispan.

Figure 18. Case 2: Distribution of Combination of Basic Span Load and Additional Span Load Coefficients at $C_L=1.0$ along the left semispan.

Figure 19. Case 3: Distribution of Combination of Basic Span Load and Additional Span Load Coefficients at $C_L=1.0$ along the left semispan.

Figure 20. Case 4: Distribution of Combination of Basic Span Load and Additional Span Load Coefficients at $C_L=1.0$ along the left semispan.

coefficient. Further examination of these two plots shows that there is a spike peak in the span load coefficient distribution which corresponds with the second dihedral angle of -25 degrees, when figure 9 is referred to for the locations of the dihedral angles. This is mainly due to the fact that the drooped section of the wing acts as a leading edge vortex flap (LEVF), which is a mechanical configuration where the leading-edge is deflected downward through a variable angle. This LEVF would show some projected frontal area when the wing is viewed along the $\hat{X} - \hat{Y}$ plane from the upstream, and the pressure over this frontal area will be low, thus increasing the load and possibly the drag associated with it. Wings with LEVF have a higher L/D ratio for a given C_L than plain flat delta wings.

Examination of figures 18 and 20 for cases 2 and 4, respectively, again shows that twist has no effect on the distribution of the span load coefficient, since the two figures are exactly identical. In these two figures, case 2 is a wing with twist and no dihedral and case 4 is a wing with no twist and no dihedral. These two figures also show that for wings without dihedral, the span load coefficient distribution is smooth and does not have any spikes associated with the location of the dihedral angles like that seen in figures 17 and 19. Furthermore, all figures indicate that the span load coefficient is higher at the root than the wing tip location, which is quite expected by the effects discussed in the lifting principles of delta wings, and by the ΔC_p distribution plots. In all plots, there is minimal deviation in the load distribution with increasing Mach numbers except for the region near the wing droop angle for wings with dihedral.

2-7 The Lift Curve and the Lift Coefficient Properties:

Below, figure 21 shows the lift curve for all four cases at a subsonic cruise of Mach 0.9. It indicates that the highest curve is achieved by case 4, which is the wing with no dihedral and no twist. Furthermore, comparison of case 1 with case 3 and case 2 with case 4 shows that the lift curve is shifted downwards slightly by the effects of the twist on the wing for wings with the same dihedral or no dihedral respectively. This comparison also shows that the value of C_L at zero angle of attack is negative for cases with wing twist. Plot of the contribution of twist on the lift coefficient is shown on the next page, where figure 22 shows the lift coefficient due to twist at zero angle of attack versus the Mach number for all four cases. Figure 22 indicates that the lift coefficient due to twist for cases 3 and 4 is zero, which is quite expected since these two wings have no twist. Having already discovered on figure 21 that the twist used for cases 1 and 2 cause a lower lift coefficient, examination of figure 22 indicates that case 1 has a lesser negative

Figure 21. Lift curve for all cases at Mach=0.9

Figure 22. Lift Coefficient due to twist at zero angle of attack vs. Mach Number

C_L due to twist than case 2. Therefore, the wing dihedral lessens the effect of the twist for the twist configuration used in cases 1 and 2.

On further examination of figure 21 and comparison of case 1 with case 2 and case 3 with case 4, it is seen that for wings with the same twist or no twist, respectively, the dihedral decreases the slope of the lift curve. This is better seen in on the following page in figure 23, where the lift curve slope per degree versus the Mach number is plotted. Figure 23 clearly shows that the two wings without dihedral (cases 2 & 4) have a higher C_L slope than wings with dihedral (cases 1 & 3). This again can be attributed to the effect of the interference of the vortices on the upper surface of the wing discussed earlier, and shown by the ΔC_p plots, which as discussed cause a large negative distribution of ΔC_p at

Figure 23. Lift Curve slope per degree vs. Mach number

the root section of the wing. In contrast, wings without dihedral had a smooth positive distribution of ΔC_p throughout the wing span. Therefore, it is not surprising that wings with dihedral would have smaller C_L than wings without dihedral and this is seen on figures 21 and 23.

On the following page, figure 24 shows the angle of attack for zero lift versus the Mach number for all four cases. By examination of figure 21, it has already been noted that the twist configuration for cases 1 and 2 caused the lift curve to shift downward. This result is better illustrated on figure 24, where it is clearly seen that for cases 1 and 2 the angle for zero lift is a positive angle of attack. The figure also indicates that in cases with no twist (cases 3 & 4), the angle of attack for zero lift is at zero degrees, which is quite

expected. Furthermore, figure 24 also shows once again that the dihedral configuration used lessens the effects of the twist configuration used, since it is clearly seen that case 1 has a smaller positive angle of attack for zero lift than case 2. These two cases have the same twist, but case 1 also has a dihedral which case 2 does not have. This effect of the wing dihedral of lessening the effects of the wing twist configuration used was also seen earlier by figures 21 and 23. Even though the dihedral lessens the negative impact of the wing twist, it still has a much larger negative impact on the C_L all by itself. Therefore, due to the negative impacts of both the twist and dihedral configuration used on the C_L , the flat delta wing without any dihedral and twist (case 4) has the most desirable lift curve as was indicated earlier on figure 21. On the following pages, figures 25, 26, 27, and 28 show individual and more detailed lift curves for cases 1, 2, 3, and 4 respectively.

Figure 24. Angle of attack for zero lift vs. Mach number

Figure 25. The Lift Curve (C_L vs. α) for case 1 at various Mach numbers

Figure 26. The Lift Curve (C_L vs. α) for case 2 at various Mach numbers

Figure 27. The Lift Curve (C_L vs. α) for case 3 at various Mach numbers

Figure 28. The Lift Curve (C_L vs. α) for case 4 at various Mach numbers

2-8 The Induced Drag Coefficient:

Figure 29 below shows the drag bucket curve, which is the plot of the induced drag coefficient as a function of the lift coefficient for all four cases at a subsonic Mach number of 0.9. Examination of the plot yields the fact that the cases with the lowest $C_{D,i}$ curves are cases 2 and 4. Although the $C_{D,i}$ curves for these two cases fall exactly on top of each other on the plot, examination of the actual output data showed that case 4 had slightly a slightly lower induced drag coefficient than case 2. This difference was on the order of 10^{-4} , which cannot be shown by the plot due to the scale used. Figure 24 also indicates that the induced drag coefficients for cases 1 and 3 are also extremely close to each other

Figure 29. The Induced Drag Coefficient ($C_{D,i}$) vs. the C_L for all cases at Mach=0.9

with a more pronounced difference than that seen between cases 2 and 4. Looking at the similarity of the results between cases 1 and 3 and that between cases 2 and 4, it is seen that the twist configuration used in the wing has little effect on the induced drag coefficient. This was also the case in the load distribution discussed in section 2-6. But the dihedral does play a large role on the induced drag coefficient as evidenced by figure 29. As seen on the figure, there is a large difference between the data of the dual set of cases of 1&3 and that of 2&4, where the only remaining difference is the wing dihedral. The twist has been excluded from the difference criterion here, since it has already been noted that the twist has little effect due to the similarity of results in the data sets 1&3 and in 2&4. Unfortunately, the large role that the dihedral plays has an adverse effect on the induced drag coefficient. Therefore, once again, as was the case in the lift coefficient behavior, the twist and dihedral configuration used on the delta have an adverse effect on the induced drag coefficient. Not only do the twist and the dihedral result in a lower C_L , but they also result in a higher $C_{D,i}$.

On the following pages, figures 30, 31, 32, and 33 show the individual drag bucket curves for cases 1, 2, 3, and 4 at various Mach numbers respectively. Through examination of figures 31 and 33, it is seen that with increasing Mach numbers, the induced drag coefficient curve does not increase for given lift coefficients for cases 2 and 4 respectively. But as seen on figures 30 and 32, the $C_{D,i}$ curve does shift upwards for increasing Mach numbers for cases 1 and 3 respectively, and a penalty is paid for increasing Mach numbers. Therefore, since cases 1 and 3 have wing dihedral, it is seen once again that the wing dihedral configuration used is disadvantageous.

Figure 30. The Induced Drag Coeff. ($C_{D,i}$) vs. C_L for case 1 at various Mach numbers

Figure 31. The Induced Drag Coeff. ($C_{D,i}$) vs. C_L for case 2 at various Mach numbers

Figure 32. The Induced Drag Coeff. ($C_{D,i}$) vs. C_L for case 3 at various Mach numbers

Figure 33. The Induced Drag Coeff. ($C_{D,i}$) vs. C_L for case 4 at various Mach numbers

On the following page, figures 34 and 35 show distribution of the section induced drag coefficient along the semi-span of the left wing for cases 1 and 3 and cases 2 and 4 at various Mach numbers respectively. Examination of figure 34 shows a spike peak which corresponds to the second negative dihedral angle present in cases 1 and 3, which starts the drooped section of the wing, just like it was the case with the spanwise load distribution in section 2-6. This high drag could be associated with the fact that a ridge is formed at this location of the wing, where the wing span slopes downwards on both sides of the location of the second dihedral angle. Therefore, due to the slope, the vortices on the wing span section inboard of the dihedral angle are forced inboard towards the root section of the wing, which as discussed earlier may interfere with the vortices from the flat root section of the wing. Likewise, the vortices on the wing span section on the outboard side of the dihedral angle are forced outboard towards the tip section of the wing.

Thus, there is probably very little vortex presence on the ridge area of the wing span formed by the negative dihedral angle. Furthermore, should the vortices not be forced away from each other, a problem would still exist at the ridge section. As the primary vortex is generated by leading edges of the first flat root span and second sloped up middle span wing sections, it will separate from the wing surface as it passes the ridge formed by the second negative dihedral angle. The reason being that the flow is forced upward by the positive sloped section of the wing span, and then after the second negative dihedral angle, the wing suddenly disappears from underneath the flow as it droops downward. Therefore, the flow will detach from the wing surface, and a region of

Figure 34. Spanwise Distribution of Nondimensional Section Induced Drag Coefficient for cases 1 and 3.

Figure 35. Spanwise Distribution of Nondimensional Section Induced Drag Coefficient for cases 2 and 4.

turbulent and separated flow with high drag will exit. This of course is illustrated by the spike in drag at the ridge section of the wing formed by the second dihedral angle.

Further examination of figure 34 also yields the fact that there is a negative sectional induced drag distribution near the wing tip section of the span. This is the interesting concept of wing thrust generated by the vortices on the surface of delta wings. Furthermore, it is clear that most of the thrust is generated at the drooped section of the wing which acts as a LEVF, where a region of extremely low pressure is present, thus causing the lift and thrust characteristics of the wing to increase. This concept of wing thrust and its properties are discussed in more detail later in section 2-10.

Examination of figure 35 for cases 2 and 4 shows that there is a smooth distribution of the section induced drag coefficient along the wing span, and that there are no spikes associated with wing dihedral like that seen on figure 34. Once again the effect of wing thrust associated with delta wings is seen near the wing tip section, where the section induced drag coefficient is negative.

Comparison of figures 34 and 35 indicates that even though cases 1 and 3 have a lower $C_{D,i}$ $C/2B$ distribution at the wing root and tip sections than cases 2 and 4, the overall drag for cases 1 and 3 is still higher than cases 2 and 4, mainly due to the large spike associated with the wing dihedral. This explains the previous results gained by the examination of figure 29 for $C_{D,i}$ vs C_L . Furthermore, even though cases 1 and 3 have a higher wing thrust generated at the wing tips than cases 2 and 4, the overall drag is still higher in cases 1 and 3 than cases 2 and 4 due once again to the large spike in drag associated with the wing dihedral angles.

2-9 The Induced Drag Parameter:

Below, figure 36 shows a plot of the induced drag parameter as a function of the lift coefficient for all cases at a subsonic cruise Mach number of 0.9. Examination of the plot indicates once again that the lowest results are achieved by case 4, where there is no dihedral and no twist on the wing. Furthermore, it also shows that the highest results are obtained for case 1, where the wing has both twist and dihedral. Therefore, once again the results indicate that the twist and dihedral configuration used on the wings are disadvantageous. Comparison of the curves for cases 1 and 2 indicates that case 1 has a much higher drag parameter as a function of C_L than case 2, where both wings have the

Figure 36. The Induced Drag Parameter ($C_{D,i}/C_L^2$) based on the near field solution vs. C_L for all cases at Mach=0.9.

same twist with case 1 having an added dihedral angle configuration. Hence, the dihedral configuration used is once again found to be disadvantageous. A similar conclusion is reached by comparison of the curves for cases 3 and 4.

Examination of all curves on the plot shows that cases 1 and 2 have non-linear drag parameter characteristics as functions of C_L and the drag parameter is constant as a function of C_L for cases 3 and 4. Looking at the common configuration of characteristics of the wing for all cases, it is found that cases 1 and 2 both have a twist configuration while cases 3 and 4 do not. Hence, it is clear that the wing twist configuration used causes a non-linear behavior in the drag parameter as a function of C_L . Further examination of the non-linear behavior of cases 1 and 2 shows that the wing with both dihedral and twist (case 1) has a much larger spike than the wing with only a twist configuration (case 2). Therefore, the addition of a wing dihedral angle, intensifies the non-linear behavior of the drag parameter as a function of the lift coefficient caused by the twist on the wing.

On the following pages, figures 37, 38, 39, and 40 show plots of $C_{D,i}/C_L^2$ as functions of the C_L at various Mach numbers in more detail for cases 1, 2, 3, and 4 respectively. Comparison of figures 37 and 38 show that even at the highest Mach number of 0.9 for case 2, the induced drag parameter is still lower than that of case 1 at the lowest Mach number of 0.1. Furthermore, once again the non-linear behavior of $C_{D,i}/C_L^2$ in cases 1 and 2 due to twist and the contrasting constant linear behavior of $C_{D,i}/C_L^2$ in cases 3 and 4 is seen at various Mach numbers. The fact that case 4 has the lowest drag parameter at corresponding Mach numbers with respect to the other cases is also seen.

Figure 37. The Induced Drag Parameter ($C_{D,i}/C_L^2$) based on the near filed solution vs. C_L for case 1 at various Mach numbers.

Figure 38. The Induced Drag Parameter ($C_{D,i}/C_L^2$) based on the near filed solution vs. C_L for case 2 at various Mach numbers.

Figure 39. The Induced Drag Parameter ($C_{D,i}/C_L^2$) based on the near filed solution vs. C_L for case 3 at various Mach numbers.

Figure 40. The Induced Drag Parameter ($C_{D,i}/C_L^2$) based on the near filed solution vs. C_L for case 4 at various Mach numbers.

Figure 41 below shows a the plot of the total induced drag parameter based on the near-field solution as a function of Mach number for all cases. This plot must be used with caution, since it only shows the total induced drag parameter. The reason being that the following figure indicates that for cases 1 and 3 have the same total induced drag parameter as a function of Mach number. Although this is the case as a function of Mach number, it was seen in the previous pages that as a function of C_L the induced drag parameter was not the same in cases 1 and 3 due to that fact that the values of C_L was different for each case. Therefore, figure 41 should only be used for the actual limited comparison that it allows. With this in mind, figure 41 shows that cases 2 and 4 have a

Figure 41. The Total Induced Drag Parameter ($C_{D,i}/C_L^2$) based on the near filed solution vs. the Mach number for all cases.

lower total induced drag parameter than cases 1 and 3. This leads to the conclusion that it is the dihedral angle which causes the increase in the drag parameter, since of the four cases, it is cases 1 and 3 that have a dihedral angle configuration. Furthermore, figure 41 shows that the total induced drag parameter is the same for cases 1 and 3 and for cases 2 and 4. Since case 1 has both dihedral and twist and case 3 has only dihedral, it is clear that the twist has no effect on the induced drag parameter. The similar result applies to cases 2 and 4, where case 2 has twist only and case 4 has no twist. Thus, only wing dihedral has an adverse effect on the total induced drag parameter.

With the above results in mind, and revisiting the results of figure 36 and figures 37-40, it is quite clear why the plots of the induced drag parameter were different as a function of the lift coefficient due to the effects of twist. Even though twist plays no role on the total induced drag parameter, it still plays a role on the lift coefficient, which was seen in section 2-7. Therefore, when the induced drag parameter is plotted against the lift coefficient, dissimilarities among the plots of the different cases becomes evident, since the values of the lift coefficient are different for each case and are affected by the wing twist configuration.

Hence, even though figure 41 indicates that cases 2 and 4 have similar total induced drag parameter performance as a function of Mach number, case 4 is still better than case 2. The reason being that when the characteristics of both the lift coefficient and the associated induced drag parameter for cases 2 and 4 are compared, case 4 has the best performance.

2-10 The Thrust and Suction Coefficients:

Figure 42 below shows the total leading edge thrust and suction coefficients at 1 radian angle of attack as a function of Mach number for all four cases. Considering the fact that the values for C_T for cases 1 and 3 fall on the same line, it is easily concluded that the wing twist configuration used has no effect on the C_T , since case 1 has both twist and dihedral and case 3 only has dihedral. A similar conclusion is reached through examination of the curve for cases 2 and 4, where case 2 has twist and case 4 has no twist. Similar comparison of the C_S curves for cases 1 and 3 and cases 2 and 4, yields the conclusion that twist has no effect on C_S either.

Figure 42. Total Leading-Edge Thrust and Suction Coefficients at 1 radian angle of attack vs. Mach number.

Comparison of the C_T curve for cases 1 and 3 with that of cases 2 and 4 indicates that a higher C_T is achieved by wings with no dihedral (cases 2 and 4). Furthermore, comparison of the C_s curve for cases 1 and 3 with that of cases 2 and 4 indicates that a higher C_s is achieved by wings with no dihedral (cases 2 and 4). Thus, dihedral has a detrimental effect on both the leading edge thrust and suction coefficients, and wings without dihedral (cases 2 and 4) have efficient values of C_T and C_s over that of wings with dihedral (cases 1 and 3). Furthermore both C_T and C_s increase from the root towards the wing tip section.

On the following pages 75-76, the spanwise distribution of the nondimensional section leading edge thrust and suction coefficients for all cases at various Mach numbers is shown. Examination of figure 43 for the nondimensional section leading-edge thrust coefficient for cases 1 and 3 shows a spike associated with the second negative dihedral angle. Comparison of figures 43 and 44 indicates that for wings without dihedral (cases 2 and 4), an overall higher and smoother distribution of section C_T exists than that for wings with dihedral (cases 1 and 3). Examination of figure 45 for the nondimensional section leading-edge suction coefficient for cases 1 and 3 once again shows a spike associated with the second negative dihedral angle. Comparison of figures 45 and 46, indicates that for wings without dihedral (cases 2 and 4), an overall higher and smoother distribution of section C_s exists than that for wings with dihedral (cases 1 and 3). Further comparison of all figures indicates that wings with dihedral (figures 43 and 45) have a higher section C_T and C_s distribution at the root section than wings without dihedral (figures 44 and 46). Thus, if the dihedral angle configuration is manipulated, it is quite possible that a better C_T and C_s distribution can be achieved than that possible with wings without dihedral.

Figure 43. Variation of the section leading-edge thrust coefficient along the span for cases 1 and 3 at various Mach numbers.

Figure 44. Variation of the section leading-edge thrust coefficient along the span for cases 2 and 4 at various Mach numbers.

Figure 45. Variation of the section leading-edge suction coefficient along the span for cases 1 and 3 at various Mach numbers.

Figure 46. Variation of the section leading-edge suction coefficient along the span for cases 2 and 4 at various Mach numbers.

2-11 The Pitching Moment Coefficient:

Figure 47 below, shows the plot of the pitching moment coefficient as a function of Mach number at $C_L=0$. This is also known as the quarter chord moment coefficient C_{mo} . Examination of the figure indicates that there is no C_{mo} for cases 2 and 4, which is quite expected, since there is no wing twist and thus no camber associated with these wings. In addition, since the C_{mo} for cases 3 and 4 lie on the same curve, it is evident that the wing dihedral has no effect on C_{mo} . Furthermore, the highest C_{mo} values are achieved by case 1, where the wing has both dihedral and twist, and the second highest values are for case 2 where the wing has twist only. Thus, although the dihedral plays no role on the value of C_{mo} for wings with no twist, it does increase the effects of twist on C_{mo} when twist is present. Thus, from the pitching standpoint, cases 3 and 4 are the most stable. In contrast, case 1 is the least stable, which is desirable in a fighter for better maneuverability.

Figure 47. The Pitching Moment Coefficient as a function of Mach number for all four cases at $C_L=0.0$.

2-12 Stability Parameters:

The longitudinal stability parameter ($\partial C_m / \partial C_L$) as a function of Mach number for all four cases is plotted on figure 48 below. From the comparison of the two curves for cases 1 and 3 and that of cases 2 and 4, it is clearly seen that the latter has a lower (i.e. more negative) value of longitudinal stability parameter. Thus, in cases 2 and 4 the value of C_m drops more rapidly than in cases 1 and 3, as the value of C_L is increased. Henceforth, cases 2 and 4 are more stable, and require fewer measures to counter act the effects of C_m as the Mach number or the lift coefficient is increased. This would be beneficial in reducing the size and surface area and thus the drag of any tail wings or canards which may be present for stability purposes. On the other hand, cases 1 and 3 are less stable, which is desirable in a fighter due to better maneuverability requirements.

Figure 48. The Longitudinal Stability Parameter ($\partial C_m / \partial C_L$) as a function of Mach number for all four cases.

Figure 49. Damping-In-Pitch Parameter (C_{mq}) and the Lift Coefficient due to pitch rate (C_{Lq}) as a function of Mach number for all four cases.

Figure 49 above, shows the two stability parameters of C_{mq} and C_{Lq} as a function of Mach number. From the comparison of the curves for the lift coefficient due to pitch rate and the damping-in-pitch parameter, it is seen that cases 2 and 4 have higher values of C_{mq} (C_{mq} is always negative and hence a more negative value is considered higher) and C_{Lq} than cases 1 and 3. Due to the higher damping, a higher C_{mq} and C_{Lq} will oppose the motion of the vehicle more than a lower C_{mq} and C_{Lq} as the pitch rate is increased. This will have a tendency to keep the aircraft on a more stable straight and level flight trajectory. Furthermore, since as the pitch rate is increased and the vehicle begins to oppose the pitching motion more, the likelihood of accidentally putting the vehicle at a stall angle of attack through minor control stick movement decreases. Furthermore, the

higher damping values also lessen the likelihood of the aircraft flipping over as the pitch rate is increased more rapidly, since the damping values will opposed the vehicle motion more rapidly as well. This stability behavior in cases 2 and 4 is preferred in passenger aircraft. On fighters, on the other hand, it is desired for the aircraft to roll over and pitch up or down easily with minor control stick movement. Thus, for better maneuverability, cases 1 and 3, where wing dihedral is present, are more desirable.

Figure 50 below, shows a plot of the damping-in-roll parameter as a function of the Mach number for all cases. Once again, the values for cases 2 and 4 are higher than that for cases 1 and 3 (C_{Lp} is always negative, thus a more negative value is considered higher).

Figure 50. Damping-In-Roll Parameter (C_{Lp}) as a function of Mach number for all four cases.

Since the higher C_{Lp} value will oppose the rolling motion of the aircraft more, the aircraft will be more stable and will not roll as easily due to stick movement or the action of wing gusts and sideslip rolls. Thus, once again cases 2 and 4, where the wings have no dihedral, would be desirable for passenger aircraft due to better stability. In contrast, cases 1 and 3, where the wings have a dihedral configuration, would be preferred for fighter aircraft due to better maneuverability. Finally, it should also be noted, that examination of figures 48-50 also shows that wing twist has no effect on $\partial C_m / \partial C_L$, C_{mq} , C_{Lq} , and C_{Lp} , but rather only dihedral has an effect.

3. CONCLUSION

This thesis successfully fixed the errors in the VLM3D FORTRAN program by inserting the proper format statements for the input/output data and thus bringing the program to its original correct configuration. To validate the correct operation of the code, some test cases which the VLM3D authors had studied were run and the results were compared. Comparison of the results indicated that the program was in perfect working order. Furthermore, in order to ease the generation of the input files due to the strict formatting requirements, a template was generated, which is attached in Appendix A-2.

Upon perfect operation of the VLM3D code, four cases of wings were studied with the same overall dimensions. Case 1 was a wing with both a dihedral and twist configuration. Case 2 had no dihedral configuration, but had the same twist configuration. Case 3 had the same dihedral configuration used in case 1, but no twist configuration was used. Case 4 was a flat delta wing with the same dimensions as cases 1-3, but had no dihedral or twist configuration. Upon the collection of the output data for all of the above four cases and a thorough understanding of the lifting properties of delta wings, an examination and comparison of the following aerodynamic performance characteristics was performed:

- the ΔC_p distribution on the wing surface
- spanwise distribution of the combination of the basic span load and additional load
- the lift curve at various Mach numbers
- the lift coefficient due to twist @ $\alpha=0$
- the lift curve slope per degree vs. Mach number

- angle of attack for zero lift vs. the Mach number
- the induced drag coefficient vs. C_L at various Mach numbers
- spanwise distribution of the nondimensional section induced drag coefficient
- the induced drag parameter vs. C_L at various Mach numbers
- total leading edge thrust and suction coefficients at $\alpha=1$ rad. vs. Mach number
- spanwise distribution of the section leading edge thrust coefficient at various Mach numbers
- spanwise distribution of the section leading edge suction coefficient at various Mach numbers
- the pitching moment coefficient at $C_L=0$ vs. Mach number
- the longitudinal stability parameter ($\partial C_m / \partial C_L$) vs. Mach number
- the damping-in-pitch ($C_{m\dot{\alpha}}$) stability parameter vs. Mach number
- the lift coefficient due to pitch rate ($C_{L\dot{\alpha}}$) vs. Mach number
- the damping-roll-parameter ($C_{l\dot{p}}$) vs. Mach number

The comparison of the above results, showed that both the dihedral and twist configuration used in cases 1-3 had an adverse effect on all of the aerodynamic performance results. Furthermore, the results showed that the wing dihedral configuration used had a much higher adverse effect than the twist configuration. In addition, on certain parameters where the dihedral alone appeared to have no effect and the only twist configuration had an adverse effect, the combination of the wing dihedral and twist configuration accentuated the adverse of the twist configuration. This was seen in the results for the induced drag parameter vs. C_L and the pitching moment coefficient at $C_L=0$ vs. the Mach number. Thus, in all of the aerodynamic

performance characteristics studied, the wing without any dihedral or twist configuration (case 4) proved to have the best characteristics.

From the stability standpoint, for application in fighter aircraft, wings with a dihedral configuration (cases 1 and 3) proved to have the best characteristics. These cases were the least stable, and thus more maneuverable, which is desirable in fighter aircraft. Thus, for application in fighter aircraft, the wing with only a dihedral configuration (case 3) is preferred, since it has the best maneuverability performance, and better aerodynamic performance than the wing with both twist and dihedral (case 1). For application in passenger aircraft, the wing with no dihedral or twist (case 4) is preferred, since it has the best aerodynamic performance and the most stable stability parameters.

Although the results of this thesis showed that both the wing dihedral and twist configuration used had a negative impact on all of the aerodynamic performance parameters, it does not necessarily mean that this is the case for all dihedral and twist configurations used on delta wings. Thus, the adverse effects of the wing dihedral and twist on the aerodynamic performance parameters of delta wings contained in the results of this thesis should be used with great caution, since they only apply to the wing twist, dihedral, and shape configuration used in this thesis. It certainly may be the case that a different wing twist and dihedral configuration will yield a better aerodynamic performance than a wing without any dihedral or twist, which warrants further study in the future. One recommendation for future study would be to perform smoke tunnel tests in order to better determine the behavior of the primary and secondary vortex cores at the locations of the wing dihedral angles, since results indicated that the most adverse behavior in the aerodynamic parameters occurred at these wing locations.

APPENDICES

A-1 VLM3D COMPUTER PROGRAM:

Given the sets of equations derived in Part I, a FORTRAN computer program known as VLM3D (Three Dimensional Vortex Lattice Method) was developed for estimating the subsonic aerodynamic characteristics of complex planforms. This section, describes the basic operating principles of the VLM3D computer program and some of its limitations. Furthermore, it also discusses the way the program carries out the solutions of the equations described in Part I, and the manner in which the input and the output is carried out.

A-1.1 Basic Concepts and Limitations:

The vortex-lattice method described in Part I is used to determine the aerodynamic characteristics of planforms at subsonic speeds. The method assumes steady, irrotational, inviscid, incompressible, attached flow. The effects of compressibility are represented by application of the Prandtl-Glauert similarity rule to modify the planform geometry. Potential flow theory in the form of the Biot-Savart law is used to represent disturbances created in the flow field by the lift distribution of the planform. It is assumed that in any plane parallel to the $\hat{X} - \hat{Z}$ plane, the vertical displacements which occur in the wing or wake are neglected, except when the boundary conditions at the control points are determined.

The lifting-surface planform is represented for the computer program by a series of up to 24 straight segments which are positioned counterclockwise around the perimeter of the left half

of the planform. Lateral symmetry is presumed. The lines start at the leading edge of the plane of symmetry, go along the leading edge to the left tip of the planform, return along the trailing edge, and end at the trailing edge of the plane of symmetry. The preciseness of the $\bar{x}$ and $\bar{y}$ Cartesian coordinates and dihedral angles, given as input data, determines the accuracy of the planform representation.

Some of the limitations of the program are as follows:

1. A maximum of two planforms may be specified such as, wing-tail and canard-wing combinations.
2. A maximum of 24 straight-line segments may be used to define the left half of a planform.
The lateral separation of the ends of these line can be critical when the horseshoe vortices are laid out by the computer program.
3. The maximum number of horseshoe vortices on the left side of the configuration plane of symmetry is 120. When two planforms are specified, the sum total of the vortices in both is limited to 120. Within this limit, the number of horseshoe vortices in any chordwise row may vary from 1 to 20 and the number of chordwise rows may vary from 1 to 50.
4. For planforms with dihedral or two planforms which do not lie in the same plane, the variation in local chord must be continuous from the tip chord to the root chord of each planform specified. Furthermore, the number of horseshoe vortices on each chordwise row must at least be two, and the number of horseshoe vortices must be constant over the semispan of each planform.
5. The program does not include source distribution, and is thus incapable of including the effects of thickness. Therefore, only thin wings can be studied.

A-1.2 Program Input:

Upon execution, the program prompts for an input file which contains the input data. This file contains two sets of data which are classified as 'Group One' and 'Group Two' data. All coordinates and slopes should be given for the left half of the wing planform. The axis system used is given in figures 3 and 4. The $\bar{y} = 0$ intercept coincides with the root chord and is positive pointing along the right wing. Although the $\bar{x} = 0$ intercept usually coincides with the intersection of the leading edge at the root chord, it may lie anywhere along the root chord. $\bar{X}$ is positive pointing into the wind.

It must be remembered that the program input is formatted. Therefore, the strict formatting of the input data must always be followed, otherwise incorrect data will be read.

A-1.2.1 Group One Input Data:

The group one data contain the total number of planforms, number of sets of group two data, some reference data, and the description for the line segments that define the perimeter of the planforms. All of the cards use a format of (8F10.6) for group one data.

The first line contains the following data in the order listed:

PLAN	Number of planforms for the configuration; use 1 for wing only and 2 for wing-tail or canard-wing combinations.
TOTAL	Number of sets of group two data specified for the configuration, which is the number of different calculations to be made for the same

wing at different Mach numbers, C_l values, etc.

CREF Reference chord of the configuration

This chord is used only to nondimensionalize the pitching-moment terms and must be greater than zero.

SREF Reference area of the configuration

This area is used only to nondimensionalize the computed output data such as lift and pitching moment and must be greater than zero.

The next set of variables contain the data to define each of the planforms. The initial line in the set contains the following data in the order listed:

AAN (IT) Number of line segments used to define left half of a wing planform
(does not include plane of symmetry) A maximum of 24 line segments may be used.

XS (IT) x location of the pivot; use 0 for a fixed wing

YS (IT) y location of the pivot; use 0 for a fixed wing.

RTCDHT (IT) Vertical distance of particular planform being read in with respect to
the wing root chord height; use 0 for a wing

The rest of the lines in the set for the given planform contain the data for each line segment used to define the basic planform. There is one line of data for each of the line segments defined with AAN (IT), and the data is listed in the following order:

XREG (I, IT) x location of the i^{th} breakpoint

The first breakpoint is located at the intersection of the left wing

leading edge with the root chord. They are numbered in increasing order for each intersection of lines in a counterclockwise direction.

YREG (I, IT) y location of the i^{th} breakpoint

DIH (I, IT) Dihedral angle (deg) in $\bar{Y} - \bar{Z}$ plane of line from breakpoint i to $i+1$.
Positive upward. Along a streamwise line, the dihedral angle is not defined, so use 0.

AMCD The move code.
This number indicates whether the line segment i is on the movable panel of a variable-sweep wing. Use 1 for a line which is fixed and 2 for a line which is movable.

A-1.2.2 Group Two Input Data:

Group two data consist of three sets of information. The first line of the group two data contains the first set of data which describes the details of the particular configuration for which the loading is desired. It has a format of (8F5.1, F10.4, F5.1, F10.4) and the data are in the order listed:

CONFIG An arbitrary configuration number which may include up to four digits.

SCW The number of chordwise horseshoe vortices to be used to represent the wing; a maximum value of 20 may be used. If set to 0, then a table of the number of chordwise horseshoe vortices from tip to root must be provided as TBLSCW (I). This SCW=0 option can be used only

	on wings without dihedral and for coplanar wing-tail combinations.
VIC	<p>The nominal number of spanwise rows at which chordwise horseshoe vortices will be located. The variable VIC must not cause more than 50 spanwise rows to be used by the program to describe the wing. In addition, the product of SSW and SCW cannot exceed 120. If SCW is 0, then the sum of the values in TBLSCW (I) cannot exceed 120.</p>
MACH	<p>Mach number</p> <p>Use a value other than 0 only if the Prandtl-Glauert compressibility correction factor $\beta = \sqrt{1 - M_\infty^2}$ is to be applied. It should be less than the critical Mach number.</p>
CLDES	<p>Desired lift coefficient</p> <p>The number specified here is used to obtain the span load distribution at a particular lift coefficient. If this answer is not desired, use 1. If a drag polar for C_L values from -0.1 to 1 is desired, use 11.</p>
PTEST	<p>C_{lp} indicator. Use 1 if damping-in-roll parameter is desired. In this case, except for the incremental pressure coefficients and C_{lp}, all other aerodynamic data will be omitted. Use 0 if C_{lp} is not desired.</p>
QTEST	<p>C_{Lq} and C_{mq} indicator; use 1 if the two quantities are desired. In this case, except for the incremental pressure coefficients, C_{Lq}, and C_{mq}, all other aerodynamic data will be omitted. Use 0 if these data are not desired.</p>

Both PTEST and QTEST cannot be set to 1 for the same run.

- TWIST (1) Twist code for the first planform.
- Use 0 for no twist and/or camber, and 1 when the planform had twist and/or camber and provide the third set of data for group two.
- SA (1) Variable-sweep angle for the first planform
- Specifies the leading-edge sweep angle (deg) for the first movable line adjacent to the fixed portion of the planform. For a fixed planform, this quantity may be omitted.
- TWIST (2) Twist code for the second planform.
- SA (2) Variable-sweep angle for the second planform.

The next line of the group two data contains the second set of data if SCW is set to 0. Data for the first variable are in the first card and for the second variable are in the second and following cards. The data supplied are in an (8F10.4) format as follows:

- STA Total number of spanwise rows of horseshoe vortices per semispan.
- This variable sets the number of values of TBLSCW (I) to be read in.
- TBLSCW (I) Number of horseshoe vortices in each row starting at the row near the tip of the first planform and proceeding to the row near the root. If a second planform has been specified, the table of chordwise rows concludes with number of horseshoe vortices in each row of the second planform.

The third set of data is for planforms with twist and/or camber. If the configuration has no twist and/or camber, the local angles of attack are not specified since the program will set them

equal to 0. If the configuration consists of two planforms, local angles of attack may be specified for both or only one of the two planforms. The twist code describes the input to the computer in an (8F10.4) format as follows:

ALP (NV) Local angles of attack in radians.

These are the values at the control point for each horseshoe vortex on the wing when the root-chord angle of attack is 0° . These data will usually require several cards. For the first value on the first card, use the local angle of attack for the horseshoe vortex nearest the first planform leading edge at the tip; for the second value, use the angle of attack for the horseshoe vortex immediately behind in a chordwise direction. Continue with the rest of the chordwise row of horseshoe vortices at the tip; then continue inboard at the next chordwise row in the same manner to the root until local angles of attack for all the control points have been specified.

A-1.3 Horseshoe Vortex Lattice Layout:

After the program reads in the input data, the planform is divided chordwise and spanwise along the surface into trapezoidally shaped elemental panels, and one horseshoe vortex is assigned to represent each panel. This discretization is shown in figure 3. The horseshoe vortex is composed of three vortex lines: a bound vortex which is swept to coincide with the elemental-panel quarter-chord sweep angle in the plane of the wing and two trailing vortices

which extend chordwise parallel to the free stream to infinity behind the wing. An illustration of this is given in figure 4. The nominal width of these horseshoe vortices is the total semispan in the plane of the wing divided by the variable VIC of the input.

The procedure for laying out the horseshoe vortices and the elemental panels is to begin at the left tip with the first chordwise row of vortices and then proceed toward the wing root. The actual spanwise locations of the chordwise rows of horseshoe vortices are adjusted so that there is always a trailing vortex filament at points where there are intersections of lines with breakpoints of the planform. This adjustment may cause the horseshoe vortex width to be narrower or wider than the nominal width. According to Margason and Lamar (ref. 1), when a horseshoe vortex has one trailing vortex filament which coincides with a breakpoint, the width of the horseshoe vortex may vary from 0.5 to 1.5 times the nominal width. Furthermore, they also discovered that when both trailing legs coincide with breakpoints, the width may vary from a maximum of 1.5 times the nominal width to a minimum width of $(b/2)/2000$. For wings with zero dihedral angles, good results can be expected for horseshoe vortices of these widths. However, for planforms having dihedral, great care must be taken not to describe a planform with these narrow horseshoe vortices, since span loading results may be poor when the narrow horseshoe vortices are used.

In the chordwise direction, the horseshoe vortices are distributed uniformly and the number of vortices is given either by the variable SCW or TBLSCW (I) of the input file. Due to the fact that the number of horseshoe vortices in the chordwise and spanwise directions is limited to 20 and 50 respectively and the total to 120, the exact number generated by the program depends on the values of VIC and SCW and on the details of the planform. The program may

generate as many as one additional row of chordwise horseshoe vortices at each breakpoint outboard of the root. Wings with dihedral must always have at least two horseshoe vortices chordwise, whereas wings without dihedral may have only one.

A-1.4 Aerodynamic Computation:

Due to the variation of sidewash velocity and the local value of vortex strength, the lift developed along the chordwise bound vortices in a chordwise row of horseshoe vortices varies from the leading edge to the trailing edge of the wing. Examination of fig. 4, indicates that there is no circulation along the chordwise bound vortex from the leading edge of the wing to the quarter-chord of the first elemental panel. Therefore, no lift is generated there. On the chordwise bound vortex from the quarter-chord of the first elemental panel to the quarter-chord of the second elemental panel, there is a constant value of circulation and a varying value of sidewash velocity. A special case occurs for the first elemental panel at the left wing tip, where the value of circulation just equals that of the first elemental panel of the first chordwise row of horseshoe vortices. Inboard from the tip, this chordwise bound vortex lies between two chord-wise rows of horseshoe vortices, and its circulation is equal to the difference between the circulations of the first elemental panel of each row. The sidewash velocity used is the one computed at the three-quarter-chord on the left chordwise bound vortex of the first elemental panel.

The next lift to be computed is that developed along the chordwise bound vortex between the quarter-chord of the second elemental panel and the quarter-chord of the third elemental

panel. This lift is computed in a manner similar to that of the first horseshoe vortex with slight differences which are explained now. At the left wing tip, the sum of the circulation values of the first two elemental panels is used. Inboard from the tip between two chordwise rows of horseshoe vortices, the circulation is equal to the sum of the difference between the circulations of the first elemental panel of each row and the difference between the circulations of the second elemental panel of each row. The sidewash velocity used is the one computed at the three-quarter-chord on the left chordwise bound vortex of the second elemental panel, and is given by eq. (25).

This procedure continues through the last elemental panel in a chordwise row. However, the final chordwise bound vortex extends from the quarter-chord of the last elemental panel to the trailing edge of the wing so that its length is equal to only three-quarters of the length of the other chordwise bound vortices in the same chordwise row of horseshoe vortices.

A-1.5 Output Data:

After the computer program runs through the aerodynamic computation and calculates the required vortex strength and velocities, it then writes the output data into the specified output file using the equations derived in Part 1. The output data are listed in two sections: geometry data and aerodynamic data.

A-1.5.1 Geometry Output Data:

The following is a listing of the data presented for each of the elemental panels:

X C/4	x location of quarter-chord at horseshoe vortex midspan
X 3C/4	x location of three-quarter-chord at horseshoe vortex midspan
	This is the x location of the control point.
Y	y location of horseshoe vortex midspan
Z	z location of horseshoe vortex midspan
S	Semiwidth of horseshoe vortex
C/4 SWEEP ANGLE	Sweep angle of quarter-chord
DIHEDRAL ANGLE	Dihedral angle of the elemental panel
LOCAL ALPHA IN RADIANS	Local angle of attack at control point in radian (X 3C/4, Y, Z)
DELTA CP AT DESIRE CL=	ΔC_p for each elemental panel when wing lift is $C_{L,d}$

REF. CHORD	Reference chord of wing
C AVERAGE	Average chord (true planform area/true span)
TRUE AREA	True area computed from planform area listed
REF. AREA	Reference area of the wing
B/2	True semispan of planform listed in second group of geometry data
REF. AR	Reference aspect ratio computed from reference planform area and true span
TRUE AR	True aspect ratio computed from true planform area and true span
MACH NUMBER	Mach number

A-1.5.2 Aerodynamic Output Data:

The aerodynamic data is listed in the order printed out by the program.

DESIRED CL	Desired lift coefficient specified in input data for complete configuration
COMPUTED ALPHA $C_{L,d}/C_{L,\alpha}$	angle of attack where desired lift coefficient is developed
CL (WB)	That portion of desired lift coefficient developed by the planform with the maximum span when two planforms are specified. When one planform is specified, this is the desired

	lift coefficient.
CDI AT CL (WB)	Induced drag coefficient for lift coefficient in previous item When two planforms are specified, this is the induced drag coefficient of only the planform with the maximum span. This result is based on the far-field solution.
CDI/(CL(WB)**2)	Induced drag parameter for an elliptic load distribution based on reference aspect ratio.
CL ALPHA	Lift-curve slope per radian and degree
CL(TWIST)	Lift coefficient due to twist and/or camber at zero angle of attack.
ALPHA AT CL=0	Angle of attack at zero lift in degrees. This is nonzero when the wing has twist and/or camber
Y CP	Spanwise distance in fraction of semispan from root chord to center of pressure on left wing panel
CM/CL	Longitudinal stability parameter based on a moment center about $\hat{Y}$ -axis.
2Y/B	For each chordwise row of horseshoe vortices, location of midpoint of each chordwise row of horseshoe vortices in fraction of semispan location are listed sequentially from near the left wing tip towards the root.

The following data are presented for a lift coefficient of 1:

SL COEF	Span-load coefficient, $c_{lc}/C_L c_{av}$
CL RATIO	Ratio of local lift to total lift, c_l/C_L
C RATIO	Ratio of local chord to average chord, c/c_{av}
LOAD DUE TO TWIST	Distribution of span-load coefficient due to twist and camber at 0° angle of attack.
ADD. LOAD AT CL=	Distribution of additional span-load coefficient required to produce zero lift when combined with lift due to twist and camber. This distribution is computed at $C_{L,lc}$
BASIC LOAD AT CL=0	Basic span-load-coefficient distribution at zero lift coefficient. These are the sum of the previous two columns of data.
SPAN LOAD AT DESIRED CL	Distribution of combination of basic span load and additional span-load coefficients at desired C_L .
SL COEF FROM CHORD BD VOR	Portion of span-load coefficient due to lift along chordwise bound vortices averaged at horseshoe vortex midspan.

The following data are presented for each chordwise row of horseshoe vortices:

L.E. SWEEP ANGLE	Leading-edge sweep angle in degrees
CDII C/2B	Nondimensional section induced-drag-coefficient term

CT C/2B	Nondimensional section leading-edge thrust-coefficient term
CS C/2B	Nondimensional section leading-edge suction-coefficient term
CDII	Contribution to total drag coefficient from each spanwise row of horseshoe vortices, $c_{d,ii}(2s \cos \phi)/(q_\infty S_{ref})$
CT	Contribution to total leading-edge thrust coefficient from each spanwise row of horseshoe vortices, $c_t(2s \cos \phi)/(q_\infty S_{ref})$
CS	Contribution to total leading-edge thrust coefficient from each spanwise row of horseshoe vortices, $c_s(2s \cos \phi)/(q_\infty S_{ref})$

Finally, the total coefficient values are listed.

CDII/CL**2	Induced-drag parameter computed from near-field solution
CT	Leading-edge thrust coefficient computed at 1 rad angle of attack
CS	Leading-edge suction coefficient computed at 1 rad angle of attack

For the case where PTEST is 1, all of the foregoing aerodynamic output data are omitted, and only CLP is printed.

Likewise, for the case where QTEST is 1, all of the foregoing aerodynamic output data are omitted, and only CMQ and CLQ are printed.

A-1.6 Effect of the Vortex Lattice Arrangement on Computed Results:

Margason and Lamar (ref. 1) carried out many investigation to determine the effect of vortex-lattice arrangement on the computed aerodynamic characteristics. Margason and Lamar (ref. 1) discovered that for planforms with two prescribed leading-edge sweep angles in combination with three different taper ratios for aspect ratios of 2, 4.5, and 7, different vortex-lattice arrangements has a smaller impact on y_{cp} and $C_{D,i}/C_L^2$ than on $C_{L\alpha}$, $\partial C_m/\partial C_L$, and $C_{D,i}/C_L^2$. Furthermore, increasing the number chordwise rows of elemental panels on a wing semispan results in converging results for all numbers of elemental panels in each chordwise row.

For unswept planforms with a leading-edge sweep angle of 45° and taper ratios of 1.0, 0.5, and 0, Margason and Lamar (ref. 1) discovered that a spanwise increase in the number of chordwise rows of horseshoe vortices leads to converging answers. For these planforms, the number of chordwise rows of elemental panels on a wing semispan should be greater than 20 for convergence of $C_{L\alpha}$ to a particular value which is sufficient for convergence of $\partial C_m/\partial C_L$ and $C_{D,i}/C_L^2$. Furthermore, the computed values of $C_{L\alpha}$, $\partial C_m/\partial C_L$, and $C_{D,i}/C_L^2$ in most instances have a definite dependence upon the number of elemental panels in a given chordwise row. In particular, the number of elemental panels in a chordwise row controls the asymptotic levels that these aerodynamic characteristics attain with varying number of chordwise rows of elemental panels in a wing semispan. These asymptotic levels approach a converged result when the number of elemental panels in a chordwise row is increased. As the number of elemental panels in a chordwise row ($\bar{N}_c$) is increased, differences between asymptotic levels

which occur for consecutive $\bar{N}_c$ decrease, and the largest difference in asymptotic levels is by increasing $\bar{N}_c$ from 1 to 2. Therefore, an $\bar{N}_c$ value of 2 should be that minimum used, and higher values of $\bar{N}_c$ have little effect on $C_{L\alpha}$ and C_{Di}/C_L^2 , and provide additional improvement in $\partial C_m/\partial C_L$ and C_{Di}/C_L^2 .

From additional studies, Margason and Lamar (ref. 1) discovered that values of 10 for the number of elemental panels in a chordwise row and 12 for the number of chordwise rows of elemental panels on a wing semispan provide reasonable results for delta wings. The large number of chordwise stations is necessary on such wings so that the effect of the induced camber loading can be properly taken into account. As it was discussed earlier in Part 1.5 on page 26, correct thrust coefficient can be obtained from the far-field induced drag and lift-curve slope directly, only by finding the appropriate combination of the number of chordwise rows of elemental panels on a wing semispan and the number of elemental panels in each chordwise row will the induced-drag result be the same for both methods. This check provides a method by which the correct distribution of section thrust can be obtained.

Studies on a cropped double-delta planform having an inboard leading-edge sweep angle of 83° , an outboard leading-edge sweep angle of 62° , and an aspect ratio of 1.49 by Margason and Lamar (ref. 1), showed to trends for C_{lp} , C_{mq} , and C_{Lq} . A large value of chordwise rows of elemental panels on a wing semispan with at least two horseshoe vortices in each row are required for estimating C_{lp} . Furthermore, 8 or more elemental panels in a chordwise row and a nominal value of 8 or 10 chordwise rows of elemental panels on a wing semispan are required for estimating C_{mq} and C_{Lq} .

A-2. Input File Template for VLM3D:

Note: Red cells are locations of the decimal for each of the input data variables.

[illegible]

-.021	0.0	0.0	0.0	-.042	-.035	-.035	-.035	-.006
-.006	-.006	0.0	0.0	0.0	0.0	0.0	-.043	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
-.041	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.0	0.0	-.004	0.0	0.0	0.0	0.0	0.0	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7. 10. 10.5 .8	11. 0.	0. 1.	0. 1.	0. 1.	0. 1.	0. 1.	0. 1.	0. 1.
-.03	-.03	-.03	-.03	-.03	-.03	-.03	-.03	-.03
-.03	-.036	-.036	-.036	-.036	-.036	-.036	-.036	-.036
-.036	-.036	-.036	-.036	-.038	-.038	-.038	-.038	-.038
-.038	-.038	-.038	-.038	-.038	-.038	-.038	-.038	-.038
-.039	-.039	-.039	-.039	-.039	-.039	-.039	-.039	-.039
-.050	-.050	-.050	-.041	-.041	-.041	-.041	-.041	-.020
-.020	-.020	-.048	-.048	-.048	-.021	-.021	-.021	-.021
-.021	0.0	0.0	0.0	-.042	-.035	-.035	-.035	-.006
-.006	-.006	-.006	0.0	0.0	0.0	0.0	-.043	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
-.041	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.0	0.0	-.004	0.0	0.0	0.0	0.0	0.0	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8. 10. 10.5 .9	11. 0.	0. 1.	0. 1.	0. 1.	0. 1.	0. 1.	0. 1.	0. 1.
-.03	-.03	-.03	-.03	-.03	-.03	-.03	-.03	-.03
-.03	-.036	-.036	-.036	-.036	-.036	-.036	-.036	-.036
-.036	-.036	-.036	-.036	-.038	-.038	-.038	-.038	-.038
-.038	-.038	-.038	-.038	-.038	-.038	-.038	-.038	-.038
-.039	-.039	-.039	-.039	-.039	-.039	-.039	-.039	-.039
-.050	-.050	-.050	-.041	-.041	-.041	-.041	-.041	-.020
-.020	-.020	-.048	-.048	-.048	-.021	-.021	-.021	-.021
-.021	0.0	0.0	0.0	-.042	-.035	-.035	-.035	-.006
-.006	-.006	-.006	0.0	0.0	0.0	0.0	-.043	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
-.041	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
0.0	0.0	-.004	0.0	0.0	0.0	0.0	0.0	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

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A-5. Case 3 Input File for Aerodynamic Computation:

1.	8.	78.53166	6297.15
8.	0.	0.	0.
0.	0.	0.	1.
-30.72	-12.32	5.	1.
-73.85	-31.84	-25.	1.
-119.313	-50.56	0.	1.
-121.5	-50.56	-25.	1.
-121.5	-31.84	5.	1.
-121.5	-17.19	5.	1.
-117.7	-12.32	0.	1.
-117.7	0.		
61.	10.	11.	0.
62.	10.	11.	0.
63.	10.	11.	0.
64.	10.	11.	0.
65.	10.	11.	0.
66.	10.	11.	0.
67.	10.	11.	0.
68.	10.	11.	0.

A-6. Case 4 Input File for Aerodynamic Computation:

1.	8.	78.53166	6297.15
8.	0.	0.	0.
0.	0.	0.	1.
-30.72	-12.32	0.	1.
-73.85	-31.84	0.	1.
-119.313	-50.56	0.	1.
-121.5	-50.56	0.	1.
-121.5	-31.84	0.	1.
-121.5	-17.19	0.	1.
-117.7	-12.32	0.	1.
-117.7	0.		
91.	10.	10.5 .1	11.
92.	10.	10.5 .3	11.
93.	10.	10.5 .4	11.
94.	10.	10.5 .5	11.
95.	10.	10.5 .6	11.
96.	10.	10.5 .7	11.
97.	10.	10.5 .8	11.
98.	10.	10.5 .9	11.

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A-7. Stability Parameter Computation Input Files and All Output Files:

The input files for stability parameter computation for cases 1, 2, 3, and 4 have been omitted here for brevity. But they are identical to the aerodynamic computation files, except that the switch for the stability parameter computation is turned on.

The 12 output files have also been omitted for brevity, since each is about 70 pages long.

A-8. VLM3D Program Listing:

```

PROGRAM MLVL                                MAIN 20
CHARACTER*20 CASEFN, OUTFIL
REAL MACH
COMMON/ALL/ BOT,M,BETA,PTEST,QTEST,TBLSCW(50),Q(300),PN(300),
1      PV(300),ALP(300),S(300),PSI(300),PHI(300),ZH(50)    MAIN 30
COMMON/TOTHE/ CIR(300,2),SECTRST(50)                      MAIN 40
COMMON/ONETHRE/TWIST(2),CREF,SREF,CAVE,CLDES,STRUE,AR,ARTRUE,
1      RTCDHT(2),CONFIG,NSSWSV(2),MSV(2),KBOT,PLAN,IPLAN,MACH    MAIN 50
2      ,SSWWA(50)                                           MAIN 60
COMMON/MAINONE/ICODEOF,TOTAL,AAN(2),XS(2),YS(2),KFCTS(2)    MAIN 70
1      ,XREG(25,2),YREG(25,2),AREG(25,2),DIH(25,2),MCD(25,2)    MAIN 80
2      ,XX (25,2),YY (25,2),AS (25,2),TTWD(25,2),MMCD(25,2)    MAIN 90
3      ,AN(2),ZZ (25,2)                                     MAIN 100
7 FORMAT ( //10X,I6,'HORSESHOE VORTICES LAIDOUT, THIS IS MORE THAN
1THE 300 MAXIMUM. THIS CONFIGURATION IS ABORTED.')          MAIN 110
8 FORMAT ( // 10X, I6,' ROWS OF HORSESHOE VORTICES LAIDOUT. THIS
1S MORE THAN THE 50 MAXIMUM. THIS CONFIGURATION IS ABORTED.' )    MAIN 120
9 FORMAT ( // 10X, 'PLANFORM',I6,' HAS',I6,                  MAIN 130
1 ' BREAKPOINTS. THE MAXIMUM DIMENSIONED IS 25. THE CONFIGURATION
2S ABORTED.')                                                 MAIN 140
100 FORMAT (A20)                                              MAIN 150
101 FORMAT (///' START OF A NEW CASE, CASE FILE NAME IS ', A20//)
102 FORMAT ( ' THE OUTPUT FILE NAME IS ', A20//)              MAIN 160
C                                                                MAIN 170
C                                                                MAIN 180
C                                                                MAIN 190
C                                                                MAIN 200
C                                                                MAIN 210
C                                                                MAIN 220
C                                                                MAIN 230
C                                                                MAIN 240
C                                                                MAIN 250
NMAX = 300
ICODEOF= 0
TOTAL = 0                                                    MAIN 260
C                                                                MAIN 270
C                                                                MAIN 280
C                                                                MAIN 290
C                                                                MAIN 300
C                                                                MAIN 310
C                                                                MAIN 320
C                                                                MAIN 330
C                                                                MAIN 340
C                                                                MAIN 350
C                                                                MAIN 360
C                                                                MAIN 370
C                                                                MAIN 380
C                                                                MAIN 390
C                                                                MAIN 400
11 CALL GEOM
IF(ICODEOF.GT.0) GO TO 99
IF(M.GT.NMAX) GO TO 2
NSW = NSSWSV(1) + NSSWSV(2)
IF ( NSW.GT.50 ) GO TO 4
ITSV = 0
DO 10 IT=1,IPLAN
IF ( AN(IT).LE.25. ) GO TO 10
WRITE (29,9) IT,AN(IT)
ITSV = 1
10 CONTINUE
IF (ITSV.GT.0)

```

GO TO 3	MAIN 410
4 WRITE (29,8) NSW	MAIN 420
GO TO 5	MAIN 430
2 WRITE(29,7) M	MAIN 440
GO TO 5	MAIN 450
3 CALL MATX	MAIN 460
CALL AERO	MAIN 470
5 TOTAL=TOTAL-1.	MAIN 480
IF (TOTAL.GT.0.) GO TO 11	
CLOSE (UNIT=28)	
CLOSE (UNIT=29)	
GO TO 1	MAIN 490
99 STOP	MAIN 500
END	MAIN 510
SUBROUTINE INFSUB (BOT, FUI, FVI, FWI)	INFS 10
COMMON/INSUB23/PSII, APHII, XXX, YYY, ZZZ, SNN, TOLRNC	INFS 20
FC =COS(PSII)	INFS 30
FS =SIN(PSII)	INFS 40
FT =FS/FC	INFS 50
FPC=COS(APHII)	INFS 60
FPS=SIN(APHII)	INFS 70
FPT=FPS/FPC	INFS 80
F1 =XXX+SNN*FT*FPC	INFS 90
F2 =YYY+SNN*FPC	INFS 100
F3 =ZZZ+SNN*FPS	INFS 110
F4 =XXX-SNN*FT*FPC	INFS 120
F5 =YYY-SNN*FPC	INFS 130
F6 =ZZZ-SNN*FPS	INFS 140
FFA= (XXX**2+(YYY*FPS)**2+FPC**2*((YYY*FT)**2+(ZZZ/FC)**2-2.*	INFS 150
1XXX*YYY*FT)-2.*ZZZ*FPC*(YYY*FPS+XXX*FT*FPS))	INFS 160
FFB=(F1*F1+F2*F2+F3*F3)**.5	INFS 170
FFC=(F4*F4+F5*F5+F6*F6)**.5	INFS 180
FFD=F5*F5+F6*F6	INFS 190
FFE=F2*F2+F3*F3	INFS 200
FFF=(F1*FPC*FT+F2*FPC+F3*FPS)/FFB - (F4*FPC*FT+F5*FPC+F6*FPS)/	INFS 210
1FFC	INFS 220
C	INFS 230
C	INFS 240
C	INFS 250
C	INFS 260
C	INFS 270
C	INFS 280
C	INFS 290
C	INFS 300
IF(ABS(FFA).LT.(BOT*15.E-5)**2) GO TO 262	INFS 310
FUONE=(ZZZ*FPC-YYY*FPS)*FFF/FFA	INFS 320
FVONE=(XXX*FPS-ZZZ*FT*FPC)*FFF/FFA	INFS 330
FWONE=(YYY*FT-XXX)*FFF/FFA*FPC	INFS 340
GO TO 265	INFS 350
262 FUONE=0	
FVONE=0	INFS 360
FWONE=0.	INFS 370
265 IF(ABS(FFD).LT.TOLRNC) GO TO 263	INFS 380
C	INFS 390
FVTWO= F6*(1.-F4/FFC)/FFD	INFS 400
FWTWO=-F5*(1.-F4/FFC)/FFD	INFS 410
GO TO 266	
263 FVTWO=0	INFS 420
FWTWO=0.	INFS 430
266 IF(ABS(FFE).LT.TOLRNC) GO TO 264	INFS 440
C	INFS 450
FVTHRE=-F3*(1.-F1/FFB)/FFE	INFS 460
FWTHRE=F2*(1.-F1/FFB)/FFE	INFS 470
C	INFS 480
GO TO 267	
264 FVTHRE=0	INFS 490
FWTHRE=0.	

PIT	= 1.5707963	GEOM 620
RAD	= 57.29578	GEOM 630
IF (TOTAL.GT.0.)	GO TO 80	GEOM 640
C		
write(*,*)	'2'	
C		
C		GEOM 650
C		GEOM 660
C		GEOM 670
C	SET PLAN EQUAL TO 1. FOR A WING ALONE COMPUTAION - EVEN FOR A	GEOM 680
C	VARIABLE SWEEP WING	GEOM 690
C	SET PLAN EQUAL TO 2. FOR A WING - TAIL COMBINATION	GEOM 700
C		GEOM 710
C	SET TOTAL EQUAL TO THE NUMBER OF SETS	GEOM 720
C	OF GROUP TWO DATA PROVIDED	GEOM 730
C		
write(*,*)	'3'	
C		
read(28,3)	test,try,tes,tr	
write(*,3)	test,try,tes,tr	
C		GEOM 740
40 READ (28,3,END=1006)	PLAN,TOTAL,CREF,SREF	GEOM 740
WRITE (29,3)	PLAN,TOTAL,CREF,SREF	GEOM 760
IPLAN	=PLAN	
C		
write(*,3)	PLAN,TOTAL,CREF,SREF	
write(*,*)	'4'	
C		GEOM 770
C		GEOM 780
C		GEOM 790
C	SET AAN(IT) EQUAL TO THE MAXIMUM NUMBER OF CURVES REQUIRED TO	GEOM 800
C	DEFINE THE PLANFORM PERIMETER OF THE (IT) PLANFORM.	GEOM 810
C		GEOM 820
C	SET RTCDHT(IT) EQUAL TO THE ROOT CHORD HEIGHT OF THE LIFTING	GEOM 830
C	SURFACE (IT),WHOSE PERIMETER POINTS ARE BEING READ IN, WITH	GEOM 840
C	RESPECT TO THE WING ROOT CHORD HEIGHT	GEOM 850
C		GEOM 860
WRITE (29,1)		GEOM 870
DO 58 IT = 1,IPLAN		GEOM 880
READ (28,3)	AAN(IT),XS(IT),YS(IT),RTCDHT(IT)	GEOM 880
write (29,3)	AAN(IT),XS(IT),YS(IT),RTCDHT(IT)	GEOM 890
N	= AAN(IT)	GEOM 900
N1	= N + 1	GEOM 910
MAK	= 0	GEOM 920
IF (IPLAN.EQ.1)	PRTCON = ' ' FIRST'	GEOM 930
IF (IPLAN.EQ.2 .AND. IT.EQ.1)	PRTCON = ' ' SECOND'	GEOM 940
IF (IPLAN.EQ.2 .AND. IT.EQ.2)	PRTCON = ' ' SECOND'	GEOM 950
WRITE (29,2)	PRTCON,N,RTCDHT(IT),XS(IT),YS(IT)	GEOM 960
WRITE(29,17)		GEOM 970
DO 59 I=1,N1		GEOM 980
READ (28,3)	XREG(I,IT) , YREG(I,IT), DIH(I,IT), AMCD	GEOM 980
WRITE (29,3)	XREG(I,IT) , YREG(I,IT), DIH(I,IT), AMCD	GEOM 990
MCD(I,IT) = AMCD		GEOM1000
IF (I.EQ.1)	GO TO 59	GEOM1010
IF (MAK.NE.0 .OR. MCD(I-1,IT).NE.2)	GO TO 49	GEOM1020
MAK	= I-1	GEOM1030
49 IF (ABS(YREG(I-1,IT)-YREG(I,IT)) .LT. YTOL)	GO TO 50	GEOM1040
AREG(I-1,IT) = (XREG(I-1,IT)-XREG(I,IT))/(YREG(I-1,IT)-YREG(I,IT))		GEOM1050
ASWP = ATAN (AREG(I-1,IT)) * RAD		GEOM1060
GO TO 51		GEOM1070
50 YREG(I,IT) = YREG(I-1,IT)		GEOM1080
AREG(I-1,IT) = AZY		GEOM1090
ASWP	= 90.	GEOM1100
51 J	= I - 1	GEOM1110
C		GEOM1120
C	WRITE PLANFORM PERIMETER POINTS AND ANGLES	GEOM1130
C		

	IF (ABS(SB - RSAR(K)).GT. (.1/RAD))	GO TO 111	GEOM1740
C	REFERENCE PLANFORM COORDINATES ARE STORED UNCHANGED FOR WINGS		GEOM1750
C	WITHOUT CHANGE IN SWEEP		GEOM1760
	DO 113 I=1,N		GEOM1770
	X(I)=XREF(I)		GEOM1780
	Y(I)=YREF(I)		GEOM1790
	IF (RSAR(I) .EQ. PIT)	GO TO 114	GEOM1800
	A(I)=TAN(RSAR(I))		GEOM1810
	GO TO 113		GEOM1820
114	A(I)=AZY		GEOM1830
113	SAR(I)=RSAR(I)		GEOM1840
	X(N1)=XREF(N1)		GEOM1850
	Y(N1)=YREF(N1)		GEOM1860
	GO TO 103		GEOM1870
C			GEOM1880
C	CHANGES IN WING SWEEP ARE MADE HERE		GEOM1890
C			GEOM1900
111	IF (MCD(K,IT) .NE.2)	GO TO 1007	GEOM1910
	KA=K-1		GEOM1920
	DO 81 I=1,KA		GEOM1930
	X(I)=XREF(I)		GEOM1940
	Y(I)=YREF(I)		GEOM1950
81	SAR(I)=RSAR(I)		GEOM1960
C	DETERMINE LEADING EDGE INTERSECTION BETWEEN FIXED AND VARIABLE		GEOM1970
C	SWEEP WING SECTIONS		GEOM1980
	SAR(K) = SB		GEOM1990
	A(K) = TAN(SB)		GEOM2000
	SAI = SB-RSAR(K)		GEOM2010
	X(K+1)=XS(IT)+(XREF(K+1)-XS(IT))*COS(SAI)+(YREF(K+1)-YS(IT))		GEOM2020
1	*SIN(SAI)		GEOM2021
	Y(K+1)=YS(IT)+(YREF(K+1)-YS(IT))*COS(SAI)-(XREF(K+1)-XS(IT))		GEOM2030
1	*SIN(SAI)		GEOM2031
	IF (ABS(SB - SAR(K-1)) .LT. (.1/RAD))	GO TO 84	GEOM2040
	Y(K)=X(K+1)-X(K-1)-A(K)*Y(K+1)+A(K-1)*Y(K-1)		GEOM2050
	Y(K)=Y(K)/(A(K-1)-A(K))		GEOM2060
	X(K)=A(K)*X(K-1)-A(K-1)*X(K+1)+A(K-1)*A(K)*(Y(K+1)-Y(K-1))		GEOM2070
	X(K)=X(K)/(A(K)-A(K-1))		GEOM2080
	GO TO 85		GEOM2090
C	ELIMINATE EXTRANEIOUS BREAKPOINTS		GEOM2100
84	X(K)=XREF(K-1)		GEOM2110
	Y(K)=YREF(K-1)		GEOM2120
	SAR(K) = SAR(K-1)		GEOM2130
85	K=K+1		GEOM2140
C	SWEEP THE BREAKPOINTS ON THE VARIABLE SWEEP PANEL		GEOM2150
C	(IT ALSO KEEPS SWEEP ANGLES IN FIRST OR FOURTH QUADRANTS)		GEOM2160
86	K=K+1		GEOM2170
	SAR(K-1)=SAI+RSAR(K-1)		GEOM2180
99	IF (SAR(K-1) .LE. PIT)	GO TO 102	GEOM2190
	SAR(K-1)=SAR(K-1)-3.1415927		GEOM2200
	GO TO 99		GEOM2210
102	IF (SAR(K-1) .GE. (-PIT))	GO TO 106	GEOM2220
	SAR(K-1)=SAR(K-1)+3.1415927		GEOM2230
	GO TO 102		GEOM2240
106	IF((SAR(K-1)) .LT..0) GO TO 108		GEOM2250
	IF (SAR(K-1) - PIT)	90,87,87	GEOM2260
108	IF (SAR(K-1) + PIT)	89,89,90	GEOM2270
87	A(K-1)=AZY		GEOM2280
	GO TO 91		GEOM2290
89	A(K-1)=-AZY		GEOM2300
	GO TO 91		GEOM2310
90	A(K-1)=TAN(SAR(K-1))		GEOM2320
91	KK = MCD(K,IT)		GEOM2330
	GO TO (93,92),KK		GEOM2340
92	Y(K)=YS(IT)+(YREF(K)-YS(IT))*COS(SAI)-(XREF(K)-XS(IT))		GEOM2350
1	*SIN(SAI)		GEOM2351
	X(K)=XS(IT)+(XREF(K)-XS(IT))*COS(SAI)+(YREF(K)-YS(IT))		GEOM2360
1	*SIN(SAI)		GEOM2361

	GO TO 86	GEOM2370
C	DETERMINE THE TRAILING EDGE INTERSECTION	GEOM2380
C	BETWEEN FIXED AND VARIABLE SWEEP WING SECTIONS	GEOM2390
93	IF (ABS (RSAR(K)-SAR(K-1)) .LT. (.1/RAD)) GO TO 96	GEOM2400
	Y(K)=XREF(K+1)-X(K-1)-A(K)*YREF(K+1)+A(K-1)*Y(K-1)	GEOM2410
	Y(K)=Y(K)/(A(K-1)-A(K))	GEOM2420
	X(K)=A(K)*X(K-1)-A(K-1)*XREF(K+1)+A(K-1)*A(K)*(YREF(K+1)-Y(K-1))	GEOM2430
	X(K)=X(K)/(A(K)-A(K-1))	GEOM2440
	GO TO 97	GEOM2450
96	X(K)=XREF(K+1)	GEOM2460
	Y(K)=YREF(K+1)	GEOM2470
97	K=K+1	GEOM2480
C	STORE REFERENCE PLANFORM COORDINATES ON INBOARD FIXED TRAILING	GEOM2490
C	EDGE	GEOM2500
	DO 98 I=K,N1	GEOM2510
	X(I)=XREF(I)	GEOM2520
	Y(I)=YREF(I)	GEOM2530
98	SAR(I-1)=RSAR(I-1)	GEOM2540
103	DO 101 I=1,N	GEOM2550
	XX(I,IT) = X(I)	GEOM2560
	YY(I,IT) = Y(I)	GEOM2570
	MMCD(I,IT)= MCD(I,IT)	GEOM2580
	TTWD(I,IT) = DIH(I,IT)	GEOM2590
101	AS (I,IT) = A(I)	GEOM2600
	XX(N1,IT) = X(N1)	GEOM2610
	YY(N1,IT) = Y(N1)	GEOM2620
	AN(IT) = AAN(IT)	GEOM2630
100	CONTINUE	GEOM2640
C		GEOM2650
C	LINE UP BREAKPOINTS AMONG PLANFORMS	GEOM2660
C		GEOM2670
299	BOTSV(1)=0	GEOM2680
	BOTSV(2)=0.	GEOM2690
	WRITE (29,16)	GEOM2700
	DO 180 IT=1,IPLAN	GEOM2710
	NIT=AN(IT)+1	GEOM2720
	DO 178 ITT=1,IPLAN	GEOM2730
	IF (ITT.EQ.IT) GO TO 178	GEOM2740
	NITT=AN(ITT)+1	GEOM2750
	DO 176 I=1,NITT	GEOM2760
	JPSV=0	GEOM2770
	DO 166 JP=1,NIT	GEOM2780
	IF(YY(JP,IT) .EQ.YY(I,ITT)) GO TO 176	GEOM2790
166	CONTINUE	GEOM2800
	DO 170 JP=1,NIT	GEOM2810
	IF (YY(JP,IT).LT.YY(I,ITT)) GO TO 168	GEOM2820
170	CONTINUE	GEOM2830
	GO TO 176	GEOM2840
168	JPSV = JP	GEOM2850
	IND = NIT -(JPSV -1)	GEOM2860
	DO 172 JP=1,IND	GEOM2870
	K2 = NIT -JP +2	GEOM2880
	K1 = NIT -JP +1	GEOM2890
	XX(K2,IT) = XX(K1,IT)	GEOM2900
	YY(K2,IT) = YY(K1,IT)	GEOM2910
	MMCD(K2,IT)= MMCD(K1,IT)	GEOM2920
	AS(K2,IT) = AS(K1,IT)	GEOM2930
172	TTWD(K2,IT)=TTWD(K1,IT)	GEOM2940
	YY(JPSV,IT) = YY(I,ITT)	GEOM2950
	AS(JPSV,IT) = AS(JPSV-1,IT)	GEOM2960
	TTWD(JPSV,IT)= TTWD(JPSV-1,IT)	GEOM2970
	XX(JPSV,IT) = (YY(JPSV,IT) - YY(JPSV-1,IT)) * AS(JPSV-1,IT)	GEOM2980
1	+ XX(JPSV-1,IT)	GEOM2990
	MMCD(JPSV,IT) = MMCD(JPSV-1,IT)	GEOM3000
	AN(IT) = AN(IT) + 1.	GEOM3010
	NIT = NIT + 1	GEOM3020
176	CONTINUE	

178	CONTINUE	GEOM3030
C		GEOM3040
C	SEQUENCE WING COORDINATES FROM TIP TO ROOT	GEOM3050
C		GEOM3060
	N1 = AN(IT)+ 1.	GEOM3070
	DO 203 I=1,N1	GEOM3080
203	Q(I) = YY(I,IT)	GEOM3090
	DO 208 J=1,N1	GEOM3100
	HIGH = 1.	GEOM3110
	DO 205 I=1,N1	GEOM3120
	IF ((Q(I)-HIGH).GE.0.) GO TO 205	GEOM3130
	HIGH = Q(I)	GEOM3140
	IH = I	GEOM3150
205	CONTINUE	GEOM3160
	IF (J.NE.1) GO TO 206	GEOM3170
	BOTSV(IT) = HIGH	GEOM3180
	KFX(IT) = IH	GEOM3190
206	Q (IH) = 1.	GEOM3200
	SPY(J,IT) = HIGH	GEOM3210
	IF (IH.GT.KFX(IT)) GO TO 209	GEOM3220
	IYL(J,IT) = 1	GEOM3230
	IYT(J,IT) = 0	GEOM3240
	GO TO 208	GEOM3250
209	IYL(J,IT) = 0	GEOM3260
	IYT(J,IT) = 1	GEOM3270
208	CONTINUE	GEOM3280
180	CONTINUE	GEOM3290
C		GEOM3300
C	SELECT MAXIMUM B/2 AS THE WING SEMISPAN	GEOM3310
C		GEOM3320
	KBOT = 1	GEOM3330
	IF (BOTSV(1).GE.BOTSV(2)) KBOT = 2	GEOM3340
	BOT = BOTSV(KBOT)	GEOM3350
C		GEOM3360
C	COMPUTE NOMINAL HORSESHOE VORTEX WIDTH ALONG WING SURFACE	GEOM3370
C		GEOM3380
	TSPAN = 0	GEOM3390
	ISAVE = KFX(KBOT) - 1	GEOM3400
	I = KFX(KBOT) - 2	GEOM3410
216	IF (I.EQ.0) GO TO 217	GEOM3420
	IF(TTWD(I,KBOT).EQ.TTWD(ISAVE,KBOT)) GO TO 218	GEOM3430
217	CTWD = COS(ATAN(TTWD(ISAVE,KBOT)))	GEOM3440
	TLGTH = (YY(ISAVE+1,KBOT) - YY(I+1,KBOT)) / CTWD	GEOM3450
	TSPAN = TSPAN + TLGTH	GEOM3460
	IF (I.EQ.0) GO TO 219	GEOM3470
	ISAVE = I	GEOM3480
218	I = I -1	GEOM3490
	GO TO 216	GEOM3500
219	VI = TSPAN / VIC	GEOM3510
	VSTOL = VI / 2	GEOM3520
C		GEOM3530
C	ELIMINATE PLANFORM BREAKPOINTS WHICH ARE WITHIN (B/2)/2000 UNITS	GEOM3540
C	LATERALLY	GEOM3550
C		GEOM3560
	DO 220 IT = 1,IPLAN	GEOM3570
	N = AN(IT)	GEOM3580
	N1= N + 1	GEOM3590
	DO 220 J=1,N	GEOM3600
	AA = ABS(SPY(J,IT) - SPY(J+1,IT))	GEOM3610
	IF (AA.EQ.0. .OR. AA.GT.ABS(TSPAN/2000.)) GO TO 220	GEOM3620
	IF (AA.GT.YTOL) WRITE(6,19) SPY(J+1,IT) , SPY(J,IT)	GEOM3630
	DO 222 I=1,N1	GEOM3640
	IF (YY(I,IT).NE.SPY(J+1,IT)) GO TO 222	GEOM3650
	YY(I,IT) = SPY(J,IT)	GEOM3660
222	CONTINUE	GEOM3670
	SPY(J+1,IT) = SPY(J,IT)	GEOM3680
220	CONTINUE	GEOM3690

C		GEOM3700
C	COMPUTE Z COORDINATES	GEOM3710
C		GEOM3720
	DO 236 IT=1,IPLAN	GEOM3730
	JM = AN(IT) + 1.	
	N1 = AN(IT) + 1.	GEOM3740
	DO 230 JZ=1,N1	GEOM3750
230	ZZ(JZ,IT) = RTCDHT(IT)	GEOM3760
	JZ = 1	GEOM3770
232	JZ = JZ + 1	GEOM3780
	IF (JZ.GT.KFX(IT)) GO TO 234	GEOM3790
	ZZ(JZ,IT) = ZZ(JZ-1,IT) + (YY(JZ,IT) - YY(JZ-1,IT)) * TTWD(JZ-1,IT)	GEOM3800
	GO TO 232	GEOM3810
234	JM = JM-1	GEOM3820
	IF (JM.EQ.KFX(IT)) GO TO 236	GEOM3830
	ZZ(JM,IT) = ZZ(JM+1,IT) + (YY(JM,IT) - YY(JM+1,IT)) * TTWD(JM,IT)	GEOM3840
	GO TO 234	GEOM3850
236	CONTINUE	GEOM3860
C		GEOM3870
C	WRITE PLANFORM PERIMETER POINTS ACTUALLY USED IN THE COMPUTATIONS	GEOM3880
C		GEOM3890
	WRITE (29,8)	GEOM3900
	DO 240 IT=1,IPLAN	GEOM3910
	N = AN(IT)	GEOM3920
	N1 = N + 1	GEOM3930
	IF (IT.EQ.2) WRITE (29,18)	GEOM3940
	DO 238 KK=1,N	GEOM3950
	TOUT = ATAN (TTWD(KK,IT)) * RAD	GEOM3960
	AOUT = ATAN(AS(KK,IT)) * RAD	GEOM3970
	IF (AS(KK,IT).EQ.AZY) AOUT=90.	GEOM3980
	WRITE (29,9) KK,XX(KK,IT), YY(KK,IT), ZZ(KK,IT), AOUT,	GEOM3990
	1 TOUT ,MMCD(KK,IT)	GEOM4000
238	CONTINUE	GEOM4010
	WRITE (29,9) N1,XX(N1,IT), YY(N1,IT), ZZ(N1,IT)	GEOM4020
240	CONTINUE	GEOM4030
C		GEOM4040
C	PART ONE - SECTION THREE - LAY OUT YAWED HORSESHOE VORTICES	GEOM4050
C		GEOM4060
	STRUE = 0.	GEOM4070
	NSSWSV(1) = 0	
	NSSWSV(2) = 0	
	MSV(1) = 0	
	MSV(2) = 0	GEOM4080
700	DO 722 IT=1,IPLAN	GEOM4090
	N1 = AN(IT) + 1.	GEOM4100
	I = 0	GEOM4110
	J = 1	GEOM4120
	YIN = BOTSV(IT)	GEOM4130
	ILE = KFX(IT)	
	ITE = KFX(IT)	GEOM4140
C	DETERMINE SPANWISE BORDERS OF HORSESHOE VORTICES	GEOM4150
701	IXL = 0	GEOM4160
	IXT = 0	GEOM4170
	I = I + 1	GEOM4180
	IF(YIN.GE.(SPY(J,IT)+VSTOL)) GO TO 703	GEOM4190
C	BORDER IS WITHIN VORTEX SPACING TOLERANCE (VSTOL) OF BREAKPOINT	GEOM4200
C	THEREFORE USE THE NEXT BREAKPOINT INBOARD FOR THE BORDER	GEOM4210
	VBORD(I) = YIN	GEOM4220
	GO TO 707	GEOM4230
C	USE NOMINAL VORTEX SPACING TO DETERMINE THE BORDER	GEOM4240
703	VBORD(I) = SPY(J,IT)	GEOM4250
C	COMPUTE SUBSCRIPTS ILE AND ITE TO INDICATE WHICH	GEOM4260
C	BREAKPOINTS ARE ADJACENT AND WHETHER THEY ARE ON THE WING LEADING	GEOM4270
C	EDGE OR THE TRAILING EDGE	GEOM4280
715	IF (J.GE.N1) GO TO 706	GEOM4290
	IF (SPY(J,IT).NE.SPY(J+1,IT)) GO TO 706	GEOM4300
	IXL = IXL + IYL(J,IT)	

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IXT      = IXT + IYT(J,IT)
J        = J + 1
GO TO 715
706 YIN   = SPY(J,IT)
IXL      = IXL + IYL(J,IT)
IXT      = IXT + IYT(J,IT)
J        = J + 1
707 CPHI  = COS ( ATAN ( TTWD(ILE,IT) ) )
IPHI = ILE - IXL
IF ( J.GE.N1 )      IPHI = 1
YIN = YIN - VI* COS ( ATAN ( TTWD(IPHI,IT) ) )
IF ( I.NE.1 )      GO TO 709
708 ILE   = ILE - IXL
ITE      = ITE + IXT
GO TO 701
C      COMPUTE COORDINATES FOR CHORDWISE ROW OF HORSESHOE VORTICES
709 YQ    = ( VBORD(I-1) + VBORD(I) ) / 2.
HW       = ( VBORD(I) - VBORD(I-1)) / 2.
IM1      = I - 1 + NSSWSV(1)
ZH(IM1)  = ZZ(ILE,IT) + ( YQ - YY(ILE,IT) ) * TTWD(ILE,IT)
PHI(IM1) = TTWD(ILE,IT)
SSWWA(IM1) = AS(ILE,IT)
XLE      = XX(ILE,IT) + AS(ILE,IT) * (YQ - YY(ILE,IT) )
XTE      = XX(ITE,IT) + AS(ITE,IT) * (YQ - YY(ITE,IT) )
XLOCAL   = ( XLE - XTE ) / TBLSCW(IM1)
C
C      COMPUTE WING AREA PROJECTED TO THE X - Y PLANE
C
STRUE    = STRUE + XLOCAL * TBLSCW(IM1) * (HW * 2.) * 2.
C
NSCW     = TBLSCW(IM1)
DO 720 JCW=1,NSCW
AJCW     = JCW - 1
XLEL     = XLE - AJCW * XLOCAL
NTS      = JCW + MSV(1) + MSV(2)
PN(NTS)  = XLEL - .25 * XLOCAL
PV(NTS)  = XLEL - .75 * XLOCAL
PSI(NTS) = ((XLE - PN(NTS))*AS(ITE,IT) + (PN(NTS) - XTE)*AS(ILE,
1      IT) ) / (XLE - XTE) * CPHI
S(NTS)   = HW / CPHI
Q(NTS)   = YQ
720 CONTINUE
MSV(IT)  = MSV(IT) + NSCW
C
C      TEST TO DETERMINE WHEN WING ROOT (Y=0) IS REACHED
C      IF ( VBORD(I) .LT. -0.)      GO TO 708
C
NSSWSV(IT) = I - 1
722 CONTINUE
M         = MSV(1) + MSV(2)
C
C      COMPUTE ASPECT RAT10 AND AVERAGE CHORD
C
BOT       = - BOT
AR        = 4. * BOT * BOT / SREF
ARTRUE    = 4. * BOT * BOT / STRUE
CAVE      = STRUE / ( 2. * BOT )
BETA      = ( 1. - MACH* MACH) ** .5
NVTWO     = 0
DO 354 IT=1,IPLAN
NVONE     = 1 + (IT-1)*MSV(1)
NVTWO     = NVTWO + MSV(IT)
IF (TWIST(IT) .LE. 0. )      GO TO 350
READ(28,3) (ALP(NV),ALP(NV+1),ALP(NV+2),ALP(NV+3),ALP(NV+4),ALP(NVGEOM4940
1      +5),ALP(NV+6),ALP(NV+7),NV=NVONE,NVTWO,8)
C      write(29,3) (ALP(NV),ALP(NV+1),ALP(NV+2),ALP(NV+3),ALP(NV+4),
C      1      ALP(NV

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C	1	+5),ALP(NV+6),ALP(NV+7),NV=NVONE,NVTWO,6)	GEOM4950
		GO TO 354	GEOM4960
	350	DO 351 NV = NVONE , NVTWO	GEOM4970
	351	ALP(NV) = 0.	GEOM4980
	354	CONTINUE	GEOM4990
		WRITE (29,24) M	GEOM5000
		WRITE (29,25) (IT,MSV(IT),NSSWSV(IT), IT=1,IPLAN)	GEOM5010
		IF (SCW.NE.0.) WRITE (29,20) SCW	GEOM5020
		IF (SCW.EQ.0.) WRITE (29,22) (TBLSCW(I),I=1,NSTA)	GEOM5030
C			GEOM5040
C		APPLY PRANDTL-GLAUERT CORRECTION	GEOM5050
C			GEOM5060
		DO 360 NV = 1,M	GEOM5070
		PSI(NV) = ATAN(PSI(NV)/BETA)	GEOM5080
		PN(NV) = PN(NV) / BETA	GEOM5090
	360	PV(NV) = PV(NV) / BETA	GEOM5100
		RETURN	GEOM5110
	1006	ICODEOF = 1	GEOM5120
		WRITE(29,11) CONFIG	GEOM5130
		RETURN	GEOM5140
	1007	ICODEOF = 2	GEOM5150
		WRITE(29,12) K,IT	GEOM5160
		RETURN	GEOM5170
	1008	ICODEOF = 3	GEOM5180
		WRITE (29,15) PTEST,QTEST	GEOM5190
		RETURN	GEOM5200
		END	GEOM5210
		SUBROUTINE MATX	MATX 20
		DIMENSION YY(2),FU(2),FV(2),FW(2),FVN(300,300),IPIVOT(300),	MATX 30
	1	INDEX(300,2)	MATX 40
		COMMON/ALL/ BOT,M,BETA,PTEST,QTEST,TBLSCW(50),Q(300),PN(300),	MATX 50
	1	PV(300),ALP(300),S(300),PSI(300),PHI(300),ZH(50)	MATX 60
		COMMON/TOTHR/ CIR(300,2),SECTRST(50)	MATX 70
		COMMON/INSUB23/APSI,APHI ,XX ,YYY,ZZ ,SNN,TOLC	MATX 80
C			MATX 90
C			MATX 100
C			MATX 110
C		PART 2 - COMPUTE CIRCULATION TERMS	MATX 120
C			MATX 130
C			MATX 140
C			MATX 150
		FPI = 12.5663704	MATX 160
		IM = 2	MATX 162
		NMAX = 300	MATX 164
C			MATX 170
C			MATX 180
C			MATX 190
C		THE TOLERANCE SET AT THIS POINT IN THE PROGRAM MAY NEED TO BE	MATX 200
C		CHANGED FOR COMPUTERS OTHER THAN THE CDC 6000 SERIES	MATX 210
C			MATX 220
		TOLC=(BOT*15.E-05)**2	MATX 230
		DO 6667 NUU=1,NMAX	MATX 240
		DO 6667 NUT=1,NMAX	MATX 250
		FVN(NUU,NUT)=0.	MATX 260
6667		CONTINUE	MATX 270
		DO 308 NV=1,M	MATX 280
		CIR(NV,1)= 12.5663704 * ALP(NV)	MATX 290
		CIR(NV,2)= 12.5663704	MATX 300
		IF (PTEST.NE.0.) CIR(NV,2) = -1.0964155 * Q(NV) / BOT	MATX 310
		IF (QTEST.NE.0.) CIR(NV,2) = -1.0964155 * PV(NV) *BETA	MATX 320
308		CONTINUE	MATX 330
		IZZ=1	MATX 340
		NNV=TBLSCW(IZZ)	MATX 350
		DO 314 NV=1,M	MATX 360
		IZ=1	MATX 370
		NNN=TBLSCW(IZ)	MATX 380
		DO 316 NN=1,M	MATX 390

APHI	= ATAN (PHI (IZ))	MATX 400
APSI	= PSI (NN)	MATX 410
XX	= PV (NV) - PN (NN)	
YY (1)	= Q (NV) - Q (NN)	
YY (2)	= Q (NV) + Q (NN)	MATX 420
ZZ	= ZH (IZZ) - ZH (IZ)	MATX 430
SNN	= S (NN)	MATX 440
DO 261 I=1,2		MATX 450
YYY	= YY (I)	MATX 460
CALL INFSUB (BOT, FU (I), FV (I), FW (I))		MATX 470
APHI	= -APHI	
APSI	= -APSI	MATX 480
261 CONTINUE		MATX 490
IF (PTEST.NE.0.) GO TO 342		MATX 500
FVN (NV, NN) = FW (1) - FV (1) * PHI (IZ) + FW (2) - FV (2) * PHI (IZ)		MATX 510
GO TO 312		MATX 520
342 FVN (NV, NN) = FW (1) - FV (1) * PHI (IZ) - FW (2) + FV (2) * PHI (IZ)		MATX 530
312 IF (NN.LT.NNN .OR. NN.EQ.M) GO TO 316		MATX 540
IZ=IZ+1		MATX 550
NNN=NNN+TBLSCW (IZ)		MATX 560
316 CONTINUE		MATX 570
IF (NV.LT.NNV .OR. NV.EQ.M) GO TO 314		MATX 580
IZZ=IZZ+1		MATX 590
NNV=NNV+TBLSCW (IZZ)		MATX 600
314 CONTINUE		MATX 610
CALL AMATINV (FVN, M, CIR, IM, DETERM, IPIVOT, INDEX, NMAX, ISCALE)		MATX 620
IZZA = IZZ		MATX 630
DO 320 NZ=1, IZZA		MATX 640
320 SECTRST (NZ) = 0.		MATX 650
IZZ=1		MATX 660
NNV=TBLSW (IZZ)		MATX 670
DO 614 NV=1, M		MATX 680
IZ=1		MATX 690
NNN=TBLSW (IZ)		MATX 700
VELIN = 0.		MATX 710
DO 616 NN=1, M		MATX 720
APHI = ATAN (PHI (IZ))		MATX 730
APSI = PSI (NN)		MATX 740
XX=PN (NV) - PN (NN)		MATX 750
YY (1) = Q (NV) - Q (NN)		MATX 760
YY (2) = Q (NV) + Q (NN)		MATX 770
ZZ=ZH (IZZ) - ZH (IZ)		MATX 780
SNN = S (NN)		MATX 790
DO 661 I=1,2		MATX 800
YYY = YY (I)		MATX 810
CALL INFSUB (BOT, FU (I), FV (I), FW (I))		MATX 820
APHI = -APHI		MATX 830
APSI = -APSI		MATX 840
661 CONTINUE		MATX 850
VELIN = ((FW (1) + FW (2)) - (FV (1) + FV (2)) * TAN (APHI)) * CIR (NN, 2)		MATX 860
1 /FPI + VELIN		MATX 870
IF (NN.LT.NNN .OR. NN.EQ.M) GO TO 616		MATX 880
IZ=IZ+1		MATX 890
NNN=NNN+TBLSCW (IZ)		MATX 900
616 CONTINUE		MATX 910
CTCP = - (VELIN - 1.) * 2. * CIR (NV, 2)		MATX 920
SECTRST (IZZ) = SECTRST (IZZ) + CTCP		MATX 930
IF (NV.LT.NNV .OR. NV.EQ.M) GO TO 614		MATX 940
IZZ=IZZ+1		MATX 950
NNV=NNV+TBLSCW (IZZ)		MATX 960
614 CONTINUE		MATX 970
RETURN		MATX 980
END		MATX 990
SUBROUTINE AMATINV (A, N, B, M, DETERM, IPIVOT, INDEX, NMAX, ISCALE)		MINV 10
C***** DOCUMENT DATE 08-01-68 SUBROUTINE REVISED 08-01-68 *****		MINV 20
C		MINV 30
C MATRIX INVERSION WITH ACCOMPANYING SOLUTION OF LINEAR EQUATIONS		MINV 40

C	DIMENSION IPIVOT(N),A(NMAX,N),B(NMAX,M),INDEX(NMAX,2)	MINV 50
	EQUIVALENCE (IROW,JROW), (ICOLUM,JCOLUM), (AMAX, T, SWAP)	MINV 60
C		MINV 70
C	INITIALIZATION	MINV 80
C		MINV 90
	5 ISCALE=0	MINV 100
	6 R1=10.0**35	MINV 110
	7 R2=1.0/R1	MINV 120
	10 DETERM=1.0	MINV 130
	15 DO 20 J=1,N	MINV 140
	20 IPIVOT(J)=0	MINV 150
	30 DO 550 I=1,N	MINV 160
C		MINV 170
C	SEARCH FOR PIVOT ELEMENT	MINV 180
C		MINV 190
	40 AMAX=0.0	MINV 200
	45 DO 105 J=1,N	MINV 210
	50 IF (IPIVOT(J)-1) 60, 105, 60	MINV 220
	60 DO 100 K=1,N	MINV 230
	70 IF (IPIVOT(K)-1) 80, 100, 740	MINV 240
	80 IF (ABS(AMAX)-ABS(A(J,K)))85,100,100	MINV 250
	85 IROW=J	MINV 260
	90 ICOLUM=K	MINV 270
	95 AMAX=A(J,K)	MINV 280
	100 CONTINUE	MINV 290
	105 CONTINUE	MINV 300
	IF (AMAX) 110,106,110	MINV 310
	106 DETERM=0.0	MINV 320
	ISCALE=0	MINV 330
	GO TO 740	MINV 340
	110 IPIVOT(ICOLUM)=IPIVOT(ICOLUM)+1	MINV 350
C		MINV 360
C	INTERCHANGE ROWS TO PUT PIVOT ELEMENT ON DIAGONAL	MINV 370
C		MINV 380
	130 IF (IROW-ICOLUM) 140, 260, 140	MINV 390
	140 DETERM=-DETERM	MINV 400
	150 DO 200 L=1,N	MINV 410
	160 SWAP=A(IROW,L)	MINV 420
	170 A(IROW,L)=A(ICOLUM,L)	MINV 430
	200 A(ICOLUM,L)=SWAP	MINV 440
	205 IF(M) 260, 260, 210	MINV 450
	210 DO 250 L=1, M	MINV 460
	220 SWAP=B(IROW,L)	MINV 470
	230 B(IROW,L)=B(ICOLUM,L)	MINV 480
	250 B(ICOLUM,L)=SWAP	MINV 490
	260 INDEX(I,1)=IROW	MINV 500
	270 INDEX(I,2)=ICOLUM	MINV 510
	310 PIVOT=A(ICOLUM,ICOLUM)	MINV 520
	IF (PIVOT) 1000,106,1000	MINV 530
C		MINV 540
C	SCALE THE DETERMINANT	MINV 550
C		MINV 560
	1000 PIVOTI=PIVOT	MINV 570
	1005 IF (ABS(DETERM)-R1) 1030,1010,1010	MINV 580
	1010 DETERM=DETERM/R1	MINV 590
	ISCALE=ISCALE+1	MINV 600
	IF (ABS(DETERM)-R1) 1060,1020,1020	MINV 610
	1020 DETERM=DETERM/R1	MINV 620
	ISCALE=ISCALE+1	MINV 630
	GO TO 1060	MINV 640
	1030 IF (ABS(DETERM)-R2) 1040,1040,1060	MINV 650
	1040 DETERM=DETERM*R1	MINV 660
	ISCALE=ISCALE-1	MINV 670
	IF (ABS(DETERM)-R2) 1050,1050,1060	MINV 680
	1050 DETERM=DETERM*R1	MINV 690
	ISCALE=ISCALE-1	MINV 700
		MINV 710

1060 IF (ABS (PIVOTI) - R1) 1090, 1070, 1070	MINV 720
1070 PIVOTI=PIVOTI/R1	MINV 730
ISCALE=ISCALE+1	MINV 740
IF (ABS (PIVOTI) - R1) 320, 1080, 1080	MINV 750
1080 PIVOTI=PIVOTI/R1	MINV 760
ISCALE=ISCALE+1	MINV 770
GO TO 320	MINV 780
1090 IF (ABS (PIVOTI) - R2) 2000, 2000, 320	MINV 790
2000 PIVOTI=PIVOTI*R1	MINV 800
ISCALE=ISCALE-1	MINV 810
IF (ABS (PIVOTI) - R2) 2010, 2010, 320	MINV 820
2010 PIVOTI=PIVOTI*R1	MINV 830
ISCALE=ISCALE-1	MINV 840
320 DETERM=DETERM*PIVOTI	MINV 850
C	MINV 860
C DIVIDE PIVOT ROW BY PIVOT ELEMENT	MINV 870
C	MINV 880
330 A(ICOLUM, ICOLUM)=1.0	MINV 890
340 DO 350 L=1, N	MINV 900
350 A(ICOLUM, L)=A(ICOLUM, L)/PIVOT	MINV 910
355 IF (M) 380, 380, 360	MINV 920
360 DO 370 L=1, M	MINV 930
370 B(ICOLUM, L)=B(ICOLUM, L)/PIVOT	MINV 940
C	MINV 950
C REDUCE NON-PIVOT ROWS	MINV 960
C	MINV 970
380 DO 550 L1=1, N	MINV 980
390 IF (L1-ICOLUM) 400, 550, 400	MINV 990
400 T=A(L1, ICOLUM)	MINV1000
420 A(L1, ICOLUM)=0.0	MINV1010
430 DO 450 L=1, N	MINV1020
450 A(L1, L)=A(L1, L)-A(ICOLUM, L)*T	MINV1030
455 IF (M) 550, 550, 460	MINV1040
460 DO 500 L=1, M	MINV1050
500 B(L1, L)=B(L1, L)-B(ICOLUM, L)*T	MINV1060
550 CONTINUE	MINV1070
C	MINV1080
C INTERCHANGE COLUMNS	MINV1090
C	MINV1100
600 DO 710 I=1, N	MINV1110
610 L=N+1-I	MINV1120
620 IF (INDEX(L, 1)-INDEX(L, 2)) 630, 710, 630	MINV1130
630 JROW=INDEX(L, 1)	MINV1140
640 JCOLUM=INDEX(L, 2)	MINV1150
650 DO 705 K=1, N	MINV1160
660 SWAP=A(K, JROW)	MINV1170
670 A(K, JROW)=A(K, JCOLUM)	MINV1180
700 A(K, JCOLUM)=SWAP	MINV1190
705 CONTINUE	MINV1200
710 CONTINUE	MINV1210
740 RETURN	MINV1220
END	MINV1230
SUBROUTINE AERO	AERO 20
REAL MACH	
DIMENSION CPM(2), YCP(2), YY(2), VOU(300, 2), UOU(300, 2), FU(2), FV(2),	AERO 30
1XTLEG(60), CHLFT(300, 2), CLCC(300, 2), YTLEG(50), SLDT(50), CLA(2), SUM(2)	AERO 40
2), AC(2), CH(2, 50), CCAV(2, 50), CLCL(2, 50), CP(120), FW(2)	AERO 50
3, DIFCIRS(25), YLEGSV(25), ZLEGSV(25), CLPT(300, 2), CLPB(300, 2)	AERO 60
COMMON/ALL/ BOT, M, BETA, PTEST, QTEST, TBLSCW(50), Q(300), PN(300),	AERO 70
1 PV(300), ALP(300), S(300), PSI(300), PHI(300), ZH(50)	AERO 80
COMMON/TOTHRE/ CIR(300, 2), SECTRST(50)	AERO 90
COMMON/ONETHRE/TWIST(2), CREF, SREF, CAVE, CLDES, STRUE, AR, ARTRUE,	AERO 100
1 RTCDHT(2), CONFIG, NSSWSV(2), MSV(2), KBOT, PLAN, IPLAN, MACH	AERO 110
2 , SSWWA(50)	AERO 120
COMMON/THRECDI/SLOAD(3, 50)	AERO 130
COMMON/INSUB23/APSI, APhi , XX , YYY, ZZ , SNN, TOLCSQ	AERO 140
CHARACTER*8 HEAD	

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1 FORMAT (/ 12X, 'SECOND PLANFORM HORSESHOE VORTEX DESCRIPTIONS' / )AERO 150
3 FORMAT(6F12.5)AERO 160
4 FORMAT ( ///58X,16HAERODYNAMIC DATA,///54X, 'CONFIGURATION AERO 170
  INO.',F7.0 // )AERO 180
5 FORMAT(///18X,'COMPLETE CONFIGURATION',31X,'WING-BODY CHARACTERISTAERO 190
  1ICS',/ 64X,'LIFT', 9X,'INDUCED DRAG (FAR FIELD SOLUTION)',// AERO 200
  2 16X, A8,' CL COMPUTED ALPHA',19X,'CL(WB)',7X,'CDI AT CL(WB)', AERO 210
  3 4X ,15HCDI/(CL(WB)**2),/ 88X,12H(1/(PI*AR) =,F8.5,' ' ) ) AERO 220
6 FORMAT (11X,2F15.5,15X,3F15.5)AERO 230
7 FORMAT(///4X,11H REF. CHORD,6X,25HC AVERAGE TRUE WING AREA,3X,13AERO 240
  1HREF WING AREA,9X,3HB/2, 8X,7HREF. AR,8X,7HTRUE AR,4X,11HMACH NUMBAERO 250
  2ER/)AERO 260
8 FORMAT(8F15.5)AERO 270
11 FORMAT (/// 47X,'COMPLETE CONFIGURATION CHARACTERISTICS',// AERO 280
  1 36X,'CL ALPHA',8X,'CL(TWIST) ALPHA AT CL=0 Y CP CM/CL AERO 290
  2 CMO',/ 27X,'PER RADIAN PER DEGREE',/ 24X,7F12.5 ) AERO 300
12 FORMAT(//25X,'ADDITIONAL LOADING AT',/23X,'L(TOTAL)/(Q*S(TRUE)) = AERO 310
  11.0',/67X,'LOAD DUE ADD. LOAD AT BASIC LOAD',2X,'SPAN LOAD AT AERO 320
  2 SL COEF FROM',/' STATION',6X,' 2Y/B',9X,'S L COEF.',4X,'CL RATIO'AERO 330
  3,4X,'C RATIO',7X,'TO TWIST CL=',F9.5,3X,'AT CL=0',5X,'DESIRED CLAERC 340
  4 CHORD BD VOR'//)AERO 350
13 FORMAT (/ 47X, 'CONTRIBUTION OF THE SECOND PLANFORM TO SPAN LOAD DAERO 360
  1ISTRIBUTION' / )AERO 370
15 FORMAT(4X,14,F12.5,5X,3F12.5,3X,3F12.5,3X,2F12.5)AFRO 380
16 FORMAT (1H )AERO 390
18 FORMAT(///55X,21HTHIS CASE IS FINISHED)AERO 400
20 FORMAT(//5X,'DELTA CP TERMS FROM LE TIP TO TE TIP THEN INBOARD AERO 410
  1 ENDING WITH THE TE OF ROOT CHORD ')AERO 420
21 FORMAT ( /54X,'CMQ AND CLQ ARE COMPUTED'//)AERO 430
22 FORMAT(/38X,'STATIC LONGITUDINAL AERODYNAMIC COEFFICIENTS ARE COMPAERO 440
  1UTED'//)AERO 450
23 FORMAT ( /59X,'CLP IS COMPUTED'//)AERO 460
24 FORMAT(8F15.5)AERO 470
25 FORMAT (/20X,'X',11X,'X',11X,'Y',11X,'Z',12X,'S',5X,'C/4 SWEEP',4XAERO 480
  1 , 'DIHEDRAL',2X,'LOCAL ALPHA',2X,'DELTA CP AT DESIRED' / AERO 490
  2 19X,'C/4',9X,'3C/4',42X,'ANGLE',7X,'ANGLE',4X,'IN RADIANS',4X, AERO 500
  3 'CL =',F10.5 / )AERO 510
303 FORMAT(12X,9F12.5)AERO 520
1013 FORMAT(/47X,'CONTRIBUTION OF THE SECOND PLANFORM TO THE CHORD OR DAERO 530
  1RAG FORCE'//)AERO 540
1070 FORMAT (/// 30X, 'INDUCED DRAG, LEADING EDGE THRUST AND SUCTIONAERO 550
  1 COEFFICIENT CHARACTERISTICS',/ AERO 560
  2 34X,'COMPUTED AT ONE RADIAN ANGLE OF ATTACK FROM A NEAR FIELD SOLAERO 570
  3UTION' ,// AERO 580
  4 58X,'SECTION COEFFICIENTS',12X,'CONTRIBUTIONS TO TOTAL COEF.',/ AERO 590
  5 92X,'FROM EACH SPANWISE ROW',/ AERO 600
  6 38X,'L. E. SWEEP',/ AERO 610
  7 15X,'STATION',9X,' 2Y/B',5X,'ANGLE',5X,'CDII C/2B',5X,'CT C/2B', AERO 620
  8 5X,'CS C/2B',8X,'CDII',9X,'CT',10X,'CS'//)AERO 630
1071 FORMAT (10X ,110, 5X, 8F12.5)AERO 640
1072 FORMAT (/// 57X,'TOTAL COEFFICIENTS',// AERO 650
  1 36X,12HCDII/CL**2 = ,F10.5 ,5X,'CT=',F10.5,5X,'CS=',F10.5 ) AERO 660
4445 FORMAT(//////////56X,4HCLP=,F9.5////)AERO 670
4446 FORMAT(//////////42X,4HCMQ=,F9.5,10X,4HCLQ=,F9.5////)AERO 680
C AERO 690
C AERO 700
C AERO 710
C AERO 720
C AERO 730
C AERO 740
C AERO 750
C AERO 760
C AERO 770
C AERO 780
C AERO 790
C AERO 800
C AERO 810
PART 3 - COMPUTE OUTPUT TERMS
RAD = 57.29578
TWST = TWIST(1) + TWIST(2)
ALREF = 1

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C	THE TOLERANCE SET AT THIS POINT IN THE PROGRAM MAY NEED TO BE	AERO 820
C	CHANGED FOR COMPUTERS OTHER THAN THE CDC 6000 SERIES	AERO 830
C		AERO 840
C		AERO 850
	TOLC= .0100*BOT	AERO 860
	TOLCSQ = TOLC*TOLC	AERO 870
	QINF=1.	AERO 880
	NSSW=NSSWSV(1)+NSSWSV(2)	AERO 890
	IF(RTCDHT(1).NE.RTCDHT(2)) GO TO 794	AERO 900
	SUMPHI=0	AERO 910
	DO 801 J=1,NSSW	AERO 920
801	SUMPHI=SUMPHI+ABS(PHI(J))	AERO 930
	IF(SUMPHI.EQ.0.) GO TO 921	AERO 940
		AERO 950
C		AERO 960
C	PART 3 - SECTION 1	AERO 970
C	COMPUTE LIFT AND PITCHING MOMENT FOR WINGS WITH DIHEDRAL	AERO 980
C		AERO 990
C	GEOMETRY FOR TIP TRAILING LEGS	AERO1000
C		AERO1010
794	CPM(1) = 0	
	CPM(2) = 0	
	YCP(1) = 0	
	YCP(2) = 0	
	IM = 0	
	CLT = 0	
	CLNT = 0	
	NSSW1 = 0	AERO1020
	NSSW2 = NSSWSV(1)	
	NSSW3 = NSSWSV(1)	
	L=1	AERO1030
	NSCW = MSV(1) / NSSWSV(1)	AERO1040
	GO TO 798	AERO1050
796	NSSW1 = NSSWSV(1)	AERO1060
	NSSW2 = NSSW	
	NSSW3 = NSSWSV(2)	
	L=NSSWSV(1)+1	AERO1070
	NSCW = MSV(2) / NSSWSV(2)	AERO1080
798	I = IM + 1	AERO1090
	J = IM + 2	AERO1100
	IUU=2	AERO1110
	DIFFCR1=0.	AERO1120
	APHI=ATAN(PHI(I))	AERO1130
	TLX1=PN(I)-S(I)*TAN(PSI(I))	AERO1140
	TLX2=PN(J)-S(J)*TAN(PSI(J))	AERO1150
	CLFTLG=TLX1-TLX2	AERO1160
	XTLEG(1)=TLX1/2.+TLX2/2.	AERO1170
	YLEG=Q(I)-S(I)*COS(APHI)	AERO1180
	IF(NSSW1.EQ.0) YLEGSV(1)=YLEG	AERO1190
	ZLEG=ZH(I)-S(I)*SIN(APHI)	AERO1200
	IF(NSSW1.EQ.0) ZLEGSV(1)=ZLEG	AERO1210
	IF(NSSW1.EQ.NSSWSV(1)) GO TO 850	AERO1220
	GO TO 852	AERO1230
850	DO 5050 IT=1,L	AERO1240
	IF((ABS(YLEGSV(IT)-YLEG).LT.TOLC).AND.(ABS(ZLEGSV(IT)-ZLEG).LT.TOLA	AERO1250
	1C)) DIFFCR1=DIFCIRS(IT)	AERO1260
5050	CONTINUE	AERO1270
852	DO 802 NV=2,NSCW	AERO1280
	NVT=NV-1	AERO1290
802	XTLEG(NV)=XTLEG(NVT)-CLFTLG	AERO1300
	NCTL=0	
	NA =1	
	NB =NSCW	AERO1310
803	DO 823 NV=NA,NB	AERO1320
	VOU(NV,1)= 0	
	VOU(NV,2)= 0	
	UOU(NV,1)= 0	

	UOU(NV,2)= 0.	AERO1330
	DO 809 NN=1,M	AERO1340
	IZ=(NN-1)/NSCW+1	AERO1350
	APHI=ATAN(PHI(IZ))	AERO1360
	APSI=PSI(NN)	AERO1370
	XX=XTLEG(NV)-PN(NN)	AERO1380
	YY(1)=YLEG-Q(NN)	AERO1390
	YY(2)=YLEG+Q(NN)	AERO1400
	ZZ=ZLEG-ZH(IZ)	AERO1410
	SNN = S(NN)	AERO1420
C		AERO1430
	DO 822 I=1,2	AERO1440
	YYY = YY(I)	AERO1450
	CALL INFSUB (BOT,FU(I),FV(I),FW(I))	AERO1460
	APHI=-APHI	
	APSI=-APSI	AERO1470
822	CONTINUE	AERO1480
C		AERO1490
9001	DO 809 IXX=1,2	AERO1500
	UOU(NV, IXX)=UOU(NV, IXX)+((FU(1)+FU(2))*CIR(NN, IXX))/12.566371	AERO1510
809	VOU(NV, IXX)=VOU(NV, IXX)+((FV(1)+FV(2))*CIR(NN, IXX))/12.566371	AERO1520
823	CONTINUE	AERO1530
	NCTL=NCTL+1	AERO1540
	IF (NCTL-2) 810,811,812	AERO1550
C		AERO1560
C	GEOMETRY FOR SPANWISE BOUND VORTICES	AERO1570
C		AERO1580
810	NA=NSCW+1	AERO1590
	NB=2*NSCW	AERO1600
	JA=IM*NSCW+1	AERO1610
	YLEG=Q(JA)	AERO1620
	ZLEG=ZH(IM+1)	AERO1630
	DO 818 J=1,NSCW	AERO1640
	JK=IM*NSCW+J	AERO1650
	NV=J+NSCW	AERO1660
818	XTLEG(NV)=PN(JK)	AERO1670
	GO TO 803	AERO1680
C		AERO1690
C	GEOMETRY ALONG RIGHT TRAILING LEGS	AERO1700
C		AERO1710
811	NA=2*NSCW+1	AERO1720
	NB=3*NSCW	AERO1730
	DIFFCR2=0.	AERO1740
	JK=IM*NSCW+1	AERO1750
	APHI=ATAN(PHI(IM+1))	AERO1760
	YLEG=Q(JK)+S(JK)*COS(APHI)	AERO1770
	IF(NSSW1.EQ.0) YLEGSV(IUU)=YLEG	AERO1780
	ZLEG=ZH(IM+1)+S(JK)*SIN(APHI)	AERO1790
	IF(NSSW1.EQ.0) ZLEGSV(IUU)=ZLEG	AERO1800
	TLX1=PN(JK)+S(JK)*TAN(PSI(JK))	AERO1810
	JK=JK+1	AERO1820
	TLX2=PN(JK)+S(JK)*TAN(PSI(JK))	AERO1830
	CRTTLG=TLX1-TLX2	AERO1840
	XTLEG(NA)=TLX1/2.+TLX2/2.	AERO1850
	NAA=NA+1	AERO1860
	IF(NSSW1.EQ.NSSWSV(1)) GO TO 851	AERO1870
	GO TO 853	AERO1880
851	DO 5051 IT=1,L	AERO1890
	IF((ABS(YLEGSV(IT)-YLEG).LT.TOLC).AND.(ABS(ZLEGSV(IT)-ZLEG).LT.TOLA	AERO1900
	1C)) DIFFCR2=DIFCIRS(IT)	AERO1910
5051	CONTINUE	AERO1920
853	DO 819 NV=NAA,NB	AERO1930
	NVT=NV-1	AERO1940
819	XTLEG(NV)=XTLEG(NVT)-CRTTLG	AERO1950
	GO TO 803	AERO1960
C		AERO1970
C	COMPUTE LIFT AND PITCHING MOMENT FOR EACH ELEMENTAL PANEL	AERO1980

C		AERO1990
812	YY(1)=0	
	YY(2)=0	AERO2000
	IF (IM.NE.NSSW1) GO TO 834	AERO2010
	DO 835 IXX=1,2	AERO2020
	DIFCIR=DIFFCR1	AERO2030
	DO 835 NPOS=1,NSCW	AERO2040
	DIFCIR=DIFCIR+CIR(NPOS, IXX)	AERO2050
	CON=1.	AERO2060
	IF (NPOS.EQ.NSCW) CON=.75	AERO2070
	CHLFT(NPOS, IXX)=CLFTLG*CON*DIFCIR*VOU(NPOS, IXX)*(2./SREF)	AERO2080
	CLPT(NPOS, IXX)=CHLFT(NPOS, IXX)*(Q(NPOS)-S(NPOS))*2.	AERO2090
835	CONTINUE	AERO2100
	IF(NSSW1.EQ.0) DIFCIRS(1)=DIFCIR	AERO2110
834	DO 815 IXX=1,2	AERO2120
	DIFCIP=DIFFCR2	AERO2130
	DO 815 NPOS=1,NSCW	AERO2140
	JK=IM*NSCW+NPOS	AERO2150
	JL=(IM+1)*NSCW+NPOS	AERO2160
	JM=NSCW+NPOS	AERO2170
	JN=2*NSCW+NPOS	AERO2180
	IF (IM.EQ.(NSSW2-1)) GO TO 836	AERO2190
	DIFCIR=DIFCIR+CIR(JL, IXX)-CIR(JK, IXX)	AERO2200
836	CON=1.	AERO2210
	IF (NPOS.EQ.NSCW) CON=.75	AERO2220
	CHLFT(JL, IXX)=CRTLG*CON*DIFCIR*VOU(JN, IXX)*(2./SREF)	AERO2230
	CLCC(JK, IXX)=(2./SREF)*CIR(JK, IXX)*2.*S(JK)*COS(APHI)*(1.-VOU(JM,	AERO2240
	1 IXX)+VOU(JM, IXX)*TAN(PSI(JK)))	AERO2250
	CLPB(JK, IXX)=CLCC(JK, IXX)*Q(JK)*2.	AERO2260
	CLPT(JL, IXX)=CHLFT(JL, IXX)*(Q(JK)+S(JK))*2.	AERO2270
	YY(IXX)=YY(IXX)+(CLCC(JK, IXX)+CHLFT(JK, IXX))*2.	AERO2280
	CPM(IXX)=CPM(IXX)+(CLCC(JK, IXX)*XTLEG(JM)*BETA+CHLFT(JK, IXX)*XTLEGAERO2290	
	1(NPOS)*BETA)*2./CREF	AERO2300
	YCP(IXX)=YCP(IXX)+(CLCC(JK, IXX)*Q(JK)+CHLFT(JK, IXX)*(Q(JK)-S(JK)*	AERO2310
	1COS(APHI)))/BOT	AERO2320
815	CONTINUE	AERO2330
	IF(NSSW1.EQ.0) DIFCIRS(IUU)=DIFCIR	AERO2340
	CLT=CLT+YY(1)	AERO2350
	CLNT=CLNT+YY(2)	AERO2360
	IM=IM+1	AERO2370
	IF(NSSW1.EQ.0) IUU=IM+2	AERO2380
	IF(IM.EQ.NSSWSV(1)) CLWNGT=CLT	AERO2390
	IF(IM.EQ.NSSWSV(1)) CLWING=CLNT	AERO2400
	IF (IM.GE.NSSW2) GO TO 816	AERO2410
	NCTL=1	AERO2420
	DO 817 IXX=1,2	AERO2430
	DO 817 NV =1,NSCW	AERO2440
	NY=NV+2*NSCW	AERO2450
	XTLEG(NV)=XTLEG(NY)	AERO2460
817	VOU(NV, IXX)=VOU(NY, IXX)	AERO2470
	GO TO 810	AERO2480
C		AERO2490
C	SUM LIFT AND PITCHING MOMENT FOR ENTIRE WING	AERO2500
C		AERO2510
816	YY(1)=CLT*SREF/STRUE	AERO2520
	YY(2)=CLNT*SREF/STRUE	AERO2530
	NUP=NSSW3 + 1	AERO2540
	YTLEG(NUP)=0.	AERO2550
	XTLEG(NUP)=0	AERO2560
	IND=1	AERO2570
	IF (TWST .EQ.0.) IND=2	AERO2580
	DO 837 IXX=IND,2	AERO2590
	DO 820 JSSW=L,NSSW2	AERO2600
	SLOAD(IXX,JSSW)=0	AERO2610
	SLDT(JSSW)=0	AERO2620
	APHI=ATAN(PHI(JSSW))	AERO2630
	JL=(JSSW-1)*NSCW+1	AERO2640

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      K=JSSW-L+1
820 YTLEG( K )=Q(JL)-S(JL)*COS(APHI)
      DO 837 INC=1,NSCW
      DO 838 JNS=L,NSSW2
      JK=(JNS-1)*NSCW+INC
      K=JNS-L+1
838 XTLEG( K )=CHLFT(JK,IXX)
      DO 837 INS=L,NSSW2
      JK=(INS-1)*NSCW+INC
      APHI=ATAN(PHI(INS))
      CALL FTLUP(Q(JK),CHTLF,+1,NUP,YTLEG,XTLEG)
      T=SREF/(2.*S(JK)*COS(APHI)*CAVE)
      SLDT(INS)=SLDT(INS)+CHTLF*T
      CLCC(JK,IXX)=(CLCC(JK,IXX)+CHTLF)*T
837 SLOAD(IXX,INS)=SLOAD(IXX,INS)+CLCC(JK,IXX)
      IF(IM.NE.NSSW) GO TO 796
      CLA(2)=CLNT/ALREF
      CMCL=CPM(2)/CLNT
      CMO=CPM(1)-CMCL*CLT
      YCP(2)=YCP(2)/(CLNT/2.)
      DO 840 I=1,NSSW
      SLDT(I)=SLDT(I)/YY(2)
      IF(TWST.EQ.0.) SLOAD(1,I)=0.
      IF(TWST.NE.0.) SLOAD(1,I)=SLOAD(1,I)/YY(1)
840 SLOAD(2,I)=SLOAD(2,I)/YY(2)
      CRL=0.
      DO 860 IAM=1,M
860 CRL=CRL+CLPB(IAM,2)+CLPT(IAM,2)
      CLP=CRL/(.08725*2.*BOT)
      GO TO 903

C                                     PART 3 - SECTION 2
C          COMPUTE LIFT AND PITCHING MOMENT FOR WINGS WITHOUT DIHEDRAL
C
921 DO 901 NV=1,2
      SUM(NV)=0
      DO 901 I=1,M
      SUM(NV)=SUM(NV)+CIR(I,NV)*S(I)
      IF(NV.EQ.1.AND.I.EQ.MSV(1)) CLWNGT = SUM(1)*8. / SREF
      IF(NV.EQ.2.AND.I.EQ.MSV(1)) CLWING = SUM(2)*8. / SREF
901 CONTINUE
      CLT = 8.*SUM(1)/SREF
      CLNT = 8.*SUM(2)/SREF
      IF(KBOT.EQ.1) GO TO 800
      CLWNGT = CLT - CLWNGT
      CLWING = CLNT - CLWING
800 CRL = 0.
      DO 905 I=1,M
      CRL=CRL+(Q(I)*CIR(I,2)*2.*S(I))*2.
      CLCC(I,1)=CIR(I,1)*2./CAVE
905 CLCC(I,2)=CIR(I,2)*2./CAVE

C          COMPUTE CLP
C
      CLP= CRL/(SREF*BOT*.08725)
      CLA(2)=CLNT
      DO 922 IXX=1,2
      SA = 0
      SB = 0
      SC = 0.
      I = 0
      DO 920 JSSW=1,NSSW
      SLDT(JSSW)=0
      SLOAD(IXX,JSSW)=0
      NSCW = TBLSCW(JSSW)
      DO 920 JSCW=1,NSCW
      IF(TWST.EQ.0..AND.IXX.EQ.1) GO TO 930

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I      = I + 1
SA=SA+CIR(I, IXX)*S(I)
SB=SB+CIR(I, IXX)*Q(I)*S(I)
SC=SC+CIR(I, IXX)*PN(I)*S(I)*BETA
SLOAD(IXX, JSSW) = SLOAD(IXX, JSSW)+(BOT*CIR(I, IXX))/(2.*SUM(IXX))
GO TO 920
930 SLOAD(1, JSSW)=0.
920 CONTINUE
IF(TWST.EQ.0..AND.IXX.EQ.1) GO TO 932
YCP(IXX)=SB/(SA*BOT)
AC(IXX)=SC/(SA*CREF)
GO TO 922
932 YCP(1)=0
AC(1)=0.
922 CONTINUE
CMCL=AC(2)
CMO=(AC(1)-AC(2))*CLT

C
C      PART 3 - SECTION 3
C      COMPUTE AND PRINT FINAL OUTPUT DATA FOR ALL WINGS
C
903 DO 902 IXX=1,2
JN      = 0
DO 902 JSSW=1, NSSW
CH      (IXX, JSSW)=0
NSCW    = TBLSCW(JSSW)
DO 904 JSCW=1, NSCW
JN      = JN + 1
CH      (IXX, JSSW)=(-2.0)*(PV(JN)-PN(JN))*BETA+CH      (IXX, JSSW)
904 CONTINUE
CCAV(IXX, JSSW)=CH(IXX, JSSW)/CAVE
CLCL(IXX, JSSW)=SLOAD(IXX, JSSW)/CCAV(IXX, JSSW)
902 CONTINUE
CLD=CLDES
IF(CLDES.EQ.11) CLD=1.
DO 1020 I=1, M
CP(I)    = (CLCC(I, 1)+CLCC(I, 2))*(CLD -CLT)/CLNT)*CAVE/(2.*(PN(I)-
1      PV(I)) * BETA )
1020 CONTINUE
WRITE(29,4) CONFIG
IF ( PTEST.NE.0. ) WRITE(29,23)
IF ( QTEST.NE.0. ) WRITE(29,21)
IF ( PTEST.EQ.0. .AND. QTEST.EQ.0. ) WRITE(29,22)
WRITE(29,25) CLD
HEAD = ' DESIRED'
IF (CLDES.EQ.11. ) HEAD = ' '
IEND = 11
IF(CLDES.NE.11.) IEND=1
DO 5000 IUTK=1, IEND
IF(IEND.EQ.11) CLDES=(FLOAT(IUTK)-1.)/10.
IF(CLDES.EQ.0.) CLDES=-.1
NR      = 0
DO 3006 NV=1, NSSW
NSCW    = TBLSCW(NV)
NP      = NR + 1
NR      = NR + NSCW
PHIPR   = ATAN(PHI(NV)) * RAD
SLOAD(3, NV)=0.
IF (NV.EQ.(NSSWSV(1)+1).AND.IUTK.EQ.1) WRITE(29,1)
DO 3006 I=NP, NR
IF ( IUTK.GT.1 ) GO TO 3006
PNPR    = PN(I) * BETA
PVPR    = PV(I) * BETA
PSIPR   = PSI(I)* RAD
WRITE(29,303) PNPR, PVPR, Q(I), ZH(NV), S(I), PSIPR, PHIPR, ALP(I), CP(I)
3006 SLOAD(3, NV)=SLOAD(3, NV)+CLCC(I, 2)*CLDES/CLNT+CLCC(I, 1)-CLCC(I, 2)*CAERO3940
1LT/CLNT
AERO3950

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IF(IUTK.GT.1) GO TO 3007	AERO3960
WRITE(29,7)	AERO3970
WRITE(29,8) CREF,CAVE,STRUE,SREF, BOT,AR,ARTRUE,MACH	AERO3980
3007 CONTINUE	AERO3990
C	AERO4000
C	AERO4010
IF(PTEST.NE.0.)WRITE(29,4445) CLP	AERO4020
IF(PTEST.NE.0.) GO TO 4444	AERO4030
C	AERO4040
C	AERO4050
C	AERO4060
CMQ=2.0*CMCL*CLNT/(0.08725*CREF)	AERO4070
CLQ=2.0*CLNT/(0.08725*CREF)	AERO4080
IF(QTEST.NE.0.) WRITE(29,4446) CMQ,CLQ	AERO4090
IF(QTEST.NE.0.) GO TO 4444	AERO4100
C	AERO4110
C	AERO4120
C	AERO4130
NSV=NSSWSV(1)+1	AERO4140
MTOT=MSV(1)+1	AERO4150
IF(KBOT.EQ.1) GO TO 1001	AERO4160
NSV=NSV+NSSWSV(2)	AERO4170
MTOT=MTOT+MSV(2)	AERO4180
1001 CALL CDICLS (AR,ARTRUE,NSSWSV(KBOT),MTOT,NSV,CDI,CDIT)	AERO4190
CLAPD=CLA(2)/57.29578	AERO4200
ALPO=-(CLT/CLA(2))*57.29578	AERO4210
ALPD=CLDES/CLAPD+ALPO	AERO4220
ALPW=1./CLAPD	AERO4230
CLWB=CLWING*ALPD/57.29578+CLWNGT	AERO4240
CDIWB = CDI / (CLWB*CLWB)	AERO4250
IF (IUTK.EQ.1) WRITE (29,5) HEAD,CDIT	AERO4260
5000 WRITE (29,6) CLDES,ALPD,CLWB,CDI,CDIWB	AERO4270
WRITE(29,11) CLA(2),CLAPD,CLT,ALPO,YCP(2),CMCL,CMO	AERO4280
WRITE(29,12) CLT	AERO4290
NR = 0	AERO4300
J = 0	AERO4310
DO 1004 NV=1,NSSW	
BCLCC = 0	
BADLAE= 0	
BASLD = 0.	AERO4320
NSCW = TBLSCW(NV)	AERO4330
NP = NR + 1	AERO4340
NR = NR + NSCW	AERO4350
DO 1002 I=NP,NR	AERO4360
ADLAE=CLCC(I,2)*CLT/CLNT	AERO4370
BSLD=CLCC(I,1)-ADLAE	AERO4380
BCLCC=BCLCC+CLCC(I,1)	AERO4390
BADLAE=BADLAE+ADLAE	AERO4400
BASLD=BASLD+BSLD	AERO4410
1002 CONTINUE	AERO4420
J = J + NSCW	AERO4430
YQ = Q(J) / BOT	AERO4440
IF (NV.EQ.(NSSWSV(1)+1)) WRITE(29,13)	AERO4450
1004 WRITE(29,15) NV,YQ,SLOAD(2,NV),CLCL(2,NV),CCAV(2,NV),BCLCC,BADLAE,	AERO4460
1 BASLD,SLOAD(3,NV),SLDT(NV)	AERO4470
WRITE (29,1070)	AERO4480
CTHRUST = 0	
CSUCT = 0	
CDRAG = 0.	AERO4490
NN=1	AERO4500
DO 1050 NV=1,NSSW	AERO4510
SSCTRST = SECTRST(NV) / (4.*BOT)	AERO4520
SSCDRAG = SLOAD(2,NV) * CAVE * SREF * CLA(2) / (STRUE * 4. * BOT)	AERO4530
1 - SSCTRST	AERO4540
CSSWWA = COS (ATAN (SSWWA(NV)))	AERO4550
SSCSUCT = SSCTRST / CSSWWA	AERO4560
IF (NV.EQ.1) GO TO 1060	AERO4570

	NN = NN + TBLSCW(NV-1)	AERO4580
1060	PHIPR = ATAN (PHI (NV))	AERO4590
	CDRAGS = SSCDRAG*4.*BOT*2.*S (NN) *COS (PHIPR) /SREF	AERO4600
	CDRAG = CDRAG + 2.0 * CDRAGS	AERO4610
	CTHRUSS = SECTRST (NV) *2.*S (NN) *COS (PHIPR) / SREF	AERO4620
	CTHRUST = CTHRUST + 2.0 * CTHRUST	AERO4630
	CSUCTS = CTHRUST / CSSWWA	AERO4640
	CSUCT = CSUCT + 2.0 * CSUCTS	AERO4650
	SWALE = ATAN (SSWWA (NV)) * RAD	AERO4660
	YQ = Q (NN) / BOT	AERO4670
	IF (NV.EQ. (NSSWSV (1)+1)) WRITE (29,1013)	AERO4680
1050	WRITE (29,1071) NV, YQ, SWALE, SSCDRAG, SSCTRST, SSCSUCT, CDRAGS, CTHRUST,	AERO4690
	1 CSUCTS	AERO4700
	CDRAGP = CDRAG / (CLA (2) *CLA (2))	AERO4710
	WRITE (29,1072) CDRAGP, CTHRUST, CSUCT	AERO4720
4444	WRITE (29,18)	AERO4730
	WRITE (29,16)	AERO4740
	RETURN	AERO4750
	END	AERO4760
	SUBROUTINE CDICLS (AR, ARTRUE, ISEMS, MTOT, NSV, CDI, CDIT)	CDIC 10
	DIMENSION ETAN (51), GAMPR (51,1), ETA (41), GAMMA (41), VE (41), B (41),	CDIC 20
	1 FVN (41,41)	CDIC 30
	COMMON/ALL/ BOT, M, BETA, PTEST, QTEST, TBLSCW (50), Q (300), PN (300),	CDIC 40
	1 PV (300), ALP (300), S (300), PSI (300), PHI (300), ZH (50)	CDIC 50
	COMMON/THRECDI/SLOAD (3,50)	CDIC 60
	CHARACTER*20 CASEFN	
	DO 15 I=1,41	CDIC 70
	DO 15 J=1,41	CDIC 80
15	FVN (I,J)=0	CDIC 90
	SPAN=2.*BOT	CDIC 100
	CAVB=SPAN/ARTRUE	CDIC 110
	PI=.314159265E+01	CDIC 120
	NST=ISEMS+1	CDIC 130
	NN=MTOT	CDIC 140
	DO 101 N=1, ISEMS	CDIC 150
	NM=NSV - N	CDIC 160
	NSCW=TBLSCW (NM)	CDIC 170
	NN=NN-NSCW	CDIC 180
	ETAN (N)=ASIN (-Q (NN) *2./SPAN)	CDIC 190
	GAMPR (N,1)=SLOAD (3,NM) *CAVB/ (2.*SPAN)	CDIC 200
101	CONTINUE	CDIC 210
	ETAN (NST)= PI/2.	CDIC 220
	GAMPR (NST,1)=0	CDIC 230
	DO 7 NP= 1,41	CDIC 240
	ANP=NP	CDIC 250
	7 ETA (NP)= (ANP-21.)*PI/42.	CDIC 260
C		CDIC 270
	DO 102 JK=21,41	CDIC 280
	CALL FTLUP (ETA (JK), GAMMA (JK), 1, NST, ETAN, GAMPR)	CDIC 290
102	CONTINUE	CDIC 300
	DO 600 NY=22,41	CDIC 310
	ETA (NY)=SIN (ETA (NY))	CDIC 320
	NR=42-NY	CDIC 330
	ETA (NR)=-ETA (NY)	CDIC 340
600	GAMMA (NR)=GAMMA (NY)	CDIC 350
	DO 589 NU=21,41	CDIC 360
	ANU=NU	CDIC 370
	DO 14 N=1,41	CDIC 380
	AN=N	CDIC 390
	NNUD=IABS (N-NU)	CDIC 400
	VE (N)=COS ((AN-21.)*PI)/42.)	CDIC 410
	IF (NNUD.NE.0) GO TO 9	CDIC 420
	B (N)=(42.)/(4.0*COS ((ANU-21.)*PI)/42.))	CDIC 430
	GO TO 14	CDIC 440
9	IF (MOD (NNUD,2).EQ.0) GO TO 12	CDIC 450
	B (N)=VE (N) / ((42.)* (ETA (N)-ETA (NU)) **2)	CDIC 460
	GO TO 14	CDIC 470

12 B(N)=0.0	CDIC 480
14 CONTINUE	CDIC 490
DO 589 NP=21,41	CDIC 500
NUST =IABS (NU-21)	CDIC 510
IF (NUST.EQ.0) GO TO 589	CDIC 520
IF (MOD (NUST,2) .EQ.0) GO TO 589	CDIC 530
NPST=IABS (NP-20)	CDIC 540
IF (MOD (NPST,2) .EQ.0) GO TO 589	CDIC 550
NPNUD=IABS (NP-NU)	CDIC 560
IF (NPNUD.EQ.0) GO TO 589	CDIC 570
IF (MOD (NPNUD,2) .EQ.0) GO TO 589	CDIC 580
FVN (NU,NP)=2.0*B (NP) /21.*COS ((ANU-21.) *PI/42.)	CDIC 590
IT=42-NU	CDIC 600
ITT=42-NP	CDIC 610
FVN (NU,ITT)=2.0*B (ITT) /21.*COS ((ANU-21.) *PI/42.)	CDIC 620
FVN (IT,NP)=FVN (NU,ITT)	CDIC 630
FVN (IT,ITT)=FVN (NU,NP)	CDIC 640
589 CONTINUE	CDIC 650
C	CDIC 660
CCC=0.0	CDIC 670
DO 10 N=1,41	CDIC 680
10 CCC=CCC+(GAMMA(N)*GAMMA(N))	CDIC 690
CCD=0.0	CDIC 700
DO 11 NUP=1,41	CDIC 710
DO 11 N=1,41	CDIC 720
CCD=CCD-2.0*FVN (NUP,N) * (GAMMA (NUP) *GAMMA (N))	CDIC 730
11 CONTINUE	CDIC 740
CDI=PI*AR/4.*(CCC+CCD)	CDIC 750
CDIT=1./(PI*AR)	CDIC 760
RETURN	CDIC 770
END	CDIC 780
SUBROUTINE FTLUP (X,Y,M,N,VARI,VARD)	TLUP 10
C*****DOCUMENT DATE 09-12-69 SUBROUTINE REVISED 07-07-69 *****	TLUP 20
C* MODIFICATION OF LIBRARY INTERPOLATION SUBROUTINE FTLUP	TLUP 30
DIMENSION VARI (1),VARD (1),V (3),YY (2)	TLUP 40
DIMENSION II (43)	TLUP 50
C*	TLUP 60
C* INITIALIZE ALL INTERVAL POINTERS TO -1.0 FOR MONOTONICITY CHECK	TLUP 70
DATA (II (J),J=1,43)/43*-1/	TLUP 80
MA=IABS (M)	TLUP 90
C*	TLUP 100
C* ASSIGN INTERVAL POINTER FOR GIVEN VARI TABLE	TLUP 110
C* THE SAME POINTER WILL BE USED ON A GIVEN VARI TABLE EVERY TIME	TLUP 120
LI = 1	TLUP 130
I=II (LI)	TLUP 140
IF (I.GE.0) GO TO 10	TLUP 150
IF (N.LT.2) GO TO 10	TLUP 160
C*	TLUP 170
C*MONOTONICITY CHECK	TLUP 180
IF (VARI (2)-VARI (1)) 1,1,3	TLUP 190
C* ERROR IN MONOTONICITY	TLUP 200
2 K = 1	TLUP 210
WRITE (29,102) J,K, (VARI (J),J=1,N), (VARD (J),J=1,N)	TLUP 220
102 FORMAT (1H1,' TABLE BELOW OUT OF ORDER FOR FTLUP AT POSITION '	TLUP 230
1,IS,/' X TABLE IS STORED IN LOCATION ',06,/(8G15.8))	TLUP 240
STOP	TLUP 250
C* MONOTONIC DECREASING	TLUP 260
1 DO 5 J=2,N	TLUP 270
IF (VARI (J)-VARI (J-1)) 5,2,2	TLUP 280
5 CONTINUE	TLUP 290
GO TO 10	TLUP 300
C* MONOTONIC INCREASING	TLUP 310
3 DO 6 J=2,N	TLUP 320
IF (VARI (J)-VARI (J-1)) 2,2,6	TLUP 330
6 CONTINUE	TLUP 340
C*	TLUP 350
C*INTERPOLATION	TLUP 360

10 IF (I.LE.0) I=1	TLUP 370
IF (I.GE.N) I=N-1	TLUP 380
IF (N.LE.1) GO TO 8	TLUP 390
IF (MA.NE.0) GO TO 99	TLUP 400
C* ZERO ORDFR	TLUP 410
8 Y=VARD(1)	TLUP 420
GO TO 800	TLUP 430
C* LOCATE I INTERVAL (X(I).LE.X.LT.X(I+1))	TLUP 440
99 IF ((VARI(I)-X)*(VARI(I+1)-X)) 61,61,40	TLUP 450
C* IN GIVES DIRECTION FOR SEARCH OF INTERVALS	TLUP 460
40 IN=SIGN(1.0,(VARI(I+1)-VARI(I))*(X-VARI(I)))	TLUP 470
C* IF X OUTSIDE ENDPOINTS, EXTRAPOLATE FROM END INTERVAL	TLUP 480
41 IF ((I+IN).LE.0) GO TO 61	TLUP 490
IF ((I+IN).GE.N) GO TO 61	TLUP 500
I=I+IN	TLUP 510
IF ((VARI(I)-X)*(VARI(I+1)-X)) 61,61,41	TLUP 520
61 IF (MA.EQ.2) GO TO 200	TLUP 530
C*	TLUP 540
C*FIRST ORDER	TLUP 550
Y=(VARD(I)*(VARI(I+1)-X)-VARD(I+1)*(VARI(I)-X))/(VARI(I+1)-VARI(I))	TLUP 560
1)	TLUP 570
GO TO 800	TLUP 580
C*	TLUP 590
C*SECOND ORDER	TLUP 600
200 IF (N.EQ.2) GO TO 2	TLUP 610
IF (I.EQ.(N-1)) GO TO 209	TLUP 620
IF (I.EQ.1) GO TO 201	TLUP 630
C* PICK THIRD POINT	TLUP 640
SK= VARI(I+1)-VARI(I)	TLUP 650
IF ((SK*(X-VARI(I-1))).LT.(SK*(VARI(I+2)-X))) GO TO 209	TLUP 660
201 L=I	TLUP 670
GO TO 702	TLUP 680
209 L=I-1	TLUP 690
702 V(1)=VARI(L)-X	TLUP 700
V(2)=VARI(L+1)-X	TLUP 710
V(3)=VARI(L+2)-X	TLUP 720
YY(1)=(VARD(L)*V(2)-VARD(L+1)*V(1))/(VARI(L+1)-VARI(L))	TLUP 730
YY(2)=(VARD(L+1)*V(3)-VARD(L+2)*V(2))/(VARI(L+2)-VARI(L+1))	TLUP 740
Y=(YY(1)*V(3)-YY(2)*V(1))/(VARI(L+2)-VARI(L))	TLUP 750
800 II(LI)=I	TLUP 760
RETURN	TLUP 770
END	

REFERENCES

1. Margason, Richard J. and John E. Lamar, *Vortex-Lattice FORTRAN Program for Estimating Subsonic Aerodynamic Characteristics of Complex Planforms*, NASA TN D-6142, Feb. 1971.
2. Anderson, John D., Jr., *Fundamentals of Aerodynamics*, Second ed., McGraw-Hill, Inc., New York, 1991.
3. Bertin, John J. and Michael L. Smith, *Aerodynamics for Engineers*, Second ed., Prentice Hall, Englewood Cliffs, New Jersey, 1989.
4. Rom, Joseph, *High Angle of Attack Aerodynamics*, Springer-Verlag, New York, 1992.
5. Houghton, E. L. and N. B. Carruthers, *Aerodynamics for Engineering Students*, Third ed., Edward Arnold, New York, 1982.
6. Moran, Jack, *An Introduction to Theoretical and Computational Aerodynamics*, John Wiley & Sons, New York, 1984.
7. Layton, Donald, *Aircraft Performance*, Matrix Publishers, Inc., Chesterland, Ohio, 1988.
8. Nelson, Robert C., *Flight Stability and Automatic Control*, McGraw-Hill, Inc., New York, 1989.
9. Abbott, Ira H. and Albert E. Von Doenhoff, *Theory of Wing Sections*, Dover Publications, Inc., New York, 1959.
10. Glauert, H., *The Elements of Aerfoil and Airscrew Theory*, Second ed., Cambridge University Press, 1948.
11. Campbell, George S., *A Finite-Step Method for the Calculation of Span Loadings of Unusual Plan Forms*, NACA RM L50L13, 1951.
12. Blackwell, James A., Jr., *A Finite-Step Method for Calculation of Theoretical Load Distributions for Arbitrary Lifting-Surface Arrangements at Subsonic Speeds*, NASA TN D-5335, 1969.
13. Rubbert, Paul E., *Theoretical Characteristics of Arbitrary Wings by Non-Planar Vortex Lattice Method*, Doc. No. D6-9244, Boeing Co., Feb. 1964.
14. Multhopp, H., *Methods for Calculating the Lift Distribution of Wings (Subsonic Lifting-Surface Theory)*, R. & M. No. 2884, Brit. A. R. C., Jan. 1950.

15. Etter, D. M., *Structured FORTRAN 77 for Engineers and Scientists*, Third ed., The Benjamin / Cummings Publishing Company, Inc., Redwood City, California, 1990.
16. Anderson, J. D., Jr., *Introduction to Flight*, McGraw-Hill, Inc., New York, 1989.
17. Anderson, Dale A., John C. Tannehill, and Richard H. Pletcher, *Computational Fluid Mechanics and Heat Transfer*, Hemisphere Publishing Corp., New York, 1984.
18. White, Frank M., *Fluid Mechanics*, Second ed., McGraw-Hill Book Company, New York, 1986.
19. Panton, Ronald L., *Incompressible Flow*, John Wiley & Sons, New York, 1984.
20. Anderson, John D., Jr., *Modern Compressible Flow*, Second ed., McGraw-Hill, Inc., New York, 1990.
21. Kreyszig, Erwin, *Advanced Engineering Mathematics*, Sixth ed., John Wiley & Sons, New York, 1988.
22. Swokowski, Earl W., *Calculus With Analytic Geometry*, Alternate ed., Prindle, Weber & Schmidt, Boston, Massachusetts, 1983.
23. Zucrow, Maurice J. and Joe D. Hoffman, *Gas Dynamics*, Vol. I, John Wiley & Sons, New York, 1976.
24. Moran, Michael J. and Howard N. Shapiro, *Fundamental of Engineering Thermodynamics*, John Wiley & Sons, New York, 1988.
25. Incropera, Frank P. and David P. DeWitt, *Fundamentals of Heat and Mass Transfer*, Third ed., John Wiley & Sons, New York, 1990.
26. Hornbeck, Robert W., *Numerical Methods*, Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1975.
27. Burden, Richard L. and J. Douglas Faires, *Numerical Analysis*, Fifth ed., PWS Publishing Company, Boston, Massachusetts, 1993.
28. Kirsznner, Laurie G. and Stephen R. Mandell, *Writing A College Rhetoric*, Second ed., Holt, Rinehart and Winston, Inc., New York, 1988.
29. Ogata, Katsuhiko, *Modern Control Engineering*, Second ed., Prentice Hall, Inc., Englewood Cliffs, New Jersey, 1990.

30. D'Azzo, John. J. and Constantine H. Houpis, *Linear Control System Analysis and Design*. Fourth ed., McGraw-Hill, Inc., New York, 1995.
31. Rosenberg, Ronald C. and Dean C. Karnopp, *Introduction To Physical System Dynamics*. McGraw-Hill, Inc., New York, 1983.