A COMPUTER MODULE USED TO CALCULATE THE HORIZONTAL CONTROL SURFACE SIZE OF A CONCEPTUAL AIRCRAFT DESIGN

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ABSTRACT

A COMPUTER MODULE USED TO CALCULATE THE HORIZONTAL CONTROL SURFACE SIZE OF A CONCEPTUAL AIRCRAFT DESIGN

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This paper discusses the creation of a computer module used to calculate the size of the horizontal control surfaces of a conceptual aircraft design. The control surface size is determined by first calculating the size needed to rotate the aircraft during takeoff, and, second, by determining if the calculated size is large enough to maintain stability of the aircraft throughout any specified mission. The tail size needed to rotate during takeoff is calculated from a summation of forces about the main landing gear of the aircraft. The stability of the aircraft is determined from a summation of forces about the center of gravity during different phases of the aircraft's flight. Included in the horizontal control surface analysis are: downwash effects on an aft tail, upwash effects on a forward canard, and effects due to flight in close proximity to the ground.

Comparisons of production aircraft with numerical models show good accuracy for control surface sizing. A modified canard design verified the accuracy of the module for canard configurations.

Added to this stability and control module is a subroutine that determines one of three design variables for a stable vectored thrust aircraft. These include forward thrust nozzle position, aft thrust nozzle angle, and forward thrust split.
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a - Height of aft horizontal tail above trailing vortex core (feet)
a - Wing lift-curve slope (per degree)
a_{cn} - Canard lift-curve slope (per degree)
a_{ht} - Tail lift-curve slope (per degree)
A - Position of wing quarter-chord at fuselage centerline (feet)
AR - Aspect Ratio
AR_{eff} - Effective aspect ratio in ground effects
AC - Aerodynamic Center
ACSYNT - AirCraft SYNThesis program
B - Angle between C.G. and main landing gear (degrees)
b - Wing span (feet)
b_{CN} - Canard span (feet)
BDMAX - Maximum fuselage width (feet)
b_{eff} - Effective wing span in ground effect (feet)
b_{HT} - Aft horizontal tail span (feet)
BODL - Fuselage length (feet)
BR - Balance Ratio, BR = ((c_l/ c_l)^2 - (t/2 c_l)^2)^{1/2}
b_{v} - Wing trailing edge vortex span (feet)
b_{1} - Change in hinge moment with respect to angle of attack
b_{2} - Change in hinge moment with respect to change in elevator angle
c - Wing Mean Aerodynamic Chord (feet)
c/4 - Wing quarter-chord point of the Mean Aerodynamic Chord
C_{root} - Wing root chord (feet)
c_B - Elevator balance chord (feet)
c_e - Elevator mean aerodynamic chord (feet)
c_f - Elevator flap chord (feet)
C.G. - Center of Gravity
C_l - 2-dimensional lift coefficient of the wing
C_L - 3-dimensional lift coefficient of the wing
C_{l\alpha} - 2-dimensional wing lift-curve slope
C_{L\alpha} - 3-dimensional wing lift-curve slope
C_{LaHT} - 3-dimensional tail lift-curve slope
C_{LaCN} - 3-dimensional canard lift-curve slope
C.M_{\alpha} - Pitching moment coefficient change due to change in angle of attack
C.M_A - Moment coefficient about point A
C.M_{cg} - Moment coefficient about the Center of Gravity
C.M_{mg} - Moment coefficient about the main landing gear
C.M_{TH} - Contribution of thrust to the pitching moment
C.M_{wb} - Wing-Body pitching moment
D - Aircraft drag
dC_{mT}/dC_l - Total pitching moment-curve slope
dC_{mT}/dC_{lCN} - Pitching moment-curve slope contribution of the canard
\( \frac{dC_m}{dC_l} f \) - Pitching moment-curve slope contribution of the fuselage

\( \frac{dC_m}{dC_l} \text{ tail} \) - Pitching moment-curve slope contribution of the tail

\( \frac{dC_m}{dC_l} \text{ TH} \) - Pitching moment-curve slope contribution of the thrust of the engine(s)

\( \frac{dC_m}{dC_l} \text{ wing} \) - Pitching moment-curve slope contribution of the wing

\( \Delta \varepsilon/\Delta \alpha \) - Change in downwash angle with respect to change in angle of attack

\( \Delta \varepsilon/\Delta \alpha_{\text{up}} \) - Change in upwash angle with respect to change in angle of attack

\( \Delta \delta/\Delta \alpha \) - Change in elevator angle due to change in angle of attack

\( d_g \) - Height of control surface root quarter-chord above the ground

\( dL \) - Incremental lift force associated with a slice of wing section

\( dy \) - Incremental distance along wing Mean Aerodynamic Chord

\( e \) - Oswald efficiency factor

\( F_e \) - Free elevator factor

\( H \) - Height of wing above the ground

\( H_{\text{HT}} \) - Height of aft horizontal tail above ground

\( h_{\text{HT}} \) - Height of aft horizontal tail above wing chord plane

\( i_{\text{CN}} \) - Canard incidence angle (degrees)

\( i_{\text{HT}} \) - Tail incidence angle (degrees)

\( i_w \) - Wing incidence angle (degrees)

\( K_f \) - Fuselage coefficient

\( L_{\text{CN}} \) - Canard lift (pounds force)

\( L_{\text{eff}} \) - Effective length from wing to horizontal tail (feet)

\( L_{\text{HT}} \) - Tail lift (pounds force)

\( L_w \) - Wing Body lift (pounds force)
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<td>$\delta_e$</td>
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<td>$\varepsilon_g$</td>
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<td>$\varepsilon_v$</td>
<td>Downwash angle in the wing vortex core (degrees)</td>
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$e_{up}$ - Upwash angle of wing on forward canard (degrees)

$\Lambda$ - Wing sweep angle (degrees)

$\gamma_1$ - Forward thrust vector angle (degrees)

$\gamma_2$ - Aft thrust vector angle (degrees)

$\lambda$ - Wing taper ratio

$\eta_{CN}$ - Canard efficiency

$\eta_{HT}$ - Tail efficiency

$\Gamma$ - Wing dihedral angle (degrees)

$\tau$ - Control effectiveness factor
CHAPTER 1

Introduction

In the aircraft conceptual design process, there are five major areas in which the designer allots most of his or her time and effort. These include aircraft layout, aerodynamics, weights, propulsion, and performance. In each of these areas the designer goes through a design process that includes a large and complex series of decisions and calculations to determine the design parameters of the aircraft. After initial parameters have been determined, the design is compared to any specified requirements, appropriate changes are made, and then another series of decisions and calculations is completed to refine the design. This cycle is repeated until the aircraft design created meets the specified requirements.

Once the conceptual design has become refined, the designer turns to more detailed areas in the aircraft design process. One of these areas is that of stability and control, where the sizes of the control surfaces on the aircraft are determined, the stability of the aircraft is determined, and where a more refined estimate of aircraft components weight and position is completed. Once again, as in the conceptual design areas, the detailed design areas involve a cycle of calculations and decisions in order to create a more refined aircraft design.

NASA Ames Research Center has created a computer program which does the calculation part of the conceptual design series. This design program, called ACSYNT (AirCraft SYNThesis design program), helps the designer in the five areas of the conceptual design. This includes geometric layout, aerodynamic analysis, weight calculations, propulsion estimates, and performance analysis. This program allows the designer to concentrate more on the decisions and less on the complex calculations.
To enhance the conceptual design process of the ACSYNT design program, the decision was made to create a module to perform calculations necessary for a longitudinal stability and control analysis. By performing more analysis of the aircraft's design at an earlier stage, a more refined product is created during the conceptual design, reducing the amount of effort required to finalize the aircraft design.

This paper discusses the development of the stability and control module. This module will be used in conjunction with the ACSYNT design program to enhance the conceptual design process. The module is used to determine the aircraft's center of gravity by positioning the different components of the aircraft at positions specified by the designer. It is used to determine the shift in center of gravity due to fuel and weapons usage during the aircraft's mission. Finally, it is used to calculate the horizontal control surface size needed to maintain controllability during take off, mission completion, and landing. The module is used to determine the horizontal control surface size of an aircraft which uses either a conventional aft tail, a forward canard, or both.

In order to determine the stability of the aircraft, there are several aerodynamic parameters that the module must determine. These include the shift in the aerodynamic center of the wing with respect to Mach number, and the lift curve slope of the canard if one is being used (lift curve slopes for the wing and aft mounted tail are calculated using ACSYNT's aerodynamics module). Also, the downwash of the wing on the tail, the upwash of the wing on the canard, and ground effects are determined using the module.

Also included in this module is an analysis that uses the stability conditions during landing to calculate three vectored thrust design parameters. The analysis includes the position of the forward thrust vector, the angle of the aft thrust vector, and the amount of thrust split between the forward thrust vector and the total thrust. This allows the designer to create a vertical landing aircraft design which is stable during transition from forward flight to hover.
CHAPTER 2
Center of Gravity

One of the more important parameters used in stability calculations is the center of gravity (C.G.). The C.G. is the equilibrium point where the weights of the different components act as one force at one point. It is a reference point in the aircraft's design in calculating both the horizontal control surface size, and in determining the stability of the aircraft design.

Determination of the C.G. is derived from the definition, finding the one point where all the weights act as one force. This is done by summing all the weights and moments of the aircraft components about the nose of the aircraft, and dividing the two as shown in Equation 1.

\[
\text{C.G.} = \frac{\sum \text{Moments}}{\sum \text{Weights}} \tag{1}
\]

The C.G. position is determined in units of length from the nose. The C.G. provides a reference point for the summation of aerodynamic moments of the aircraft.

Since ACSYNT is a conceptual design program, there is little information calculated in ACSYNT about the positions of the different aircraft components. The program calculates only the positions of the engine(s), wing, vertical tail, aft tail, and forward canard. Positions for all other components are selected by the designer and input into the stability program.

The component positions that ACSYNT does calculate are determined in the following manner. The position of the engine is determined from the value of the length and position of the engine pod supplied through ACSYNT's geometry module. This can be modified in the stability module by the designer through adding or subtracting a fraction
of the pod length. The locations of the wing, horizontal tail, vertical tail, and canard are determined from values of the quarter-chord (C/4) points for each aerodynamic surface as supplied by ACSYNT. The stability module allows the designer to vary these positions by adding or subtracting a fraction of the mean aerodynamic chord (MAC).

All other component positions are left as the designers choice, allowing placement of the components at any point along the fuselage. The component positions are calculated using multiplying factors input by the designer. Equation 2 shows an example of this process where the C.G. of the fuselage is calculated as a function of the fuselage length (BODL), and a multiplying factor (Xf_{fus}). Designer input multiplying factors allow for a more flexible design process, giving the designer more freedom in the internal layout of the aircraft. If the designer does not desire to specify any or all of the component positions, default positions inside the module are used.

\[ X_{c_{g_{fus}}} = X_{f_{fus}} \times BODL \] (2)

When determining the stability of the aircraft, it is important to calculate the shift in C.G. throughout the flight as fuel is used and/or weapons deployed. If the C.G. travels too far forward or aft during the flight, it can put the aircraft in an unstable condition, making it impossible to fly.

The ACSYNT program allows the designer to divide the aircraft mission into a maximum of 12 different phases. The designer specifies for each phase the altitude, range, Mach number, and engine thrust setting of the aircraft. The designer can also specify if any weapons are to be deployed at the end of a phase. For each phase, ACSYNT determines the lift, drag, and fuel usage. It can also determine the optimum altitude, and optimum Mach number. ACSYNT then determines a new aircraft weight for the next phase of flight by subtracting out the weight of the fuel used and weights of any weapons deployed.
To determine the shift in the C.G. position at the end of each phase, the weight sum and moment sum are calculated using the weights for each phase. To further enhance the capabilities of the stability module, modifications were made to include the weight of any external tanks. When the amount of fuel used becomes greater than the amount of fuel stored in the external tanks, the module removes the external tank weights, removing them from future C.G. calculations.

Once the C.G. had been determined and the C.G. range calculated, the position of the C.G. with respect to the point of ground contact of the main gear is evaluated. The position of the point of ground contact is important when determining the horizontal control surface size necessary for take-off rotation. From U.S. Navy aircraft design specifications\(^1\), the C.G. should be located at least 15 degrees forward of the main gear, as shown in Figure 1. This assures the aircraft will not accidentally tip over during ground handling operations, and that enough weight rests on the nose gear to maintain positive nose wheel steering of the aircraft. The stability module compares the C.G.

![Figure 1: Aft C.G. Limits Relative to Main Landing Gear](image)

position calculated for each phase of flight to the position of the main gear. If the 15 degree requirement is not satisfied, the module moves the point of contact aft a fraction of the fuselage length (0.01 * BODL). It recalculates the C.G. until the requirement is satisfied. Comparison between the C.G. and the point of contact during all phases of flight assures the aircraft will meet the requirements if an inflight emergency were to require landing prior to mission completion.
CHAPTER 3
Horizontal Control Surface Sizing

There are two different methods used in determining the size of the horizontal control surface. The first method determines the size of the surface necessary to rotate the aircraft during takeoff. The second method, discussed in a later section, determines if the control surface size calculated by the first method is large enough to satisfy the forward and aft C.G. limits of the aircraft. When both methods are satisfied, the aircraft will have a horizontal control surface large enough to maintain positive longitudinal control.

The first method sizes the horizontal control surface by a summation of moments and forces about the point of ground contact of the main landing gear on the aircraft, as shown in Figure 2. The forces on the aircraft at takeoff rotation include: wing lift, horizontal control surface lift, wing-body pitching moment, aircraft drag, aircraft weight, and engine thrust. Also included is the aircraft acceleration. The moment arms for each of these are taken with respect to their positions from the point of ground contact. This point is used since the aircraft needs to rotate about the point of contact during takeoff. The wing lift is positioned at the aerodynamic center (AC). It is modified to include ground effects, which

Figure 2: Summation of Moments about the Main Landing Gear
discussed in a later section. The aircraft's drag and acceleration are placed along the aircraft centerline (ACSYNT calculates the drag of the entire aircraft and uses that in all its calculations. This includes the drag of the wing, fuselage, and tail surfaces. The total drag force as defined by ACSYNT is therefore put at the aircraft's centerline). The maximum takeoff weight, as determined by ACSYNT's weight module, is located at the C.G. The horizontal control surface lift, also modified for ground effect, is placed at its AC. In ACSYNT, the ACs of all lifting surfaces are the quarter-chord point of the mean aerodynamic chord (MAC). The wing-body pitching moment is positioned about the point of ground contact. For the takeoff analysis, a conventional takeoff is assumed. This means that the forward and aft thrust vectors \( T_1 \) and \( T_2 \) point in the aft direction. Therefore the only moments created by the thrust vectors are due to the vertical displacements \( Z_{T1} \) and \( Z_{T2} \) of the two vectors.

The horizontal positions of the wing lift, aircraft weight, and horizontal control surface lift depend on whether the horizontal control surface used for pitch control is a conventional aft tail or a forward canard. For an aft tail configuration, the wing lift is placed forward of the C.G. and the point of ground contact, and the horizontal tail lift is placed at its AC in the rear of the aircraft. For a canard configuration, the wing lift is placed behind the C.G., and the canard lift is placed at its AC in the front of the aircraft. This leads to two different equations for the summation of moments about the point of contact of the aircraft. A third equation is derived for the case of both a canard and a horizontal tail. In this case, the canard is assumed to be used only for trim.\(^2\) Its size and shape are specified by the designer, who then uses the stability module to size the aft

\(^2\) The ACSYNT program in its Aerodynamics and Geometry modules is unable to handle cases where there exists both a horizontal tail and a canard on the same aircraft. Having the designer input the canard size and shape, allows the designer to evaluate those cases where both exist.
horizontal tail. This uses the same equation as the case with a horizontal tail, but includes a term for the moment created by the lift of the canard acting at its AC.

Examining first the case for which there exists a horizontal tail only, the equation for the summation of moments created by the aerodynamic forces about the point of ground contact is seen in Equation 3. This is modified to account for any vertical thrust offset as specified by the designer.

\[
M_{mg} = -CM_{wb} q S c - [Z_{mg} - Z_{T1}] T_1 - [Z_{mg} - Z_{T2}] T_2 - [X_{mg} - X_{cG}] W + [X_{mg} - X_{ac}] L_{wb} + [Z_{mg} - Z_D] D + \frac{W}{g} V Z_{mg} - [X_{ac ht} - X_{mg}] L_{ht}
\]

In this equation, the lift of the wing (\(L_{wb}\)) is replaced with the classic definition

\[
L_{wb} = C_{L_w g} \alpha_w q S
\]

where the subscript \(g\) represents changes due to ground effect which is discussed in the section titled "Aerodynamic Calculations".

From Newton's equation of motion \(F = ma\), the aircraft acceleration component is replaced by the forces in the horizontal direction. These forces include thrust, drag, and the reaction force of the aircraft acting on the runway. This replaces the acceleration term for the aircraft with

\[
\frac{W}{g} V = T - D - \mu R
\]

---

3 Roskam, Jan.; Airplane Flight Dynamics and Automatic Flight Controls, Part 1; Published by the author; 519 Boulder, Lawrence KA; Second Printing, 1982; 373-375
From the summation of forces in the vertical direction, it is possible to solve for the reaction force of the aircraft acting on the ground.

\[ R = W - L_{wb} - L_{ht} \quad (6) \]

In order to solve for the minimum tail size needed to rotate the aircraft, the forces about the point of ground contact must be balanced. To achieve this condition, \( M_{mg} \) in Equation 3, is set equal to zero. The horizontal tail lift \( (L_{HT}) \), is solved using the combination of Equations 3, 4, 5, and 6, as seen in Equation 7. Equation 7 is used to calculate the necessary tail lift in terms of the different aerodynamic forces.

\[ L_{ht} = \frac{L_1 + L_2 - L_4 - L_6 + L_7 - L_8}{\Delta} \quad (7) \]

where

\[ L_1 = -C_{M_{wb}} q S c \quad L_6 = [X_{mg} - X_{cg} + \mu Z_{mg}] W \]
\[ L_2 = [X_{mg} \cdot (X_{cg} - X_{T}) + \mu Z_{mg}] T_1 \quad L_7 = [X_{mg} - X_{ac_{wb}} + \mu Z_{mg}] L_{wb} \]
\[ L_4 = [X_{T} + X_{cg} \cdot X_{mg} \cdot \mu Z_{mg}] T_2 \quad \Delta = [X_{ac_{ht}} \cdot X_{mg} \cdot \mu Z_{mg}] \]
\[ L_8 = [Z_{D}] D \]

It is important to note that the lift required by the horizontal tail to rotate about the point of ground contact is negative in value (that is pointing down), which requires a negative horizontal tail angle of attack. From the classic equation for lift, the horizontal tail lift is expressed in Equation 8.
Where the angle of attack of the tail as shown in Figure 3. Rearranging Equation 8,

\[ L_{ht} = C_{L_{\alpha_{ht}}} g \left( \alpha_w + i_{ht} - i_w + \tau \delta_c \right) q S_{ht} - C_{L_{\alpha_{ht}}} e g q S_{ht} \]  

creates Equation 9. Equation 9 is used to solve for the horizontal control surface size at take off rotation, while in ground effect.

\[ S_{ht} = \frac{L_{ht}}{C_{L_{\alpha_{ht}}} g \left( \alpha_w + i_{ht} - i_w + \tau \delta_c \right) q - C_{L_{\alpha_{ht}}} e g q} \]  

For the case where there exists both an aft horizontal tail and a forward canard, the assumption is made that the canard is used only for trim, and not for longitudinal control purposes. With the size of the canard defined by the designer, the stability module is used to determine the size of the aft horizontal tail. Modifying Equation 7 to include the moment created by the canard lift placed at its AC, leads to Equation 10.

\[ L_{ht} = \frac{L_1 + L_2 - L_4 - L_6 + L_7 - L_8 + L_9}{\Delta} \]  

where

\[ L_9 = [X_{mg} - X_{ac_{cn}} + \mu Z_{mg} ] C_{L_{ac_{cn}}} g \left( \alpha_w \varepsilon_{up} + i_{cn} - i_w \right) \]
The canard angle of attack is shown in Figure 4, where the subscript "up" indicates the effects of upwash due to the wing.

The case for which there exists only a canard on the aircraft, a slightly different equation is derived due to the different directions the moments act about the point of ground contact. The C.G. of the aircraft is now put forward of the wing AC, which gives a negative moment created by the wing lift about the point of contact. Since the canard is placed forward of the C.G., the moment created by its lift needs to have a positive value to balance the moment equation. Equation 11 is derived from the summation of moments about the point of ground contact with a forward canard and any thrust offset.

\[
M_{mg} = -C_{Mwb} q S c \cdot [Z_{mg} \cdot Z_{T_1}] T_1 - [Z_{mg} \cdot Z_{T_2}] T_2 - [X_{mg} \cdot X_{cg}] W \\
- [X_{ac cg} - X_{mg}] L_{wh} + [Z_{mg} \cdot Z_{D}] D + \frac{W}{g} \dot{V} Z_{mg} + [X_{mg} - X_{ac cg}] L_{cn}
\]

Combining Equation 11 with Equations 4, 5, and 6, gives a solution for the canard lift

\[
L_{cn} = \frac{-L_1 \cdot L_2 + L_4 + L_6 + L_7 + L_8}{\Delta}
\]
where \( L_1 \) through \( L_6 \), and \( L_8 \) are defined in the same terms as used in Equation 7, and where \( L_7 \) and \( \Delta \) are defined as

\[
L_7 = \left[ X_{ac_{wb}} - X_{mg} - \mu Z_{mg} \right] L_{wb}
\]
\[
\Delta = \left[ X_{mg} - X_{ac_{cn}} + \mu Z_{mg} \right]
\]

Finally, using the classic equation for the canard lift, the canard surface area is calculated using

\[
S_{cn} = \frac{L_{cn}}{C_{l\alpha e} \left( \alpha_w + \epsilon_{up} + i_{cn} - i_w + \tau \delta_c \right) q}
\] (13)

The above set of equations are used to determine the size of the horizontal control surface used for longitudinal control. These equations work for either an aft mounted horizontal tail, or a forward mounted canard. The next procedure is developed to insure that the horizontal stabilizer and control surface are large enough to permit an acceptable amount of C.G. travel while maintaining aircraft stability.
CHAPTER 4
Forward and Aft Center of Gravity Limits

Using the preceding methods, it is possible to determine the minimum horizontal control surface size for take off rotation, and the C.G. range for the specified mission. Longitudinal controllability is now determined by calculating the forward and aft C.G. limits. These limits are then compared to the C.G. range determined. Then if necessary, the control surface size is increased to encompass the C.G. range within the forward and aft limits.

First the forward C.G. limit is determined. This is calculated for the worst possible case of flight, that is with the aircraft in the landing configuration. This includes effects due to high-lift at low-speed while in ground effect. Using the minimum horizontal control surface area calculated earlier, the equation for the summation of moments about the aircraft's C.G. is determined from Figure 5. This results in Equation 14. The forces acting on the aircraft in the landing configuration are: the maximum wing lift, wing-body pitching moment with flaps extended, aircraft drag, horizontal control surface lift, and forces due to thrust offset. The wing lift is positioned at the AC. The wing-body
pitching moment is placed about the C.G. As was done earlier, the aircraft's drag is placed along the centerline of the aircraft (this force then falls out of the equation when assuming that the C.G. lies along the centerline). The horizontal control surface lift is placed at its respective AC, and the thrust offset forces are positioned according to the designer's inputs.

\[
C_{Mcg} = C_L \frac{X_{cg} - X_{ac}}{c} - C_{Mwb} - C_{La\delta e}(\alpha_w \cdot \epsilon + i_{HT} - i_w + \tau \delta_e) \frac{S_{HT} (X_{ac\delta e} - X_{cg})}{c} \eta_{HT}
\]

\[
+ C_{LaCN}(\alpha_w + \epsilon_{up} + iCN \cdot i_w + \tau \delta_e) \frac{S_{CN} (X_{cg} - X_{acCN})}{c} \eta_{CN} + \frac{(T_1 Z_{T1} + T_2 Z_{T2})}{q S} N
\]

(14)

To find a solution for the forward C.G. position, the moment coefficient about the C.G. is set equal to zero, and Equation 14 is rearranged for determination of \( X_{cg} \). The final solution is shown in Equation 15. By setting the moment coefficient to zero, a solution is calculated for the most forward C.G. position where the horizontal control surface is just able to maintain the aircraft in a level attitude with maximum elevator deflection.

\[
X_{cg\text{fore}} = \frac{C_L X_{ac} + c C_{Mwb} + C_{La\delta e}(\alpha_w \cdot \epsilon + i_{HT} - i_w + \tau \delta_e) \frac{S_{HT} X_{ac\delta e} \eta_{HT}}{\Delta}}{\Delta}
\]

\[
+ C_{LaCN}(\alpha_w + \epsilon_{up} + iCN \cdot i_w + \tau \delta_e) \frac{S_{CN} X_{acCN} \eta_{CN} - \frac{(T_1 Z_{T1} + T_2 Z_{T2})}{q S}}{\Delta} N
\]

(15)

where

\[
\Delta = C_L + C_{La\delta e}(\alpha_w \cdot \epsilon + i_{HT} - i_w + \tau \delta_e) \frac{S_{HT} \eta_{HT}}{\Delta}
\]

\[
+ C_{LaCN}(\alpha_w + \epsilon_{up} + iCN \cdot i_w + \tau \delta_e) \frac{S_{CN} \eta_{CN}}{\Delta}
\]
In Equation 15, the $C_L$ is the maximum lift coefficient, and $\delta_e$ is the maximum elevator deflection.

Once the forward C.G. limit is determined, it is compared to the different C.G. positions calculated for each of the mission phases of the aircraft. If any of these positions fall forward of the calculated limit, the horizontal control surface size is increased an incremental amount (5 square feet), and the forward limit is recalculated. This process is repeated until all the mission C.G. positions fall aft of the forward C.G. limit.

The second comparison for the horizontal control surface size is the calculation of the aft C.G. limit. The aft C.G. limit is that point where the moments about the C.G. no longer change with angle of attack. This is written as

$$\frac{dC_M}{d\alpha} = 0 \quad (16)$$

This aft limit is also known as the neutral point ($N_0$) of the aircraft. If the C.G. moves behind this point, the change in moment with respect to angle of attack becomes negative which makes the aircraft unstable. The $N_0$ of the aircraft is determined starting with the equation for the moment about the aircraft’s C.G., Equation 14. Differentiating this equation with respect to angle of attack results in Equation 17. This includes contributions

$$C_M = C_L \alpha \left[ X_{cg} - X_{ac} \right] - C_L a_{flare} \eta \frac{S_{H_T}}{S} \left( X_{ac_{ij}} - X_{cg} \right) \left( 1 - \frac{d\delta}{d\alpha} + \tau \frac{d\delta_{H_T}}{d\alpha} \right)$$

$$+ C_L \alpha \eta \frac{S_{CN}}{S} \left( X_{cg} - X_{ac_{N}} \right) \left( 1 - \frac{d\delta}{d\alpha_{up}} + \tau \frac{d\delta_{CN}}{d\alpha} \right) \quad (17)$$
of the wing, horizontal tail, and forward canard. The fuselage and thrust vector contributions are small with respect to the other effects, and are therefore removed from the differentiation.

The neutral point is determined using the equation derived from the following procedure. Equating $C_{Ma}$ to zero, and rearranging Equation 17 in terms of $X_{cg}$ results in Equation 18. In this equation, the $N_0$ is given in percent wing MAC. In order to compare

$$N_0 = \frac{X_{ac_wb} + \frac{C_{La_Hr} S_{HT}}{C_{La}} \frac{S}{S} \eta_{HT} \left( 1 - \frac{d\epsilon}{d\alpha} + \tau \frac{d\delta}{d\alpha} \right)}{\Delta}$$

$$+ \frac{X_{ac_CN} \frac{C_{La_CN} S_{CN}}{C_{La}} \frac{S}{S} \eta_{CN} \left( 1 - \frac{d\epsilon}{d\alpha_{up}} + \tau \frac{d\delta}{d\alpha} \right)}{\Delta}$$

where

$$\Delta = 1 + \frac{C_{La_Hr} S_{HT}}{C_{La}} \frac{S}{S} \eta_{HT} \left( 1 - \frac{d\epsilon}{d\alpha} + \tau \frac{d\delta}{d\alpha} \right) + \frac{C_{La_CN} S_{CN}}{C_{La}} \frac{S}{S} \eta_{CN} \left( 1 - \frac{d\epsilon}{d\alpha_{up}} + \tau \frac{d\delta}{d\alpha} \right)$$

it to the C.G. range determined earlier, the aft C.G. limit must be calculated in units of length from the nose of the aircraft. The equation used for this calculation is shown in Equation 19.

$$X_{cg_{aft}} = X_{LE} + MAC N_0$$

Once the aft C.G. limit is determined, it is compared to the C.G. range. If any the C.G. positions do not fall forward of the aft limit, then the horizontal control surface is
increased in size an incremental amount (5 sq. ft.). The C.G. limit is reevaluated, and compared to the C.G. positions. This is repeated until the forward and aft limits encompass all the C.G. positions, and the aircraft becomes stable throughout all phases of the mission.

The horizontal control surface size is now the minimum size necessary to maintain aircraft control during take off rotation and low speed landing flight. It is also large enough to insure controllability, since the module assures the C.G. will remain in limits. The final step in sizing the horizontal control surface is to determine if it is large enough to insure aircraft stability during flight.
CHAPTER 5
Aircraft Stability

In order to determine if the conceptual aircraft design is statically stable in flight, the pitching moment curve slope \( \frac{dC_{M_{cg}}}{dC_L} \) or the static margin (SM) must be determined. The aircraft is stable if it has a negative value for the \( \frac{dC_M}{dC_L} \), the larger the negative value, the more stable the aircraft. The equation used in solving for the \( \frac{dC_M}{dC_L} \) is developed from a summation of moments about the C.G., and is nondimensionalized by dividing by the dynamic pressure, the reference wing area, and the reference chord length. (Note that the drag does not appear since it is placed at the C.G. of the aircraft due to the limitations of the ACSYNT program in its calculations of total aircraft drag.)

\[
C_{M_{cg}} = C_L \frac{X_{cg} - X_{ac}}{c} - C_{M_{wb}} - C_{L_{a_{rr}}} (\alpha_w - \epsilon + i_{HT} - i_w + \tau \delta_c) \frac{S_{HT}}{S} \frac{(X_{a_{rr}} - X_{cg})}{c} \eta_{HT} \\
+ C_{L_{a_{CN}}} (\alpha_w + \epsilon_{up} + i_{CN} - i_w + \tau \delta_c) \frac{S_{CN}}{S} \frac{(X_{cg} - X_{a_{CN}})}{c} \eta_{CN} + \frac{(T_1 Z_{T1} + T_2 Z_{T2})}{q S c} N
\]

which is modified by taking the derivative with respect to the lift coefficient.

\[
\frac{dC_M}{dC_L} = \frac{X_{cg} - X_{ac}}{c} - \frac{dC_{M_{wb}}}{dC_L} - \frac{C_{L_{a_{rr}}}}{C_L} \frac{S_{HT}}{S} \frac{(X_{a_{rr}} - X_{cg})}{c} \eta_{HT} \left(1 - \frac{d\epsilon}{d\alpha}\right) \\
+ \frac{C_{L_{a_{CN}}}}{C_L} \frac{S_{CN}}{S} \frac{(X_{cg} - X_{a_{CN}})}{c} \eta_{CN} \left(1 - \frac{d\epsilon}{d\alpha}\right)_{up} + \frac{(T_1 Z_{T1} + T_2 Z_{T2})}{W c} N
\]
Each section in the equation represents the contribution of the different aircraft components such as the wing, fuselage, aft and forward horizontal control surface, and engine thrust. The contribution of each component is calculated separately, and the results summed to determine the total aircraft stability coefficient.

The wing contribution is solved using

$$\frac{dC_m}{dC_{L_\text{wing}}} = \frac{X_{cg} - X_{ac}}{c}$$  \hspace{1cm} (22)

This solution assumes small angles of attack. The contribution of the wing is stabilizing when the C.G. is forward of the AC, and destabilizing when it is aft.

The contribution of the fuselage and nacelles, $dC_{M_{wfb}}/dC_L$, is estimated using the equation

$$\frac{dC_{M_{wfb}}}{dC_L} = K_f \frac{BDMAX^2 BODL}{S c C_L \alpha}$$  \hspace{1cm} (23)

where the fuselage stability coefficient $K_f$ is determined from Figure 6. $K_f$ is given as a function of the position of the wing root quarter-chord.

The contribution of an aft mounted horizontal control surface is solved using

$$\frac{dC_m}{dC_{L_{HT}}} = \frac{a_{HT}}{a} \frac{S_{HT}}{S} \frac{X_{ac_{HT}} - X_{c_{HT}}}{c} \eta_{HT} \left(1 - \frac{de}{d\alpha}\right)$$  \hspace{1cm} (24)


5 ibid., 229.

6 ibid., 219-220.
which includes effects due to downwash of the wing on the tail. In this solution, stick-free effects are included by multiplying the horizontal control surface contribution by a factor called the free elevator factor ($F_e$), determination of which is discussed in the chapter on "Additional Aerodynamic Calculations".

The effect of a forward canard on the stability of the aircraft is calculated using

$$\frac{dC_m}{dC_{\text{CN}}} = \frac{a_{\text{CN}}}{a} \frac{S_{\text{CN}}}{S} \frac{X_{\text{CN}}}{c} \eta_{\text{CN}} (1 - \frac{d\alpha}{d\alpha_{\text{up}}})$$

(25)

The upwash of the wing on the canard, and the lift curve slope of the canard are calculated in the section "Additional Aerodynamic Calculations". The stick-free effects are included in the canard calculations only if there is no aft horizontal tail on the aircraft. If both a canard and an aft mounted tail exist, then it is assumed the canard is used for trim only. The stick-free effects for the canard are calculated in the same manner as the stick-free effects for the horizontal tail.

![Figure 6: Fuselage Stability Coefficient ($K_f$) vs. Position of Wing Root Quarter-Chord ($C_{\text{root}}/4$)](Reproduced from Reference 4)
The contribution of engine thrust is calculated using Equation 26, which assumes that the lift equals the weight of the aircraft.

\[
\frac{dC_{M_{th}}}{dC_L} = \left( \frac{T_1 Z_{T1} + T_2 Z_{T2}}{W c} \right) N
\]  

(26)

Where the contribution of the thrust is due to offset of the thrust vectors from the aircraft centerline. This solution allows the designer to include effects for multi-engine aircraft where not all engines are located the same height from the centerline, and engines with more than one exhaust nozzle creating multiple thrust vectors.

A second measure of aircraft stability is the SM. This is simply the distance the \(N_0\) is aft of the C.G. at any given time during flight. It is an indication of how easy or hard it is for the aircraft to rotate about the lateral axis. The larger the distance between the \(N_0\) and the C.G., the larger the SM, and the harder it is for the aircraft to be rotated. Conversely, the smaller the SM, the easier it is to rotate the aircraft. The SM is calculated from

\[
SM = N_0 - \frac{X_{cg} - X_{LE}}{C}
\]  

(27)

which gives the distance of the C.G. in front of the \(N_0\) in percent MAC. Accepted values for the SM for stable aircraft range from 10% MAC for a transport, to 5% MAC for a fighter (from reference 5). All the aircraft stability and control parameters have now been determined. There are however a few aerodynamic effects that influence the aircraft which need to be determined.
CHAPTER 6
Aerodynamic Effects

In calculating the size of the horizontal control surface needed to maintain stability, certain aerodynamic characteristics that affect the solution need to be determined. Three important characteristics are included in the stability module. These include the downwash of the wing on an aft horizontal tail, the upwash of the wing on a forward canard, and ground effects.

Following a method outlined in reference 1, the downwash of the wing on an aft mounted horizontal tail is calculated\(^7\). This method gives the downwash angle at the tail as a function of the effective wing aspect ratio (AR\(_{\text{eff}}\)), the effective wing span (b\(_{\text{eff}}\)), and the tail height above or below the trailing wing vortex. This method assumes a large wing span to horizontal tail span ratio (b/b\(_{\text{ht}}\) ≥ 1.5), wing trailing edge vortex separation, and subsonic flow.

First AR\(_{\text{eff}}\) and b\(_{\text{eff}}\) are determined. These are determined from Figure 7, dependent on the angle-of-attack and geometry of the wing. Once the AR\(_{\text{eff}}\) and the b\(_{\text{eff}}\) have been determined, the downwash angle is calculated. Equations 28, 29, and 30 are used to solve for the span of the vortex core (b\(_{v}\)) at the horizontal tail quarter-chord.

\[
b_v = b_{\text{eff}} - (b_{\text{eff}} - b_v) \left( \frac{2 L_{\text{eff}}}{b \xi_{\text{nl}}} \right)^{1/2}
\]

\(^7\) Hoak, D.E. et al; USAF Stability and Control Datcom; Wright Patterson AFB Ohio, 45433; Revised 1970; sect. 4.4.1
Angle of attack $\alpha$ for zero lift

$C_L = C_{L_{max}} - \sigma_0$

Sweepback at wing quarter chord $\Lambda_{1/4}$

Wing taper ratio $\lambda$

Effective wing aspect ratio $A_{eff}$

Effective wing span $b_{eff}$

Figure 7: Effective Wing Aspect Ratio and Span
- Low Speeds
(Reproduced From Reference 1)
where $L_{\text{eff}}$ is the distance from the wing tip trailing edge to the horizontal tail quarter-chord and $\xi_{ru}$ is a dummy variable. The following are used in solving for $b_v$.

$$b_{vr_i} = \left[ 0.78 + 0.10(\lambda - 0.4) + 0.003 \Lambda_{c/4} \right] b_{\text{eff}}$$  \hspace{1cm} (29)

and

$$\xi_{ru} = \frac{0.56 AR_{\text{eff}}}{C_L}$$  \hspace{1cm} (30)

Equation 31 is used to solve for the height of the horizontal tail above or below the trailing wing vortex core ($a$). This is a function of the tail height above the wing chord line ($h_{HT}$), and the effective distance from the wing vortex separation to horizontal tail.

$$a = h_{HT} - L_{\text{eff}} \left( \frac{\alpha - 0.41 C_L}{\pi AR_{\text{eff}}} \right) - \frac{b_{\text{eff}}}{2} \tan(\Gamma)$$  \hspace{1cm} (31)

Finally, the ratio of the downwash at the tail to the downwash at the vortex core is calculated using

$$\varepsilon = \frac{1}{1 + \left( \frac{2a}{b_v} \right)^2}$$  \hspace{1cm} (32)

where the downwash at the vortex core is determined from

$$\varepsilon_{v} = \frac{1.62 C_L}{\pi AR}$$  \hspace{1cm} (33)

where the AR for this equation is the actual wing AR, and not effective.

The upwash effects of the wing acting on a forward canard are determined as a function of wing AR, wing root chord ($C_{\text{root}}$), and the distance the canard is ahead of the
wing quarter-chord. The upwash effects are determined using Figure 8\textsuperscript{8}. This solution is accurate only for Mach numbers less than one, and as such currently limits the analysis to subsonic missions.

\begin{figure}[h]
\centering
\includegraphics[width=0.7\textwidth]{figure8.png}
\caption{Upwash Effects in Front of the Wing \(l \cdot d \epsilon/d\alpha\) vs. Forward Position in Percent Wing Quarter-Chord \(x/c_{\text{root}}\) as a Function of Wing Aspect Ratio (AR)}
\end{figure}

Since principal sizing of the horizontal control surface is computed at takeoff or landing, it is important to include ground effects. Ground effects can adversely affect the horizontal control surface size by decreasing the downwash or upwash angles on the horizontal control surface. This decreases the overall angle of attack of the surface, which then increases the control surface size needed to generate the required amount of lift. There are three areas which are influenced by ground effect. The first two are the lift of

\textsuperscript{8} McCormick, B.W., \textit{Aerodynamics, Aeronautics, and Flight Dynamics}, J. Wiley and Sons. 1979. pg. 520.
the wing and the lift of the horizontal control surface. The third is the downwash angle of the wing on the horizontal control surface.

As discussed earlier, when sizing the horizontal control surface, it is necessary to calculate the lift produced by the wing and the lift required of the horizontal control surface. Once these have been determined, it is a simple matter of modifying them to include ground effects. Ground effects increase the closer the aircraft is to the ground. To get maximum effect of the ground on the horizontal control surface size, the height of the aircraft above the ground is reduced to a minimum. The minimum possible height occurs when the aircraft is resting on the ground with landing gear extended. This height is used along with Figure 9 to calculate the ratio of lift-curve slopes in and out of ground effect. Once the change in the lift curve slope is determined, the wing lift in ground effect is calculated using

$$L_{wb} = C_{L_{\alpha}} g \alpha_w q S$$

(34)

the horizontal tail lift is determined using

$$L_{ht} = C_{L_{\alpha ht}} g \left( \alpha_w + i_{ht} - i_w + \tau \delta_c \right) q S_{ht} - C_{L_{\alpha ht}} g \left( \alpha_w + i_{ht} - i_w + \tau \delta_c \right) q S_{ht}$$

(35)

and the canard lift is found using

$$L_{cn} = C_{L_{\alpha cn}} g \left( \alpha_w + e_{up} + i_{cn} - i_w + \tau \delta_c \right) q S_{cn}$$

(36)

In determining how ground effects influence the downwash angle, the method used calculates the change in downwash as a function of several wing and horizontal tail

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geometric and aerodynamic parameters. These include effective wing span, wing aspect ratio, wing taper ratio, wing height above ground, and horizontal tail height above ground. These also include the wing lift coefficient, and the change in wing lift coefficient due to flaps. The effective span of the wing is determined as a function of the wing lift.

![Figure 9: Ratio of Lift Curve Slopes in and out of Ground Effects (a/a)](attachment:figure9)

The change in wing lift due to flaps, and the effective wing and flap span ratios, as calculated using Equation 37. Once the effective span is calculated, the change in downwash is determined using Equation 38, where H is the wing height above the ground.
ground, and $H_{HT}$ is the height of the horizontal tail above ground. This solution for the change in downwash angle is good for Mach numbers less than one.

$$b_{\text{eff}} = \left[ \frac{C_{Lw} + \Delta C_{Lf}}{C_{Lw} + \Delta C_{Lf}} \right] \left[ \frac{b_w}{b_f} \right]$$

where

$$b_w' = \left( \frac{b_w}{b} \right) b$$

and where

$$b_f' = \left( \frac{b_f}{b_w} \right) \left( \frac{b_w'}{b} \right) b$$

The ratios $b_w'/b$, and $b_f'/b_w'$ are determined using Figures 10 and 11.

$$\Delta \varepsilon_g = \varepsilon \left[ \frac{b_{\text{eff}}^2 + 4 (H_{HT} - H)^2}{b_{\text{eff}}^2 + 4 (H_{HT} + H)^2} \right]$$
Figure 10: Effective Span of Wing in Presence of Ground (Reproduced from reference 1)

Figure 11: Effective Span of Flaps in Presence of Ground (Reproduced from reference 1)
CHAPTER 7
Module Verification

The stability module was validated by comparing computer generated data of two types of aircraft to data available on the production aircraft. Computer models of a General Dynamics F-16A and a Boeing 727-200 were generated for comparison using the ACSYNT program. These computer models matched various parameters of the production aircraft which included geometry, aerodynamics, and weights. A comparison was then made between the horizontal control surface size for the production aircraft versus the control surface size calculated using the stability module.

The method used to solve for the horizontal control surface size consists of three main steps. First, the different aircraft components are placed along the fuselage. It is important to place the major components of the aircraft in their specific locations. These include the wing, horizontal control surface, engine(s), fuel in the fuselage, fuel in the wing, and the main landing gear. Once the components are in place, the C.G. position and range for the actual aircraft is matched in the computer model, allowing for a shift in C.G. from the fully forward to fully aft positions. This matching necessitates the moving of less important components to different positions, or recalculating the weights of some components. After determining the C.G., the ACs of the wing and horizontal control surface are compared to assure the lift forces are acting at the correct positions.

Once the C.G. position, C.G. range, and the ACs correspond to their respective positions on the actual aircraft, the horizontal control surface sizes are compared. The results, which are presented in Tables 1, and 2, show excellent agreement between the actual and computer aircraft. Since more data was available for the F-16A, the solution of the computer module matched best with the production aircraft.
To validate the module for an aircraft with a forward mounted canard, the geometry and weight of the F-16A model was modified to match the dimensions of a Saab Viggen. The comparison for this model is shown in Table 3. This showed good accuracy, with an error in the canard size of only 3.91 percent.

An important factor in these tables is the position of the main landing gear. The stability module uses rotation about the main landing gear to determine the size of the horizontal control surface. It was found that the position of the main landing gear at takeoff had a large effect on the calculated size. Using the F-16A as an example, Figure 12 shows that small changes in main landing gear position can affect the calculated control surface size. Figure 13 shows how changes in main landing gear position affect the forward and aft C.G. limits. Note in both of these figures the tail size and the C.G. limits stabilize when the main gear fall at or forward of the 60 percent fuselage length. This is a result of the stability module forcing the main gear aft to satisfy the 15 degree angle requirement between the main gear and the C.G.
Table 1: F-16A Production and Computer Model Comparison

<table>
<thead>
<tr>
<th></th>
<th>Production</th>
<th>Computer Model</th>
<th>% Difference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Horizontal Tail Size</td>
<td>49.0</td>
<td>48.18</td>
<td>1.67</td>
</tr>
<tr>
<td>Main Gear Position</td>
<td>29.0</td>
<td>28.69</td>
<td>1.07</td>
</tr>
<tr>
<td>MAC</td>
<td>11.3</td>
<td>11.20</td>
<td>0.88</td>
</tr>
<tr>
<td>Quarter-chord Position</td>
<td>25.6</td>
<td>25.55</td>
<td>0.20</td>
</tr>
<tr>
<td>Half-chord Position</td>
<td>28.4</td>
<td>28.35</td>
<td>0.18</td>
</tr>
<tr>
<td>Forward C.G.</td>
<td>*</td>
<td>26.10</td>
<td></td>
</tr>
<tr>
<td>Aft C.G.</td>
<td>*</td>
<td>27.21</td>
<td></td>
</tr>
<tr>
<td>Quarter-chord of Horizontal Tail</td>
<td>41.5</td>
<td>41.32</td>
<td>0.43</td>
</tr>
</tbody>
</table>

Note: All distances are in feet from the nose of the aircraft, except the Mean Aerodynamic Chord is in feet from the leading to trailing edges. Horizontal tail size is in feet squared.

*: The author was unable to obtain information on the forward or aft C.G. limits of the Production aircraft.
Table 2: Boeing 727-200 Production and Computer Model Comparison

<table>
<thead>
<tr>
<th></th>
<th>Production</th>
<th>Computer Model</th>
<th>% Difference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Horizontal Tail Size</td>
<td>376.0</td>
<td>379.64</td>
<td>0.96</td>
</tr>
<tr>
<td>Main Gear Position</td>
<td>66.18</td>
<td>68.47</td>
<td>3.34</td>
</tr>
<tr>
<td>MAC</td>
<td>15.5</td>
<td>13.98</td>
<td>9.81</td>
</tr>
<tr>
<td>Quarter-chord Position</td>
<td>63.3</td>
<td>62.84</td>
<td>0.73</td>
</tr>
<tr>
<td>Half-chord Position</td>
<td>67.2</td>
<td>66.34</td>
<td>1.28</td>
</tr>
<tr>
<td>Forward C.G.</td>
<td>*</td>
<td>62.14</td>
<td></td>
</tr>
<tr>
<td>Aft C.G.</td>
<td>*</td>
<td>65.19</td>
<td></td>
</tr>
<tr>
<td>Quarter-chord of</td>
<td>113.7</td>
<td>113.44</td>
<td>0.23</td>
</tr>
<tr>
<td>Horizontal Tail</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Note: All distances are in feet from the nose of the aircraft, except the Mean Aerodynamic Chord is in feet from the leading to trailing edges.

Horizontal tail size is in feet squared.

*: The author was unable to obtain information on the forward or aft C.G. limits of the Production aircraft.
Table 3: Saab JA-37 Viggen Production and Computer Model Comparison

<table>
<thead>
<tr>
<th></th>
<th>Production</th>
<th>Computer Model</th>
<th>% Difference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Horizontal Canard Size</td>
<td>66.7</td>
<td>64.09</td>
<td>3.91</td>
</tr>
<tr>
<td>Main Gear Position</td>
<td>34.8</td>
<td>36.14</td>
<td>3.71</td>
</tr>
<tr>
<td>MAC</td>
<td>17.15</td>
<td>17.94</td>
<td>4.40</td>
</tr>
<tr>
<td>Quarter-chord Position</td>
<td>35.3</td>
<td>32.64</td>
<td>7.54</td>
</tr>
<tr>
<td>Half-chord Position</td>
<td>39.64</td>
<td>37.13</td>
<td>6.33</td>
</tr>
<tr>
<td>Forward C.G.</td>
<td>*</td>
<td>31.40</td>
<td></td>
</tr>
<tr>
<td>Aft C.G.</td>
<td>*</td>
<td>36.05</td>
<td></td>
</tr>
<tr>
<td>Quarter-chord of Horizontal Tail</td>
<td>-</td>
<td>-</td>
<td>-</td>
</tr>
</tbody>
</table>

Note: All distances are in feet from the nose of the aircraft, except the Mean Aerodynamic Chord is in feet from the leading to trailing edges.

Horizontal canard size is in feet squared.

*: The author was unable to obtain information on the forward or aft C.G. limits of the Production aircraft.
Figure 12: Horizontal Tail Size ($S_{ht}$) vs. Main Landing Gear Position ($X_{mg}$)
Figure 13: Forward and Aft C.G. Limits ($X_{cg\,fore}$ and $X_{cg\,aft}$) vs. Main Landing Gear Position ($X_{mg}$)
The stability and control module includes a subroutine which allows for the vectoring of thrust during landing. This gives the designer the ability to evaluate aircraft with hovering and vertical landing capabilities, using the stability equations to create stable designs.

This subroutine is used to calculate one of three different parameters for a stable aircraft in high-lift, low-speed transitioning flight. The three parameters include the forward thrust vector position ($X_{T1}$), the aft thrust vector angle ($\gamma_2$), and the thrust split between the forward thrust vector and the total thrust available (TSPLIT). The stability module is used to solve for these three parameters because they have the greatest impact on the aircraft’s design. The forward thrust vector position has a great effect on the internal arrangement of the aircraft. It represents a large lift producing system, either ducting from the main engine, or a separate auxiliary lift engine. It therefore requires a large amount of internal volume, limiting the placement of other aircraft components. The aft thrust vector angle and the thrust split determine the amount of bleed air that is removed from the main engine. This has a direct effect on the size and thrust of the engine. The more bleed air removed from the engine, the larger it needs to be.

The forward thrust angle is not considered important since the most effective thrust angle is at 90 degrees to the ground. The aft thrust vector position is not considered important in the stability module because it is more economical both in design and weight of the engine to put the aft thrust at the rear of the aircraft (as in conventional aircraft).

The general layout for the thrust vector angles and distances is shown in Figure 14, which shows distances for both the C.G. and the main gear. During take off and cruise...
flight, it is assumed the thrust vectors point in the aft direction, as with a conventional aircraft. This allows the designer to include any thrust offset due to the positions of the engines. During landing, the thrust nozzles are "rotated" and the thrust forces are applied in both the horizontal and vertical directions.

Summation of the thrust and aerodynamic forces about the C.G. result in the familiar equation for $C_{Mcg}$ (Equation 14). This now includes the thrust forces shown in Figure 14, the final result shown in Equation 39.

$$C_{Ma} = C_L \left( \frac{X_{cg} - X_{ac}}{c} \right) - C_{Ma} - C_L \alpha_{HT} \frac{S_{HT}}{S} \left( \frac{X_{acg} - X_{cg}}{c} \right) \eta_{HT}$$

$$+ C_{Ma} \alpha_{CN} \frac{S_{CN}}{S} \left( \frac{X_{acg} - X_{cg}}{c} \right) \eta_{CN} + \left( X_{cg} - X_{T1} \right) \frac{T_1 \sin \gamma_1}{q S c}$$

$$+ Z_{T1} \frac{T_1 \cos \gamma_1}{q S c} - \left( X_{T2} - X_{cg} \right) \frac{T_2 \sin \gamma_2}{q S c} + Z_{T2} \frac{T_2 \cos \gamma_2}{q S c}$$
In order to maintain stability, the summation of moments about the C.G. is set equal to zero. Solutions can now be found for the forward thrust vector position, the aft thrust vector angle, and the thrust split.

Solving for $X_{T1}$ is a straightforward algebraic rearrangement of the moment equation given above. Equation 40 is used to solve for the forward thrust vector position in units of length from the nose of the aircraft. This solution is for a stable aircraft in low-speed flight transitioning from forward flight to hover.

$$X_{T1} = \frac{\text{SUMVT}}{T_1 \sin \gamma_1} + X_{cg}$$

where SUMVT is given by

$$\text{SUMVT} = C_L \left( \frac{X_{cg} - X_{ad}}{c} \right) - C_{M_{xy}} - C_{L_{\text{eff}}} \left( \frac{S_{HT}}{S} \right) \frac{(X_{ac_{HT}} - X_{cg})}{c} \eta_{HT}$$

$$+ C_{L_{\text{eff}}} \frac{S_{CN}}{S} \frac{(X_{ac_{CN}} - X_{cg})}{c} \eta_{CN} + Z_{T1} \frac{T_1 \cos \gamma_1}{q S C}$$

$$+ [X_{T2} - X_{cg}] \frac{T_2 \sin \gamma_2}{q S C} + Z_{T2} \frac{T_2 \cos \gamma_2}{q S C}$$

By definition, the thrust split (TSPLIT) is the ratio of the forward thrust over the total thrust. A solution for the thrust split can be determined by using relationships between the forward thrust, the aft thrust, the total thrust, and the thrust split. The first relationship comes from the definition of the thrust split, as shown in Equation 41. The second relationship comes from the assumption that the aft thrust equals the total thrust minus the forward thrust (this ignores any frictional and heat losses in the forward thrust).

$$\text{TSPLIT} = \frac{T_1}{\text{THRUST}}$$
\[ T_2 = \text{THRUST} - T_1 = \text{THRUST} \{ 1 - \text{TSPLIT} \} \] (42)

Equations 41 and 42 are substituted into Equation 39, which is rearranged to solve for TSPLIT. The final solution is shown in Equation 43.

\[ \text{TSPLIT} = \frac{\text{SUMVT}}{\text{THRUST}} - \frac{X_{T2} \sin \gamma_2}{q S c} + \frac{Z_{T2} \cos \gamma_2}{q S c} \] (43)

where

\[ \text{SUMVT} = C_L \left( \frac{X_{cg} - X_{ac}}{c} \right) - C_{M_ab} - C_{L_{off}} \alpha \frac{S_{HT}}{S} \left( \frac{X_{ac_{HT}} - X_{cg}}{c} \right) \eta_{HT} \]

\[ + C_{L_{acCN}} \alpha \frac{S_{CN}}{S} \left( \frac{X_{ac_{CN}} - X_{cg}}{c} \right) \eta_{CN} \]

and

\[ \Delta = Z_{T2} \frac{\cos \gamma_2}{q S c} - X_{T2} \frac{\sin \gamma_2}{q S c} - X_{T1} \frac{\sin \gamma_1}{q S c} - Z_{T1} \frac{\cos \gamma_1}{q S c} \]

Solving for the aft thrust vector angle (\( \gamma_2 \)) is more involved because it appears in the moment equation twice, once in a sine function, and once in a cosine function. Since a direct solution cannot be found, an iterative approach is used. The boundaries of the solution are known to be 0 degrees and 90 degrees, and are therefore as starting points. The moment equation, Equation 39, is solved for \( C_{Meg} \) with \( \gamma_2 \) set to 0 degrees, and with \( \gamma_2 \) set to 90 degrees. The two solutions are compared, and the one that has the largest magnitude for \( C_{Meg} \) is reduced to halfway between the two boundaries. The moment equation is recalculated for the remaining boundary and for the new boundary value. Again, the solution with the largest magnitude for \( C_{Meg} \) is reduced to halfway between the two boundaries. This series of computations and comparisons continues until the
magnitude for the moment equation approaches zero. The corresponding $\gamma_2$ is a solution for a stable aircraft design for transition from horizontal flight to hover.

The vectored thrust subroutine was used to compare the forward thrust vector positions to the thrust split, and the aft thrust vector angle. The results of this can be seen in Figure 15.
Figure 15: Forward Thrust Vector Position (Xr1) vs. Thrust Split (TSPLIT) for Various Aft Thrust Angles (γ2)

* This graph is for an F-16A modified with a forward nozzle set at γ = 90 degrees and with the aft nozzle set at a position of 0.95 percent of body length.
CHAPTER 9
Additional Aerodynamic Calculations

In the ACSYNT design program, most of the aerodynamic parameters needed by the stability module are solved for in the aerodynamics module. However, some variables are not solved for and others are currently impossible to transfer between the two different modules. These additional variables are calculated inside the stability module using the methods discussed, as taken from the specified references.

The aerodynamic center (AC) of the wing is the first of the aerodynamic variables to be determined. In the aerodynamics module of ACSYNT, the AC is assumed to be the quarter chord point of the MAC of the wing, dependent only on the geometry of the wing and not the Mach number of the aircraft. This is not an accurate representation when designing high speed fighter aircraft. The aerodynamics module does, however, determine the change in the lift-curve slope of the wing with a change in Mach number. This information can be used to determine the change in AC with the change in Mach number with a fair degree of accuracy.

Figure 16 shows the forces and distances used in the calculation of the AC\textsuperscript{11}. The summation of moments about point A results in Equation 44. This equation is nondimensionalized by dividing by the dynamic pressure, the wing chord, and the wing area. This equation is then reduced by taking the derivative with respect to the angle-of-

\[ M_A = q \int_{b/2}^{b} c^2 C_{M_{ac}} dy - q \int_{b/2}^{b} c C_1 y \tan \Lambda dy \quad (44) \]

Figure 16:
Calculation of Wing Aerodynamic Center
by Moments about Root Quarter-Chord, point A.

attack. Noting that by definition the moment coefficient about the AC ($C_{MAC}$) does not change with angle-of-attack, it is therefore zero. This reduced form of this equation is shown in Equation 45.

$$\frac{d C_{MA}}{d \alpha} = - \int_{-b/2}^{b/2} \frac{C_l \alpha}{S} y \tan \Lambda \, dy$$  (45)

Defining $X_A$ to be the distance from the point A to the AC, the the moment about the AC ($M_{AC}$) is defined as the summation of the moment about point A ($M_A$), and the lift
force multiplied by the distance \( X_A \). This is seen in Equation 46.

\[
M_{ac} = M_A + LX_A
\]  

(46)

This equation can be nondimensionalized by dividing by the dynamic pressure, the wing chord, and the wing area. This can also be reduced by taking the derivative with respect to angle-of-attack. This results in Equation 47, again noting that the moment about the AC does not change with change in angle-of-attack.

\[
\frac{dC_{MA}}{d\alpha} = C_{La} \frac{X_A}{c}
\]  

(47)

The distance \( X_A \) is determined using the equation that results from the combination of Equations 45 and 47, as shown in Equation 48.

\[
X_A = \frac{1}{C_{La} S} \int_{-b/2}^{b/2} c C_{la} y \tan \Lambda \, dy
\]  

(48)

This general equation is simplified by assuming a constant lift curve slope along the span of the wing (thereby removing it from the integral), and by assuming a linearly tapered wing (thereby removing the integral completely). This simplified form is shown in Equation 49.

\[
X_{ac} = \left( \frac{1 + 2\lambda}{1 + \lambda} \right) \frac{1}{3} \left[ \frac{b}{2} \tan \Lambda \right] \frac{C_{la}}{C_{La}} + \left( X_{LE_{max}} + \frac{C_{root}}{4} \right)
\]  

(49)

This equation is used to calculate the position of the AC from the point A (the quarter-chord point of the wing center), and accounts for wing sweep and taper ratio. The final step is to account for the effects due to Mach number.
The only aerodynamic characteristic in this equation is the ratio of the two-dimensional to three-dimensional lift curve slopes. It is therefore necessary to determine the change in the lift curve slope with respect to Mach number. This works well with the ACSYNT design program since the only variable that ACSYNT determines Mach effects on is the three-dimensional lift curve slope of the wing. The stability module therefore needs only to be used to determine the ratio of the two-dimensional to three-dimensional lift curve slopes to include Mach effects on the AC.

The ratio of two-dimensional to three-dimensional lift curve slopes is determined by rearranging Equation 4812.

\[
C_{La} = \frac{C_{1a}}{1 + 57.29 \frac{C_{1a}}{\pi \ AR}}
\]  

(50)

The AC is now determined for any geometric shape, and at varying Mach numbers.

It was found that this method is good for Mach numbers up to Mach 1.2, at which point the AC changes at a rapid rate. Comparisons between the method used in the stability module and a graphical method discussed in Reference 1, are shown on Figure 17. This shows the AC shift for a Boeing 727-200 in subsonic flight, and a F-16A in subsonic and supersonic flight. The equations used to determine the AC tend to give a higher value, averaging 4% for the 727-200.

The second aerodynamic parameter not determined in the aerodynamics module of ACSYNT is the three-dimensional lift curve slope of the canard. A solution is found first using the Helmbold equation13, Equation 51. This gives the three-dimensional lift curve


Figure 17: Numerical and Graphical Aerodynamic Center Comparisons

(Graph Reproduced From Reference 1)
slope as a function of the two-dimensional lift curve slope and the AR of the canard. Equation 51 is modified to include the effects of canard sweep angle and Mach number through modification of the two-dimensional lift curve slope$^{14}$. Sweep angle effects are included by the multiplication of the two-dimensional lift curve slope and the cosine of the sweep angle. The Mach number effects are included through division of the two-dimensional lift curve slope by the Prandtl-Glauert compressibility factor. The compressibility factor depends on the subsonic or supersonic flow. The final three-dimensional lift curve slope equation is

$$C_{L\alpha} = C_{1\alpha} \frac{AR}{\left(\frac{C_{1\alpha}}{\pi}\right) + \sqrt{\left(\frac{AR}{\cos \alpha}\right)^2 + \left(\frac{C_{1\alpha}}{\pi}\right)^2 - \left(\frac{AR M_{\infty}}{14} \right)^2}}$$

This equation is simplified somewhat in the stability module by assuming the two-dimensional lift curve slope of the canard equals two-pi.

Also in the stability module, the contribution of the horizontal control surface is modified to include the effects of a stick-free condition. This is determined by calculating the hinge moment parameters of the elevator. The method is used to modify the contribution of the horizontal control surface to the $dC_M/dC_L$, and the equations for the control surface lift. This is done by multiplying with the free elevator factor ($F_e$)$^{15}$, which is calculated from

$$F_e = 1 - \tau \frac{d \delta}{d \alpha} = 1 - \tau \frac{b_1}{b_2}$$

$^{14}$ ibid., 283 - 284.
Where \( b_1 \) and \( b_2 \) are determined from the multiplication of the coefficients found in Figures 18, 19, 20, and 21. These are given as functions of the elevator span, \( t/c \), AR, and balance ratio (BR), each specified by the designer.

\[
b_1 = -0.55 k_1(ce/c) k_1(t/c) k_1(BR) k_1(1/A)
\]

\[
b_2 = -0.89 k_2(ce/c) k_2(t/c) k_2(BR) k_2(1/A)
\]

The control effectiveness factor (\( \tau \)), is determined from Figure 22 as a function of elevator to stabilizer chord ratio.

15 ibid., 495 - 508.
Figure 18: Curves Used in Determining the $k_1$ Coefficients to Calculate Stick-Free Effects. $BR, 1/A, Ce/C$

Figure 19: Curve Used in Determining the $k_1$ Coefficients to Calculate Stick-Free Effects. $t/c$
Figure 20: Curves Used in Determining the k2 Coefficients to Calculate Stick-Free Effects. BR, 1/A, Ce/C

Figure 21: Curve Used in Determining the k2 Coefficient to Calculate Stick-Free Effects. t/c
Figure 22: Control Surface Efficiency Factor ($\tau$) vs. Percent Elevator Chord ($ce/c$)
CHAPTER 10
Conclusions and Recommendations

The use of a design program can enhance many areas in the conceptual aircraft design stage. NASA Ames Research Center has created a computer program, ACSYNT, which does calculations of aircraft geometry, aerodynamics, propulsion, mission performance, and weights. This allows designers to examine a wide range of designs in a relatively short period of time, to create the best possible aircraft.

A stability and control module was created to enhance the conceptual design program, ACSYNT. This module calculates the size of the horizontal control surface, the center of gravity for each mission phase, the forward and aft center of gravity limits, and the longitudinal stability of the design. The stability module calculates the control surface size needed for take off rotation, it examines the stability of the aircraft during the mission, and it determines if the tail size is large enough to maintain controllability during landing. If the tail size is not large enough to meet any of these goals, it is increased until stability and controllability are established.

Comparisons between production aircraft and computer models show that the stability module accurately sizes the tail size for a range of aircraft types. Three aircraft, a B727-200 transport, and a F-16A fighter, and a JA-37 Viggen fighter with a forward mounted canard, were used to evaluate the stability module. In each case, the tail size determined by the module was an accurate representation of the actual aircraft.

In addition to analyzing the horizontal control surface size and the stability of the aircraft, an analysis was done to evaluate vectored thrust applications on the design of a conceptual aircraft. This allows the designer to evaluate one of three different parameters associated with vectored thrust systems, while maintaining stability of the aircraft during
landing. These include the forward thrust vector position, the aft thrust angle, and the thrust split. These parameters were determined to be the most important in the design of the aircraft. The forward thrust position affecting the internal layout of the aircraft, and the aft thrust vector angle and the thrust split affecting the required thrust of the engine.

Recommendations for improvement on this module include the following:

- Calculations of aerodynamic center at higher Mach numbers than what is currently being used.
- Include calculations to determine the downwash and upwash effects of the wing at supersonic Mach numbers.
- Calculations of ground effects on the upwash of the wing.
- Modification of wing lift to include interaction between the wing and the canard vortices.
- In the Vectored thrust analysis, include the ability to augment the forward thrust with an afterburner, duct burner, or ejector.
- In the Vectored thrust analysis, include changes in lift and pitching moment due to jet interaction.
REFERENCES


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Miller, Jay. Aerograph1, General Dynamics F-16 Fighting Falcon; Aerofax Inc., Austin TX, 1982.

Nicolai, Leland M. Fundamentals of Aircraft Design; Mets Inc. 6520 Kingsland, San Jose CA 95120; Revised 1984.


Roskam, J. Airplane Flight Dynamics and Automatic Flight Controls, Part 1; Published by the author, 519 Boulder Lawrence KA 66044; Second Printing 1982.
APPENDIX A

Module Inputs

The following is a list and description of inputs used by the stability and control module. All the input variables have default values, so not all of the variables need to be input. The user inputs these in the namelist 'STABIN' and selecting the module number 5 in the COPES inputs.

Format for the 'STABIN' namelist includes a title line, a maximum of 80 characters long, followed on the next line by 'STABIN' and then the desired inputs. Finally, at the end of the inputs, a 'SEND' statement is needed to tell ACSYNT the input has been completed.

Example:

******* Stability and Control Inputs, F-16A Falcon *******
$STABIN ETACN=.90, XT1=0.40, XFCREW=0.25, XFINST=0.21 ........
SEND

Name     Default    Description
---Real format-----------------------
AWTOT    2.00       Angle of attack of the wing at take off rotation. (degrees)
CBALHT   0.00       Percent chord of the longitudinal control
                 surface ahead of the hinge line.
CELV     0.30       Percent chord of the horizontal stabilizer
                 that is the movable surface.
ETACN    0.90       Canard efficiency.
ETAHT    0.90       Horizontal tail efficiency.
<table>
<thead>
<tr>
<th>Variable</th>
<th>Value</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>ELVDEF</td>
<td>-10.00</td>
<td>Amount of elevator deflection needed for takeoff rotation. (degrees)</td>
</tr>
<tr>
<td>ELVDMAX</td>
<td>-15.00</td>
<td>Maximum amount of elevator deflection needed for controllability at low speed. (degrees)</td>
</tr>
<tr>
<td>GAMAT1</td>
<td>0.00</td>
<td>Forward thrust nozzle rotation angle from the horizontal. (degrees)</td>
</tr>
<tr>
<td>GAMAT2</td>
<td>0.00</td>
<td>Aft thrust nozzle rotation angle from the horizontal. (degrees)</td>
</tr>
<tr>
<td>ICN</td>
<td>0.00</td>
<td>Canard incidence angle. (degrees)</td>
</tr>
<tr>
<td>IHT</td>
<td>0.00</td>
<td>Horizontal tail incidence angle. (degrees)</td>
</tr>
<tr>
<td>SCN1</td>
<td>0.0</td>
<td>Area of canard when both a canard and an aft tail are being used (feet²).</td>
</tr>
<tr>
<td>SPANCN1</td>
<td>0.0</td>
<td>Span of the canard when both a canard and an aft tail are being used (feet).</td>
</tr>
<tr>
<td>TRCN1</td>
<td>0.0</td>
<td>Taper ratio of the canard when both a canard and an aft tail are being used.</td>
</tr>
<tr>
<td>TSPLIT</td>
<td>0.00</td>
<td>Percent thrust split, front-to-rear.</td>
</tr>
<tr>
<td>TZCHT</td>
<td>0.10</td>
<td>Elevator thickness-to-chord ratio.</td>
</tr>
<tr>
<td>VROT</td>
<td>0.30</td>
<td>Mach number at take off rotation.</td>
</tr>
<tr>
<td>WLGFR1</td>
<td>0.30</td>
<td>Weight ratio of front-to-rear landing gear for C.G. calculations.</td>
</tr>
<tr>
<td>XFAMMUN</td>
<td>0.30</td>
<td>C.G. position of the ammunition in percent BODL (fuselage length).</td>
</tr>
<tr>
<td>XFAPU</td>
<td>0.95</td>
<td>C.G. position of A.P.U. in percent BODL.</td>
</tr>
<tr>
<td>XFBB2</td>
<td>0.50</td>
<td>C.G. position of advanced weapons in percent BODL.</td>
</tr>
<tr>
<td>XFBOD</td>
<td>0.50</td>
<td>C.G. position of body in percent BODL.</td>
</tr>
<tr>
<td>Symbol</td>
<td>Value</td>
<td>Description</td>
</tr>
<tr>
<td>----------</td>
<td>-------</td>
<td>-----------------------------------------------------------------------------</td>
</tr>
<tr>
<td>XFBOMB</td>
<td>0.50</td>
<td>C.G. position of bombs in percent BODL.</td>
</tr>
<tr>
<td>XFCAN</td>
<td>0.20</td>
<td>C.G. position of the canard in percent of the Mean Aerodynamic Chord from the leading edge.</td>
</tr>
<tr>
<td>XFCARGO</td>
<td>0.50</td>
<td>C.G. position of cargo in percent BODL.</td>
</tr>
<tr>
<td>XFCREW</td>
<td>0.10</td>
<td>C.G. position of crew in percent BODL.</td>
</tr>
<tr>
<td>XFELT</td>
<td>0.05</td>
<td>C.G. position of Avionics in percent BODL.</td>
</tr>
<tr>
<td>XFENG</td>
<td>0.50</td>
<td>C.G. position of the engine in percent of PODL (engine pod length).</td>
</tr>
<tr>
<td>XFEP</td>
<td>0.50</td>
<td>C.G. position of electrical system in percent BODL.</td>
</tr>
<tr>
<td>XFEX</td>
<td>0.50</td>
<td>C.G. position of fuselage external tank in percent BODL.</td>
</tr>
<tr>
<td>XFFFUS</td>
<td>0.50</td>
<td>C.G. position of fuselage fuel in percent BODL.</td>
</tr>
<tr>
<td>XFFS</td>
<td>0.60</td>
<td>C.G. position of fuel system in percent BODL.</td>
</tr>
<tr>
<td>XFFUR</td>
<td>0.50</td>
<td>C.G. position of furnishings in percent BODL.</td>
</tr>
<tr>
<td>XFFWG</td>
<td>0.50</td>
<td>C.G. position of the fuel in the wing in percent BODL.</td>
</tr>
<tr>
<td>XFHDP</td>
<td>0.50</td>
<td>C.G. position of the hydraulics and pneumatics in percent BODL.</td>
</tr>
<tr>
<td>XFHT</td>
<td>0.20</td>
<td>C.G. position of the horizontal tail in percent of the Mean Aerodynamic Chord from the leading edge.</td>
</tr>
<tr>
<td>XFINST</td>
<td>0.15</td>
<td>C.G. position of the instruments in percent BODL.</td>
</tr>
<tr>
<td>XLECN</td>
<td>0.0</td>
<td>Leading edge position of canard when both a canard and an aft tail are used (feet).</td>
</tr>
<tr>
<td>XFLIFTF</td>
<td>0.50</td>
<td>C.G. position of the lift-fan in percent BODL.</td>
</tr>
<tr>
<td>XFLGFRT</td>
<td>0.10</td>
<td>C.G. position of the nose gear in percent BODL.</td>
</tr>
<tr>
<td>XFLGR</td>
<td>0.55</td>
<td>C.G. position of the main gear in percent BODL.</td>
</tr>
<tr>
<td>XFMISS</td>
<td>0.50</td>
<td>C.G. position of the missiles in percent BODL.</td>
</tr>
<tr>
<td>XFNA</td>
<td>0.98</td>
<td>C.G. position of the nacelles in percent PODL.</td>
</tr>
<tr>
<td>Symbol</td>
<td>Value</td>
<td>Description</td>
</tr>
<tr>
<td>--------</td>
<td>-------</td>
<td>-------------</td>
</tr>
<tr>
<td>XFPA</td>
<td>0.55</td>
<td>C.G. position of passenger accommodations in percent BODL.</td>
</tr>
<tr>
<td>XFPAYL</td>
<td>0.50</td>
<td>C.G. position of any payload in percent BODL.</td>
</tr>
<tr>
<td>XFPIV</td>
<td>0.65</td>
<td>C.G. position of the control surface pivots in percent BODL.</td>
</tr>
<tr>
<td>XFSC</td>
<td>0.70</td>
<td>C.G. position of control surfaces in percent BODL.</td>
</tr>
<tr>
<td>XFVT</td>
<td>0.20</td>
<td>C.G. position of the vertical tail in percent of the Mean Aerodynamic Chord from the leading edge.</td>
</tr>
<tr>
<td>XFWG</td>
<td>0.15</td>
<td>C.G. position of the wing in percent of the Mean Aerodynamic Chord from the leading edge.</td>
</tr>
<tr>
<td>XT1</td>
<td>0.30</td>
<td>Horizontal position of the forward thrust vector in percent BODL.</td>
</tr>
<tr>
<td>XT2</td>
<td>0.60</td>
<td>Horizontal position of the aft thrust vector in percent BODL.</td>
</tr>
<tr>
<td>ZFD</td>
<td>0.00</td>
<td>Vertical distance of the aircraft centerline from the C.G. in percent BDMAX (maximum fuselage length).</td>
</tr>
<tr>
<td>ZFMG</td>
<td>2.00</td>
<td>Vertical distance of the main gear from the C.G. in percent BDMAX.</td>
</tr>
<tr>
<td>ZFT1</td>
<td>0.00</td>
<td>Vertical distance of the forward thrust vector from the C.G. in percent BDMAX.</td>
</tr>
<tr>
<td>ZFT2</td>
<td>0.00</td>
<td>Vertical distance of the aft thrust vector from the C.G. in percent BDMAX.</td>
</tr>
<tr>
<td>ZMU</td>
<td>0.02</td>
<td>Friction coefficient of the runway on the main gear.</td>
</tr>
<tr>
<td>ZRTCN1</td>
<td>0.0</td>
<td>Height of the canard quarter-chord above the aircraft centerline (feet).</td>
</tr>
</tbody>
</table>
---Integer Format---------------------

<table>
<thead>
<tr>
<th>Variable</th>
<th>Value</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>IAS</td>
<td>0</td>
<td>Trajectory mission phase in which ammunition is used.</td>
</tr>
<tr>
<td>IBS</td>
<td>0</td>
<td>Trajectory mission phase in which bombs are dropped.</td>
</tr>
<tr>
<td>ICGPRT</td>
<td>0</td>
<td>Print flag to output aircraft component C.G. positions for C.G. analysis.</td>
</tr>
<tr>
<td>IDBPRT</td>
<td>0</td>
<td>Print flag to output a greater amount of information during each mission phase, used for debugging purposes.</td>
</tr>
<tr>
<td>IETANK</td>
<td>0</td>
<td>Used to specify various positions of any external tanks on the aircraft.</td>
</tr>
<tr>
<td></td>
<td>0</td>
<td>a single tank at the fuselage centerline.</td>
</tr>
<tr>
<td></td>
<td>1</td>
<td>Two tanks mounted on the sides of the fuselage.</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>Two tanks mounted on the wing quarter-chord.</td>
</tr>
<tr>
<td></td>
<td>3</td>
<td>Two wing-tip tanks.</td>
</tr>
<tr>
<td></td>
<td>4</td>
<td>One fuselage centerline tank and two tanks mounted at the wing quarter-chord.</td>
</tr>
<tr>
<td></td>
<td>5</td>
<td>One fuselage centerline tank and two wingtip tanks.</td>
</tr>
<tr>
<td></td>
<td>6</td>
<td>Two wing quarter-chord and two wingtip tanks.</td>
</tr>
<tr>
<td>IMS</td>
<td>0</td>
<td>Trajectory phase in which missiles are used.</td>
</tr>
<tr>
<td>IVECT</td>
<td>0</td>
<td>Used to specify which vectored thrust parameter is to be solved for.</td>
</tr>
<tr>
<td></td>
<td>0</td>
<td>No vectored thrust variable is to be solved for.</td>
</tr>
<tr>
<td></td>
<td>1</td>
<td>Calculate longitudinal position of the forward thrust vector.</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>Solve for the Forward over aft thrust ratio (Thrust Split).</td>
</tr>
<tr>
<td></td>
<td>3</td>
<td>Solve for the angle of the aft thrust vector from the horizontal axis.</td>
</tr>
<tr>
<td>NPHASE</td>
<td>1</td>
<td>Total number of trajectory phases.</td>
</tr>
</tbody>
</table>
APPENDIX B

Sample Output
### AIRCRAFT COMPONENT WEIGHTS AND POSITIONS

<table>
<thead>
<tr>
<th>COMPONENT</th>
<th>WEIGHT</th>
<th>POSITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>AIRFRAME</td>
<td>17022.54922</td>
<td>42.939999</td>
</tr>
<tr>
<td>WING</td>
<td>17129.294922</td>
<td>64.515587</td>
</tr>
<tr>
<td>CANARD</td>
<td>0.000000</td>
<td>0.000000</td>
</tr>
<tr>
<td>HT</td>
<td>1581.655640</td>
<td>114.010048</td>
</tr>
<tr>
<td>VT</td>
<td>1310.330078</td>
<td>107.414925</td>
</tr>
<tr>
<td>NOSE GEAR</td>
<td>1223.618774</td>
<td>11.300000</td>
</tr>
<tr>
<td>MAIN GEAR</td>
<td>2855.110352</td>
<td>70.512001</td>
</tr>
<tr>
<td>NACELLS</td>
<td>1503.169800</td>
<td>45.200001</td>
</tr>
<tr>
<td>PIVOTS</td>
<td>0.001953</td>
<td>45.200001</td>
</tr>
<tr>
<td>AIR COND.</td>
<td>309.482300</td>
<td>44.365093</td>
</tr>
<tr>
<td>APU</td>
<td>605.342590</td>
<td>70.059998</td>
</tr>
<tr>
<td>AVIONICS</td>
<td>1441.599487</td>
<td>1.130000</td>
</tr>
<tr>
<td>ELECTRICAL</td>
<td>1902.595215</td>
<td>22.600000</td>
</tr>
<tr>
<td>INSTRUMENTS</td>
<td>796.524597</td>
<td>5.650000</td>
</tr>
<tr>
<td>HYDRAULICS</td>
<td>563.688660</td>
<td>56.500000</td>
</tr>
<tr>
<td>CONTROL SURF</td>
<td>2981.909424</td>
<td>56.500000</td>
</tr>
<tr>
<td>PA</td>
<td>0.000000</td>
<td>45.200001</td>
</tr>
<tr>
<td>FURNISHINGS</td>
<td>450.000000</td>
<td>36.160000</td>
</tr>
<tr>
<td>ENGINES</td>
<td>6954.871582</td>
<td>91.914940</td>
</tr>
<tr>
<td>FUEL SYS.</td>
<td>1682.630249</td>
<td>33.900002</td>
</tr>
<tr>
<td>LIFT FAN</td>
<td>0.000610</td>
<td>56.500000</td>
</tr>
<tr>
<td>PAYLOAD</td>
<td>60.000000</td>
<td>22.600000</td>
</tr>
<tr>
<td>CREW</td>
<td>510.000000</td>
<td>11.300000</td>
</tr>
<tr>
<td>CARGO</td>
<td>0.000000</td>
<td>33.900002</td>
</tr>
<tr>
<td>AMMUNITION</td>
<td>0.000000</td>
<td>33.900002</td>
</tr>
<tr>
<td>BB2</td>
<td>0.000000</td>
<td>56.500000</td>
</tr>
<tr>
<td>BOMBS</td>
<td>0.000000</td>
<td>56.500000</td>
</tr>
<tr>
<td>MISSILES</td>
<td>0.000000</td>
<td>53.110001</td>
</tr>
<tr>
<td>FUS. FUEL</td>
<td>0.000000</td>
<td>56.500000</td>
</tr>
<tr>
<td>WING FUEL</td>
<td>16347.398</td>
<td>64.235992</td>
</tr>
</tbody>
</table>

**MISSION PHASE NUMBER  1**

XCG = 64.0515594
XAC = 64.0994263
DCMDCL = -0.0539999
NEUTPT = 0.4538571
STATIC MARGIN = 11.7050486
AFT CG = 65.6878738
SUMMO = 5729981.000000  SUMWT = 89458.882813
DCMDCLT = -0.05395812  DCMDCW = -0.00342405
DCMDCLF = 0.00338223  DCMDCCLN = 0.00000000
DCMDCDJ = 0.00000000
LWB = 33122.109375  LHT = -9019.528320  LCN = 0.0000000
SHT = 379.950439  SCN = 0.000000
XCGFRWD[ 1] = 61.9601

B727-200 Center of Gravity Output and Detailed Output
### STABILITY OUTPUT DATA

| Tail Size = | 379.9504395 sq. ft. |
| XCGFRWD = | 62.3969193 ft. |
| AFT CG = | 65.0767975 ft. |

### DATA AT END OF TAKE-OFF

| XAC = | 63.1220779 ft. |
| NEUTPT = | 0.4401450 |
| XCG = | 64.60528641 ft. |
| CGBAR = | 33.6899948 % chord |
| DCMDC = | 0.0047093 |
| STATIC MARGIN = | 7.3245034 % chord |

#### MISSION PHASE NUMBER 1

| XAC = | 64.0094263 ft. |
| NEUTPT = | 0.4538571 |
| XCG = | 64.0515594 ft. |
| CGBAR = | 33.6806602 % chord |
| DCMDC = | 0.00539999 |
| STATIC MARGIN = | 7.3338361 % chord |

#### MISSION PHASE NUMBER 2

| XAC = | 64.9836954 ft. |
| NEUTPT = | 0.4740317 |
| XCG = | 64.0514832 ft. |
| CGBAR = | 33.6801144 % chord |
| DCMDC = | 0.00784065 |
| STATIC MARGIN = | 7.3343816 % chord |

#### MISSION PHASE NUMBER 3

| XAC = | 64.1274567 ft. |
| NEUTPT = | 0.4552816 |
| XCG = | 64.0369263 ft. |
| CGBAR = | 33.5759850 % chord |
| DCMDC = | 0.0568189 |
| STATIC MARGIN = | 7.4385114 % chord |

#### MISSION PHASE NUMBER 4

| XAC = | 64.5148773 ft. |
| NEUTPT = | 0.4756619 |
| XCG = | 64.0367661 ft. |
| CGBAR = | 33.5748367 % chord |
| DCMDC = | 0.0814076 |
| STATIC MARGIN = | 7.4396572 % chord |

#### MISSION PHASE NUMBER 5

| XAC = | 64.2071304 ft. |
| NEUTPT = | 0.4593702 |
| XCG = | 64.0241013 ft. |
| CGBAR = | 33.4842453 % chord |
| DCMDC = | 0.0627594 |
| STATIC MARGIN = | 7.5302525 % chord |

END STABILITY OUTPUT
APPENDIX C

Production Aircraft Layouts
Figure 23: F-16A Geometric Layout
(Reproduced from Reference 9)

Figure 24: Saab J-37 Viggen Geometric Layout
(Reproduced from Reference 9)
Figure 25: B727-200 Geometric Layout

(Reproduced from Reference 2)
Figure 26: B727-200 Aerodynamic Data

(Reproduced from Reference 7)
APPENDIX D

Program Listing
SUBROUTINE STBLTY

THIS SUBROUTINE IS USED IN CONJUNCTION WITH NASA AMES AIRCRAFT DESIGN
PROGRAM 'ACSYNT'. THIS PROGRAM CALCULATES THE SIZE OF THE HORIZONTAL
CONTROL SURFACE FOR EITHER AN AFT TAIL OR A FORWARD CANARD. IT CAN
ALSO HANDLE AN AIRCRAFT WITH BOTH CONTROLS, BUT IT ONLY SIZES THE AFT
TAIL, THE SIZE OF THE CANARD IS AN INPUT FOR THE USER.

THIS SUBROUTINE ALSO CALCULATES THE CENTER OF GRAVITY OF THE AIRCRAFT,
THE CENTER OF GRAVITY SHIFT DURING THE TRAJECTORY MISSION, AND THE
FORWARD AND AFT CENTER OF GRAVITY LIMITS IMPOSED BY THE SIZE OF THE
HORIZONTAL CONTROL SURFACE.

IT ALSO DETERMINES THE PITCHING MOMENT CURVE SLOPE (DCM/DCL) OF THE
AIRCRAFT FOR EACH PHASE IN THE MISSION.

THE REFERENCES SITED THROUGHOUT THE PROGRAM ARE LISTED AT THE END
OF THE LISTING.

STEPHEN SWANSON 4/89

THE FILE 'STBCM.INC' MUST BE IN THE SAME DIRECTORY AS THIS MODULE
FOR IT TO WORK. THE FILE CONTAINS ALL THE COMMON BLOCK STATEMENTS
NEEDED TO RUN THIS MODULE. THE 'INCLUDE' STATEMENT IS USED TO
INCLUDE THE FILE IN WITH THE MODULE DURING COMPILATION. S.M.S. 1/89

THE FOLLOWING IS A LISTING OF THE STBCM.INC FILE......

• THIS CONTAINS THE COMMON BLOCK STATEMENTS NEEDED TO RUN THE STABILITY
AND CONTROL MODULE 'STBLCON.FOR'. THIS FILE IS CONNECTED WITH THE
MODULE THROUGH THE 'INCLUDE' STATEMENT FOUND AT THE BEGINING OF EACH
OF THE SUBROUTINES.

COMMON/OVER/ICALC,NERROR,MODNO,IGEO,KGPRT,IOPLT,IF

COMMON/STBCM/CLIFHT,DEDA,XQCBNC,XQCBWG,XQCBHT,XQCBVT,
1 CBARCN,CBARGW,CBART,CLAVRT,ROOTW,TRWG,XLEVG,SPANWG,
2 SWPGP,WNL,EN,BOD,TLELEG(10),WACAND,WFTOT,FWFrac,WMINC,WFG,
3 WLG,WHT,WVT,WETANK,WFEXT,WBODY,MAIR,WAPU,WSC,WE,WSCN,SWC,SHT,
4 SWT,WCREW,WEL,WEP,WISP,WINS,WA,WFPA,WFACO,WAMMUN,WMISS,
5 ARWG,DHED,WRTW,ARVT,SWPCN,TRCN,AMTD,
6 ZRTCN,VDLB,BOBM,FRN,FRAB,ALPHA,CL,THRUST,ARTH,TNT(12),WFT(12),
7 WFTD,WFTC,ARCW,CMWG,WTG,WFNB,WEBB,WAF,WF,WPS,WS,EFOM,WFUEL,
8 TWD,MCW,ALT,SWT,WST,WSTC,WSTN,SPANHT,SPANC,
9 STARTM(12),DUMMY,IAO

COMMON/LOCALS/ETAHT,XFCAN,XFAN,XFAT,XFACF,XFNC, XFENG,
1 XFEK,XFEP,XFINST,XFHDP,XPFA,XPAYL,XFUS,XFCREW,XPAYL,XFUR,
2 XFLGFT,WLGFT,XFLC,XXS,XXFGBW,XFSC,XXFIV,XXFLFT,WFIV,WFUR,
3 XEF1,XFAMNC,XXBB2,XFMISS,XXCORTH,XACNG,ETANC,ODEDA,XAMW,CLACN,
4 WZET,WEF,WEF,WFF,WIF,WZENG,WMADD,SWMAD,XXLE,WL,WFUG,
5 DIFPRT,XSC(13),XCMDFC(13),SM(13),XGAM(13),WLGFT,WGFT,WFG,
6 WFP,WZB,WZM,WZT,IBS,IBS,IFS,IPSR,ICGPR,TLFT,WFFL,WFP,LHT,LCN,
7 CELV,CBALT,TLFT,NEUTPT(13),XFBOD,XXFGX,XFEX,XXLFT,IXETK,
8 ZMTOR,AWT,JUM,XT1,XT2,UFMC,SMG,2FT1,2FT2,2FT3,2FT,
9 TPLIT,GMAT1,GMAT2,HTT,ICN,ELVDEF,VRD,ALPHA,CAU,T2,
10 SHT1,SCN1,ELVDMAX,IWG,XXGFRW(13),QTOR,CSBGR(13),ALFT0,CLTO,
INCLUDE 'STBCM.INC'

IF (ICALC.GT.1.) GO TO 10
READ IN THE INPUT PARAMETERS.
CALL STABINPT

10 CONTINUE

IF (ICALC.NE.2.) GO TO 20
DO ALL THE NECESSARY CALCULATIONS.
CALL STABCALC1

20 CONTINUE

IF (ICALC.NE.3.) GO TO 30
DO A FINAL PRINTOUT OF THE DATA.
CALL.Stdout

30 CONTINUE

RETURN

END

SUBROUTINE STABINPT

THIS SUBROUTINE READS IN THE INPUT FROM THE INPUT FILE CREATED BY
THE USER IN NAMELIST 'STABIN'. IT ALSO INITIALIZES ALL VALUES SO
THE USER DOES NOT NEED TO SPECIFY ALL VALUES, ONLY THE ONES DESIRED

INCLUDE 'STBCM.INC'

NAMELIST/STABIN/ETAHT, XFCAN, XFHT, XFN A, XVF T, XFWG, XFAPU, XFB2,
1 XFPIV, XFELT, XFEP, XFINST, XFDP, XFA, XFFUS, XFB0D, XFFW G, XFCARGO,
2 XFLGFR T, WFLGFR T, XFLOR, XFFUR, XFM ISS, XFA M M UN, XFPAYL, XFE X, XFFS,
THE FOLLOWING IS A LIST OF THE INPUTS AND WHAT THEY STAND FOR...

- ETAHT - AFT HORIZONTAL TAIL EFFICIENCY
- ETACN - FORWARD CANARD EFFICIENCY
- XFBOD - FUSELAGE BODY C.G. POSITION IN PERCENT BODL
- XFCAN - FORWARD CANARD C.G. POSITION IN PERCENT MAC
- XFHT - AFT HORIZONTAL TAIL C.G. POSITION IN PERCENT MAC
- XFN - NACELL(S) C.G. POSITION IN PERCENT BODL
- XFT - VERTICAL TAIL C.G. POSITION IN PERCENT MAC
- XFW - WING C.G. POSITION IN PERCENT MAC
- XFLGRT - NOSE GEAR C.G. POSITION IN PERCENT BODL
- WLQRT - PERCENT NOSE GEAR OF TOTAL LANDING WEIGHT
- XFLGR - MAIN LANDING GEAR C.G. POSITION IN PERCENT BODL
- XFP - CONTROL PIVOTS C.G. POSITION IN PERCENT BODL
- XFUR - FURNISHINGS C.G. POSITION IN PERCENT BODL
- XAP - APU C.G. POSITION IN PERCENT BODL
- XFE - AVIONICS C.G. POSITION IN PERCENT BODL
- XFPN - ELECTRICAL SYSTEMS C.G. POSITION IN PERCENT BODL
- XFINST - INSTRUMENTS C.G. POSITION IN PERCENT BODL
- XFFS - FUEL SYSTEM C.G. POSITION IN PERCENT BODL
- XFD - HYDRAULIC AND PNEUMATIC C.G. POSITIONS IN PERCENT BODL
- XFC - SURFACE CONTROLS C.G. POSITION IN PERCENT BODL
- XFP - PASSENGER ACCOMODATIONS C.G. POSITION IN PERCENT BODL
- XFPAYL - PAYLOAD C.G. POSITION IN PERCENT BODL
- XFCARGO - CARGO C.G. POSITION IN PERCENT BODL
- XFFWG - FUEL IN WING C.G. POSITION IN PERCENT BODL
- XFFUS - FUEL IN FUSELAGE C.G. POSITION IN PERCENT BODL
- XFX - EXTERNAL TANKS C.G. POSITION IN PERCENT BODL
- XFCREW - CREW C.G. POSITION IN PERCENT BODL
- XFENG - ENGINE(S) C.G. POSITION IN PERCENT PODLength
- XFLF - LIFT FAN C.G. POSITION IN PERCENT BODL
- XFBOMB - BOMBS C.G. POSITION IN PERCENT BODL
- XFB2 - ADVANCED WEAPONS SYSTEMS C.G. POSITION IN PERCENT BODL
- XFMIS - MISSILES C.G. POSITION IN PERCENT BODL
- XFAAMUN - AMMUNITION C.G. POSITION IN PERCENT BODL
- IDBBRT - DEBUG PRINT FLAG, CREATES MORE OUTPUT
- CEB - PERCENT OF STABILIZER WHICH IS MOBILE ELEVATOR
- CBALHT - PERCENT OF ELEVATOR WHICH IS FORWARD OF THE HINGE LINE
- TZN - THICKNESS TO CHORD RATIO OF THE ELEVATOR
- ZM - ANGLE OF ATTACK OF THE WING AT TAKE OFF ROTATION
- ZT1 - HORIZONTAL POSITION OF FORWARD THRUST VECTOR PERCENT BODL
- ZT2 - HORIZONTAL POSITION OF AFT THRUST VECTOR IN PERCENT BODL
- ZFMG - VERT. DIST. FROM FUSS. CENTERLINE TO MAIN GEAR, % BDMAX
- ZFT1 - VERT. DIST. FROM FUSS. CENTERLINE TO FORE THRUST, %BDMAX
- ZFT2 - VERT. DIST. FROM FUSS. CENTERLINE TO AFT THRUST, %BDMAX
- ZFD - VERT. DIST. FROM FUSS. CENTERLINE TO DRAG VECTOR, %BDMAX
- TSPLII - FORWARD THRUST TO TOTAL THRUST RATIO IN PERCENT
- GAMAT1 - FORWARD THRUST VECTOR ANGLE, DEGREES
C CAMAT2  - AFT THRUST VECTOR ANGLE, DEGREES
C SCNI  - CANARD AREA USED WHEN A.C. HAS A AFT TAIL ALSO, SQ. FT.
C XLECN  - CANARD LEADING EDGE POSITION USED WITH AFT TAIL, FEET
C SPANCN1  - CANARD SPAN USED WITH AFT TAIL, FEET
C ZRTCNI  - VERTICAL CANARD HEIGHT USED WITH AFT TAIL, FEET
C TRCN1  - TAPER RATIO OF CANARD USED WITH AFT TAIL
C IHT  - AFT TAIL INCEDENCENCE ANGLE, DEGREES
C ICN  - FORWARD CANARD INCEDENCENCE ANGLE, DEGREES
C ELVDEF  - ELEVATOR ANGLE NEEDED FOR TAKE OFF ROTATION, DEGREES
C ELVDMAX  - MAXIMUM ELEVATOR DEFLECTION FOR LANDING, DEGREES
C VROT  - MACH NUMBER AT TAKEOFF ROTATION
C IETANK  - EXTERNAL TANK FLAG, USED TO SPECIFY POSITIONS OF
C ANY EXTERNAL TANKS. 0=TANK ON FUSS. CENTERLINE,1=TW
C FUSS. SIDE MOUNTED TANKS, 2=TW MAC MOUNTED TANKS,
C 3=TW MAC MOUNTED TANKS AND TWO
C WING-TIP TANKS, 4=ONE CENTERLINE TANK AND TWO
C MAC MOUNTED TANKS, 5=CENTERTLINE TANK AND WING-TIP TANKS,
C 6=TW MAC TANKS AND TWO WING-TIP TANKS.
C IBS  - INDICATOR TO REMOVE BOMB WEIGHT ON A SPECIFIED MISSON PHASE
C IMS  - INDICATOR TO REMOVE MISSLE WEIGHT ON SPECIFIED PHASE
C IAS  - INDICATOR TO REMOVE AMMUNITION WEIGHT ON SPECIFIED PHASE
C NPHASE  - NUMBER OF PHASES IN MISSION
C ICGPRT  - PRINT FLAG TO PRINT C.G. INFORMATION FOR EACH PHASE
C IVECT  - VECTORED THRUST INDICATOR, 0= SOLVE FOR NO VECTORED
C THRUST PARAMETERS, 1=SOLVE FOR FORWARD THRUST
C POSTION, 2=SOLVE FOR THRUST SPLIT, 3=SOLVE FOR AFT
C THRUST VECTOR ANGLE.

INITIALIZE THE STABIN INPUT VALUES

ETAHT = 0.90
ETACN = 0.90
XFBD = .50
XFCAN = .20
XFHT = .20
XFNA = .40
XFT = .20
XFNC = .15
XLFGR = .10
WLFGR = .3
XLFGR = .55
XFP = .65
XFFUR = .50
XFPUR = .95
XFEU = .95
XFE = .50
XFIN = .15
XFSS = .60
XFHDP = .50
XFSC = .70
XFP = .55
XFP = .50
XFSC = .50
XFCARGO = .50
XFFGC = .50
READ IN THE VALUES FROM THE NAMELIST

IF (ICALC .GT. 1.) GO TO 200
READ(5,40) TITLE
READ(5,STABIN)

OUTPUT THE INPUTS TO THE OUTPUT FILE SO THE USER CAN DOUBLE
CHECK THAT THE VALUES INPUT ARE THOSE DESIRED.

WRITE(6,300)
WRITE(6,310) ETAHT, XFBOD, XFCAN
WRITE(6,315) ETACN, WFLGFR
WRITE(6,320)XFHT,XFNA,XFVT
WRITE(6,330)XFNC,XFLGFRT,XFLGR
WRITE(6,340)XFPIV,XFFUR,XFAPU
WRITE(6,350)XFPIV,XFFUR,XFINST
WRITE(6,360)XFTS,XFHDPE, XFSC
WRITE(6,370)XFPA,XFPAYL,XFFWG
WRITE(6,380)XFFUS,XFEX,XFCREW
WRITE(6,390)XFCN,F,LIFTF
WRITE(6,400)XFCAMUN,XFBOMB,XFMISS
WRITE(6,405)XFCARGO,XFB2
WRITE(6,410)XFLT, CBALHT,TZCHT
WRITE(6,411)AWOT, ZMU
WRITE(6,412)XT1,XT2,ZFGC
WRITE(6,413)ZFT1, ZFT2, ZFD
WRITE(6,414)TSPLIT, GAMAT1, GAMAT2
WRITE(6,415)IHT, ICON, ELVDEF
WRITE(6,416)VROR, SPANCN1, ZRTC1
WRITE(6,417)SCN1, XLECN, ELVMAX
WRITE(6,418)TRC1, IVECT
WRITE(6,420)IETANK, IBS, IMS, IAS
WRITE(6,430)NPHASE, IDBPRT, ICGRPRT
CONTINUE
FORMAT(20A4)
FORMAT(/,1H11,15X, 28HSTABILITY AND CONTROL INPUTS,//)
FORMAT(10X,7HETHTA=,F10.5,5X,7HXFBO=,F10.5,5X,7HXFCAN=,F10.5)
FORMAT(10X,7HETACN=,F10.5,5X,8HXF walnut=,F10.5)
FORMAT(10X,7HXFHT=,F10.5,5X,8HXFNA=,F10.5,5X,7HXFVT=,F10.5)
FORMAT(10X,7HXFWG=,F10.5,5X,8HXFLGFRT=,F10.5,5X,8HXFLGR=,F10.5)
FORMAT(10X,7HXFPIV=,F10.5,5X,8HXFFUR=,F10.5,5X,7HXFINST=,F10.5)
FORMAT(10X,7HXELT=,F10.5,5X,8HXFEP=,F10.5,5X,7HXFINST=,F10.5)
FORMAT(10X,7HXFFS=,F10.5,5X,8HXFHDP=,F10.5,5X,7HXFSC=,F10.5)
FORMAT(10X,7HXFFUS=,F10.5,5X,8HXFFWG=,F10.5)
FORMAT(10X,7HXFENG=,F10.5,5X,7HXFCREW=,F10.5)
FORMAT(10X,7HXFAMUN=,F10.5,5X,8HXFBOMB=,F10.5,5X,7HXFMASS=,F10.5)
FORMAT(10X,7HXFAMUN=,F10.5,5X,8HXFB2=,F10.5)
FORMAT(10X,7HCELV=,F10.5,5X,8HCBALHT=,F10.5,5X,7HTEZCHT=,F10.5)
FORMAT(10X,8HAWTO=,F10.5,5X,7HZMU=,F10.5)
FORMAT(10X,7HXFT1=,F10.5,5X,8HXT2=,F10.5,5X,7HZFMG=,F10.5)
FORMAT(10X,7HFT2=,F10.5,5X,7HZFD=,F10.5)
FORMAT(10X,7HGAMAT1=,F10.5,5X,7HGAMAT2=,F10.5)
FORMAT(10X,7HICN=,F10.5,5X,7HELDEF=,F10.5)
FORMAT(10X,7HVECT=,F10.5,5X,8HVECT=,F10.5)
FORMAT(10X,7HICNPRT=,F10.5,5X,8HICPRT=,F10.5)
CONTINUE
RETURN
END
SUBROUTINE STABCALC1

C THIS SUBROUTINE DETERMINES THE WEIGHTS OF FUEL AND WEAPONS ON THE
C AIRCRAFT, AND REMOVES THOSE WEIGHTS THAT HAVE BEEN USED DURING
C EACH PHASE OF THE MISSION AS SPECIFIED IN THE TRAJECTORY MODULE.
C IT THEN CALLS THE DIFFERENT SUBROUTINES USED FOR CALCULATING
C THE DIFFERENT PARAMETERS AND SIZING THE TAIL.
C
****** NOTE: NPHASE, IBS, IAS, AND IMS, SHOULD BE SET IN THE STABILITY
C INPUTS TO THE DESIRED VALUES TO ACCOUNT FOR THE NUMBER OF MISSION
C PHASES AND THE WEIGHT LOSSES DUE TO THE USE OF WEAPONS.
C
INCLUDE 'STBCM.INC'

C IFLAG = 0
CGFLAG = 0
LCFLAG = 0
CONTINUE

I = 1
IPHASE = 0
XCGFRWD(1) = 0
XCGAFT(2) = BOOL

C SAVES THE ACSYNT CALCULATED WEIGHTS IN LOCAL VARIABLES AND
C SETS THE FUEL WEIGHT TO THE MAXIMUM VALUE FOR TAKE-OFF
C CONDITIONS.
WFP = WFTOT
WZB = WBOMB
WZM = WMISS
WZA = WAMMUN
WZET = WETANK
WZEF = WFEXT

C SIZE THE TAIL FOR TAKE-OFF ROTATION IF IT HAS NOT ALREADY BEEN DONE.
IF(IFLAG .EQ. 0 .AND. CGFLAG .EQ. 0) THEN
C
CALL TAILSIZE(I,IPHASE)
C
ENDIF
IFLAG = 0
CGFLAG = 0

C REMOVE THE FUEL WEIGHT USED IN TAKE-OFF AND MAKE FIRST CALCULATIONS.
C CHECK TO SEE IF THE FUEL USED FOR TAKE-OFF IS GREATER THAN THE FUEL
C IN THE EXTERNAL TANKS. IF IT IS THE TANKS ARE REMOVED.
C
I = 2
IF(WFEXT .GT. 0.) THEN
WFP = WFP - WFEXT
IF (WFTO .GE. WFEXT) THEN
  WZET = 0.
  WZEF = 0.
  WFP = WFP - (WFTO-WFEXT)
ELSE
  WZEF = WFEXT - WFT0
ELSE
  WFP = WFP - WFT0
ENDIF

CALL STABCALC2(I,IPHASE,CGFLAG)
IF (CGFLAG .EQ. 1) THEN
  GO TO 112
ENDIF

C
C REMOVE THE FUEL WEIGHT USED IN EACH PHASE, AND REMOVE ANY WEAPONS
C USED DURING EACH PHASE. ALSO REMOVE EXTERNAL TANKS WHEN THE FUEL
C USAGE BECOMES GREATER THEN THE SIZE OF THE EXTERNAL TANKS.

DO 100 IPHASE = 1,NPHASE
  IF (WZEF .GT. 0.) THEN
    IF (WFT(IPHASE) .GE. WZEF) THEN
      WFP = WFP - (WFT(IPHASE) - WZEF)
      WZET = 0.
      WZEF = 0.
    ELSE
      WZEF = WZEF - WFT(IPHASE)
    ENDIF
  ELSE
    WFP = WFP - WFT(IPHASE)
  ENDIF
ELSE
  IF (IPHASE.GE.IBS) WZB = 0.
  IF (IPHASE.GE.IWS) WZM = 0.
  IF (IPHASE.GE.IAS) WZA = 0.
I = IPHASE + 2

CALL STABCALC2(I,IPHASE,CGFLAG)
IF (CGFLAG .EQ. 1) THEN
  GO TO 112
ENDIF

C
C CALCULATE THE FORWARD C.G. LIMIT FOR THE TAIL SIZE DETERMINED EARLIER.
C
CALL XCGFORWARD(I)

C
C THESE SET OF INSTRUCTIONS DETERMINE IF THE CENTER OF GRAVITY IS
C BEHIND THE FORWARD C.G. LIMIT DETERMINED, AND IF IT IS NOT, THEN IT INCREASES
C THE SIZE OF THE HORIZONTAL CONTROL SURFACE AND RECALCULATES THE
C FORWARD C.G. LIMIT.
C
IF (XCGFRWD(I) .GT. XCG(I)) THEN
  WRITE(6,300) XCG(I), XCGFRWD(I)
  IF (SCN .EQ. 0.0) THEN
    SHT1 = SHT1 + 5.0
    IFLAG = 1
  ELSE
    SCN1 = SCN1 + 5.0
    IFLAG = 1
  ENDIF
ENDIF
ENDIF
GO TO 112
ENDIF

C
100 CONTINUE
C
THIS SET OF INSTRUCTIONS ACTIVATES THE VECTORED THRUST ANALYSIS
IF IVECT IS NOT SET TO 0.
C
IF(IVECT .EQ. 0) GO TO 200
I = NPHASE + 2
CALL VECTHRUST(I)
200 CONTINUE
C
300 FORMAT(/,1X,35HWARNINO!! C.G. IS FORWARD OF LIMITS.,/,15X,
139H CONTROL SURFACE INCREASED BY 5 SQ. FT. ,/,15X,6HXCG = ,F12.6,
25X,10HXCFGFRWD = ,F12.6)
RETURN
END
C
C
SUBROUTINE TAILSIZE(I,IPHASE)
C
THIS SUBROUTINE IS USED TO CALL THOSE ROUTINES NEEDED TO SIZE THE
HORIZONTAL TAIL AT TAKEOFF.
C
INCLUDE 'STBCM.INC'
C
DATATRANS SUBROUTINE TRANSFERS OVER THE NECESSARY AERODYNAMICS
PARAMETERS.
C
CALL DATATRANS
C
THIS SUBROUTINE DOES ALL THE SECONDARY AERODYNAMICS CALCULATIONS.
C
CALL STABCALC3(I,IPHASE)
C
THIS SUBROUTINE DETERMINES THE TAIL SIZE NEEDED FOR ROTATION.
C
CALL TOROTATE(I,IPHASE)
C
RETURN
END
C
C
SUBROUTINE STABCALC2(I,IPHASE,COFLAG)
C
THIS SUBROUTINE IS USED TO CALCULATE ANY PARAMETERS NEEDED
BY THE SUBROUTINES WHICH ARE NOT CALCULATED BY THE REST OF ACSYNT.
C
INCLUDE 'STBCM.INC'
C
THIS SERIES OF INSTRUCTIONS SETS THE MACH NUMBER EITHER TO THE TAKE OFF MACH NUMBER OR THE MACH NUMBER AT THE START OF A MISSION PHASE. IT THEN CALLS AERODYNAMICS TO GET THE NEW CLALPHA OF THE WING AND THEN CALLS THE AERODYNAMIC CENTER SUBROUTINE.

IF(IPHASE .EQ. 0) THEN
   IF(AMTO .LE. 0.0) THEN
      AMTO = VROT
   ENDIF
   MACH = AMTO
ELSE
   MACH = STARTM(IPHASE)
ENDIF
ICALC = 2
IAO = 1
CALL STBDT(ICALC,NERR,1GE0,KOPRNT,IGPLT)
ZCLALFA = CLALFA
ZMACH = MACH

NOW CALCULATE THE AERODYNAMIC CENTER INCORPORATING MACH EFFECTS, SWEEP ANGLE EFFECTS, AND TAPER EFFECTS.

CALL XACCALC(ZMACH,ZCLALFA)

NOW CALCULATE THE CENTER OF GRAVITY OF THE AIRCRAFT.

CALL XCCCALC(I)

CONTINUE

THE FOLLOWING SERIES OF COMMANDS CALCULATE THE PITCHING MOMENT CURVE SLOPE FOR THE AIRCRAFT. THESE CALCULATIONS ARE BASED ON EQUATIONS TAKEN FROM REFERENCE 4, CHAPTER 5. THEY INCLUDE THE TAIL, WING, CANARD, FUSELAGE, AND JET(S).

**TAIL CONTRIBUTION TO AIRCRAFT STABILITY

CSC=CLALFA*SWG*CBARWG
DCMDCL1=-CLAFHT*SHT1*({XQCBHT-XCG(I)})*ETAHT*(1.-DEDA)/CSC

THIS ACCOUNTS FOR STICK FREE EFFECTS, DETERMINED IN STABCALC3.

DCMDCLT=DCMDCL1*DCMTMUL

**CALCULATE THE PITCHING MOMENT SLOPE CONTRIBUTION DUE TO THE CANARD. NOTE THAT THERE ARE TWO SETS OF COMMANDS FOR THE CANARD, ONE FOR A CANARD ALONE, AND ONE FOR A CANARD AND AFT TAIL CONFIGURATION

IF(XLECN .EQ. 0.0) THEN
   VOLCN = (SCN1/SWG)*((XCG(I)-XLECN)/CBARWG)
ELSE
   VOLCN = (SCN1/SWG)*((XCG(I)-XLECN)/CBARWG)
ENDIF
DCMCLCN = (CLACN/CLALFA) * VOLCN * ETA CN * ODEDA

COMPUTE THE STICK FREE DCMDCI. OF THE CANARD IF THERE IS NO HORIZONTAL STABILIZER, OTHERWISE THE H.T. IS ASSUMED TO BE FREE AND THE CANARD IS USED ONLY FOR TRIM.

IF (SCN .GT. 0. AND. SHT .LE. 0.) THEN
  DCMCMUL = DCMTMUL
  DCMCLCN = DCMCLCN * DCMCMUL
ELSE
  DCMCMUL = 1.
ENDIF

** WING CONTRIBUTION TO AIRCRAFT STABILITY
DCMDCIW = ((XCG(I) - XLE)/CBARWG) - ((XACWG - XLE)/CBARWG)

** FUSELAGE CONTRIBUTION TO STABILITY
THIS USES THE EASY METHOD FROM REFERENCE 4 WHICH CURVE-FITS THE GRAPH OF KF (FIGURE 5-16) TO DETERMINE THE FUSELAGE EFFECTS.

KF = ((XLEWG + ROOTWG/4.) / BODL) * 100.
KF = 0.0023 * 0.000005762 * KF + 0.0000002499 * KF ** 3

CLARAD = CLALFA * 57.29578
DCMDCLF = (KF * (BDMAX * 2) * BODL) / (SWG * CBARWG * CLARAD)

CALCULATE THE AERODYNAMIC CENTER FOR THE WING AND FUSELAGE BY ADDING THE EFFECTS OF THE FUSELAGE TO THE AERODYNAMIC CENTER OF THE WING.

XAWB = ((XACWG - XLE) / CBARWG) + DCMDCLF

** STABILITY EFFECTS DUE TO THE JET(S)
1; EFFECTS DUE TO THRUST
TN1 = TNT(IPHASE) * TSPLIT
TN2 = TNT(IPHASE) - TN1
DDT1 = (TN1 * ZT1) / (SUMWT * CBARWG)
DDT2 = (TN2 * ZT2) / (SUMWT * CBARWG)
DCMTH = DDT1 + DDT2

;; EFFECTS DUE TO NORMAL FORCE (THERE ARE NOT ANY YET.)
DCMNF = 0.

TOTAL JET EFFECTS
DCMDCLJ = DCMTH + DCMNF

** OVERALL MOMENT-CURVE SLOPE
THIS IS JUST A SUMMATION OF ALL THE PARTS.
DCMDCL(I) = DCMDCIW + DCMCLCN + DCMDCLF + DCMDCLT + DCMDCIJ

** AIRCRAFT STICK-FREE NEUTRAL POINT
THIS USES AN EQUATION FROM REFERENCE 2 (CHAPTER 5-EQN. 5.145) MODIFIED TO INCLUDE A CANARD. ASSUMES THRUST EFFECTS AND FUSELAGE EFFECTS DO NOT CHANGE WITH ANGLE OF ATTACK.

\[ CLHR = \frac{(CL_{AFHT}/CL_{ALFA}) \times (SHT/\text{SWG}) \times ETAHT}{(SHT \neq 0.0) \text{ THEN}} \]
\[ CNCR = \frac{(CL_{ACN}/CL_{ALFA}) \times (SCN/\text{SWG}) \times ETACN}{\text{ELSE}} \]

\[
\text{IF} (SHT \neq 0.0) \text{ THEN} \]
\[
FNEUPT(I) = (XAWB \times CLHR \times (XQCBHT/CBARGW) \times (1.-DEDA) \times DCMTMUL)/(1. + CLHR \times (1.-DEDA) \times DCMTMUL)
\]

\[
\text{ELSE} \]
\[
FNEUPT(I) = (XAWB \times CLCR \times (XQCBCN/CBARGV) \times ODED \times DCMCMUL)/(1. + CLCR \times ODED \times DCMCMUL)
\]

**CG AFT LIMIT**

THE AFT C.G. LIMIT IS THE NEUTRAL POINT OF THE AIRCRAFT.

\[ XCGAFT(I+1) = XLE + (FNEUPT(I) \times CBARWG) \]

**STATIC MARGIN**

THE STATIC MARGIN IS SIMPLY THE DIFFERENCE BETWEEN THE NEUTRAL POINT AND THE CENTER OF GRAVITY.

\[ SM(I) = ((XCGAFT(I+1) - XCG(I))/CBARWG) \times 100. \]

CHECK TO SEE IF THE C.G. FOR EACH PHASE IS AFT OF THE AFT C.G. LIMIT. IF IT IS, THEN THE TAIL SIZE IS INCREASED, AND THE ENTIRE SUBROUTINE RERUN.

\[
\text{IF}(XCGAFT(I+1) \lt XCG(I))\text{THEN} \]
\[
\text{WRITE}(6,300)XCG(I),XCGAFT(I+1)
\]
\[
\text{IF}(SCN \text{.EQ. 0.0})\text{THEN} \]
\[
SHT1 = SHT1 + 5.0
\]
\[
CGFLAG=1
\]
\[
\text{RETURN}
\]
\[
\text{ELSE} \]
\[
SCN1 = SCN1 + 5.0
\]
\[
CGFLAG=1
\]
\[
\text{RETURN}
\]
\[
\text{ENDIF}
\]
\[
\text{ENDIF}
\]

IF (IDBPRT.EQ.0) GO TO 400

THIS IS THE WRITE STATEMENTS TO BE USED WHEN COMPLETE DEBUGGING INFORMATION IS DESIRED. THIS IS STARTED BY SETTING IDBPRT TO 1.

\[
\text{IF}(IPHASE \text{.LE. 0})\text{THEN} \]
\[
\text{WRITE}(6,59)
\]
\[
\text{ELSE} \]
\[
\text{WRITE}(6,60) IPHASE
\]
\[
\text{ENDIF}
\]

\[
\text{ENDIF}
\]

\[
\text{IF}(IDBPRT.EQ.0) \text{GO TO 400}
\]

\[
\text{ENDIF}
\]
WRITE(6,100) XCG(I)
WRITE(6,105) (XAMB+CBARWG)*XLE
WRITE(6,110) DCMDCCL(I)
WRITE(6,120) FNEUTPT(I)
WRITE(6,130) SM(I)
WRITE(6,140) XCGAFT(I+1)
WRITE(6,410) SUMMO,SUMWT
WRITE(6,420) DCMDCCL,T,DCMDCCLW
WRITE(6,425) DCMDCCLF,DCMDCCLCN
WRITE(6,430) DCMDCCLJ
WRITE(6,440) LWB,LHT,LCN
WRITE(6,450) SHT1,SCN1

CONTINUE
FORMAT(5X,19HDATA AFTER TAKE-OFF)
FORMAT(5X,50HMISSION PHASE NUMBER,IS)
FORMAT(IOX,6HXCG = F15.7)
FORMAT(IOX,6HXAC = F15.7)
FORMAT(IOX,6HXDCMDCL = F15.7)
FORMAT(IOX,6HXNEUTPT = F15.7)
FORMAT(IOX,6HSTATIC MARGIN = F15.7)
FORMAT(IOX,8HSUMMO = ,F15.6,5X,8HSUMWT = ,F15.6)
FORMAT(IOX,8HSUMMFT = ,F15.8,5X,8HSUMWFT = ,F15.8)
FORMAT(IOX,8HCMDCL = ,F15.8,5X,8HCMDCLCN = ,F15.8)
FORMAT(IOX,8HDCMDCLJ = ,F15.8)
FORMAT(IOX,8HLWB = ,F15.8,5X,8HLHT = ,F15.8,5X,8HLCN = ,F15.8)
FORMAT(IOX,8HSTI = ,F15.8,5X,8HSCN = ,F15.8)
FORMAT(IOX,31HWARNIN!! C.G. IS AFT OF LIMITS,/)15X,
RETURN
END

SUBROUTINE STABCALC3(I,IPHASE)

THIS SUBROUTINE DETERMINES THE AERODYNAMIC CENTER AND C.G. DURING
TAKEOFF, IT DETERMINES THE STICK FREE EFFECTS, IT DETERMINES
THE UPWASH EFFECTS ON THE CANARD, AND THE LIFT-CURVE SLOPE
OF THE CANARD.

INCLUDE 'STBCM.INC'

DETERMINE THE AERODYNAMIC CENTER OF THE WING,
USING THE GEOMETRIC SHAPE AND THE 2-D AND 3-D
LIFT COEFFICIENTS. THIS USES THE INPUT OF THE TAKEOFF MACH
NUMBER.

IF(AMTO .LE. 0.0) THEN
    AMTO = VROT
ENDIF

NEED TO TRANSFER LIFT-CURVE SLOPE DATA FROM AERODYNAMICS MODULE
TO DETERMINE THE AERODYNAMIC CENTER AT TAKE OFF MACH NUMBER.

MACH = AMTO
ICALC = 2
IAO = 1
CALL STBDT(ICALC,NERR,IGEO,KGPRNT,IGPLT)
ZCLALFA = CLALFA
ZMACH = MACH

CALL XACCALC(ZMACH,ZCLALFA)

*** DETERMINE THE CENTER-OF-GRAVITY FOR THE AIRCRAFT ***

CALL XCGCALC(I)

** DETERMINE THE STICK FREE MULTIPLIER FOR THE HORIZONTAL TAIL.

THIS WAS DONE BY CURVE FITTING TO DETERMINE THE TAIL HINGE MOMENTS DUE TO ALPHA AND ELEVATOR DEFLECTION.


INPUTS INCLUDE THE ELEVATOR SPAN, BALLANCE SPAN, AS WELL AS THE AR AND T/C OF THE TAIL.

IF (SCN .GT. 0.0 .AND. SHT .LE. 0.0) THEN
   CZELV = CELV * CBARCN
   CZBALHT = CBALHT * CELV
   CHT = CBARCN
   ARTAIL = ARCN
   THT = T2CHT * CBARCN
ELSE
   CZELV = CELV * CBARHT
   CZBALHT = CBALHT * CELV
   CHT = CBARHT
   ARTAIL = ARHT
   THT = T2CHT * CBARHT
ENDIF

NEED TO DETERMINE THE BALLANCE RATIO OF THE ELEVATOR. IF IT IS GREATER THEN 0.38, THEN IT IS SET TO 0.38 TO FIT THE GRAPHS.

CF = CZELV - CZBALHT
IF (((CZBALHT/CF)**2 .GT. (THT/(CF*2.))**2) THEN
   BR = SQRT(((CZBALHT/CF)**2) - ((THT/(CF*2.))**2))
ELSE
   BR = 0.
ENDIF
IF (BR .GT. 0.38) BR=0.38

CEC = CZELV/CHT
A1A = 1./ARTAIL
TC = T2CHT

HERE WE ARE CURVE-FITTING THE TWO GRAPHS TO GET THE VALUES FOR K1 AND K2, BY WHICH WE THEN GET BG1 AND BG2.

THESE GRAPHS ARE FROM REFERENCE 3, FIGURES R.11 AND R.12

K11=1.027+3.5772*CEC-1.8662*CEC**2+0.174*CEC**3-3.587-CEC**4
K12=1.1692-2.5003*TC+23.4944*TC**2-202.1809*TC**3+546.3254*TC**4
\[ K_{13} = 1.428032 \times 10^{-8} - 3.15474 \times 10^{-2} + 7.28412 \times 10^{-3} - 5.8764 \times 10^{-4} \\
K_{14} = 1.22462 \times 10^{-1} - 1.2067 \times 10^{-2} + 1.378 \times 10^{-3} - 1.1478 \times 10^{-4} \\
B_{G1} = -5.5 \times 10^{-5} \times 10^{-1} - 10.12 \times 10^{-2} - 20.43 \times 10^{-3} - 30.76 \times 10^{-4} \\
K_{21} = 0.7504 + 0.8193 \times CE - 0.0829 \times CE^2 + 0.2082 \times CE^3 - 0.1449 \times CE^4 \\
K_{22} = 1.22462 \times 10^{-1} - 1.2067 \times 10^{-2} + 1.378 \times 10^{-3} - 1.1478 \times 10^{-4} \\
K_{23} = 8.9175 - 0.7178 \times CE^3 - 3.9307 \times CE^4 \\
K_{24} = 4.1065 \times CE - 8.9175 \times CE^2 + 9.7178 \times CE^3 - 3.9307 \times CE^4 \\
B_{G2} = -0.89 \times K_{21} \times K_{22} \times K_{23} \times K_{24} \\
\]

TAU IS DETERMINED FROM REFERENCE 3, CHAPTER 3, FIGURE 3.32. THIS IS KNOWN AS THE ELEVATOR EFFECTIVENESS FACTOR. 

\[ \text{TUU} = 4.1065 \times CE - 8.9175 \times CE^2 + 9.7178 \times CE^3 - 3.9307 \times CE^4 \]

HERE WE CALCULATE THE 'FREE ELEVATOR FACTOR' (DCMTMUUL), WHICH WILL BE MULTIPLIED TO THE ELEVATOR ANGLE OF ATTACK TO DETERMINE STICK FREE EFFECTS.

\[ \text{DCMTMUUL} = 1 - \left( \text{TUU} \times (B_{G1}/B_{G2}) \right) \]

**CANARD CONTRIBUTION TO AIRCRAFT STABILITY**

FIRST DETERMINE THE UPWASH ON THE CANARD DUE TO THE WING. THIS IS FOUND BY CURVE FITTING FIGURE 8.22c FROM REFERENCE 3.

\[ \text{CO} = \text{ROOTWG} \]

IF(XLECN .EQ. 0.0) THEN
    XFOR = (XLEWG + ROOTWG/4.) - XQBCN
ELSE
    XFOR = XLECN
ENDIF

\[ X_C = XFOR/CO \]

IF (X_C .GE. 5.) THEN
    ODEDA = 1.
    GO TO 190
ENDIF

IF (ARWG .LE. 8.) THEN
    X1 = 6.
    X2 = 8.
    Y1=2.8419 - 3.6494 \times X_C + 3.1799 \times X_C^2 - 1.3833 \times X_C^3 + 2.706 \times X_C^4 - 0.0219 \times X_C^5.
\]

\[ Y_2=2.8419 - 3.6494 \times X_C + 3.1799 \times X_C^2 - 1.3833 \times X_C^3 + 2.706 \times X_C^4 - 0.0219 \times X_C^5. \]

\[ \text{ODEDA} = \left( (X_2-ARWG) \times Y_1 - (X_1-ARWG) \times Y_2 \right) / (X_2-X_1) \]

GO TO 190

ENDIF

IF (ARWG .GT. 8. .AND. ARWG .LE. 10.) THEN
    X1 = 8.
    X2 = 10.
    Y1=2.8419 - 3.6494 \times X_C + 3.1799 \times X_C^2 - 1.3833 \times X_C^3 + 2.706 \times X_C^4 - 0.0219 \times X_C^5.
\]

\[ Y_2=2.9405 - 3.7399 \times X_C + 3.1866 \times X_C^2 - 1.3592 \times X_C^3 + 2.706 \times X_C^4 - 0.0219 \times X_C^5. \]

\[ \]
ODEDA=((X2-ARWG)*Y1-(X1-ARWG)*Y2)/(X2-X1)
GO TO 190
ENDIF
IF (ARWG .GT. 10.) THEN
X1 = 10.
X2 = 12.
Y2=3.1183-4.1271*XC-.5627*XC**2.-1.5362*XC**3. +.3217*XC**4.-.0259*XC**5.
ODEDA=((X2-ARWG)*Y1-(X1-ARWG)*Y2)/(X2-X1)
ENDIF
190 CONTINUE
CALCULATE LIFT CURVE SLOPE FOR THE CANARD. THIS ASSUMES
A THEORETICAL 2-D LIFT-CURVE SLOPE, AND A LINEARLY TAPERED
CANARD PLANFORM. THIS USED EQN 5.85 FROM REFERENCE 3 TO
DETERMINE THE LIFT-CURVE SLOPE, AND EQNS 5.89 a AND b TO DETERMINE THE
SWEEP AT THE ONE-HALF CHORD OF THE WING.
HERE IS CALCULATED THE SWEEP AT THE ONE-HALF CHORD POINT.
IF (SCN .NE. 0.0 . OR. SCN1 .NE. 0.0) THEN
IF (ARCN .EQ. 0.0) ARCN = (SPANCN/SPANCN) / SCN1
IF (TRCN .EQ. 0.0) TRCN = TRCN1
TANSWP2 = TAN(SWPCN/57.29578)-((1.-TRCN)/(1.+TRCN))*(1./ARCN)
A1 = 0.0
A2 = 1.55334
700 T1 = TAN(A1)
T2 = TAN(A2)
IF (ABS(T1-TANSWP2) .LE. 0.001) THEN
SWP2 = A1
GO TO 750
ENDIF
IF (ABS(T2-TANSWP2) .LE. 0.001) THEN
SWP2 = A2
GO TO 750
ENDIF
IF (ABS(T1-TANSWP2) .LE. ABS(T2-TANSWP2)) THEN
IF (ABS(T1-TANSWP2) .EQ. ABS(T2-TANSWP2)) THEN
SWP2 = 0.78539
GO TO 750
ENDIF
A2 = A2 - (A2 - A1)/2.0
GO TO 700
ELSE
A1 = A1 + (A2 - A1)/2.0
GO TO 700
ENDIF
750 CONTINUE
HERE IS CALCULATED THE LIFT-CURVE SLOPE
CL1 = 6.28319 * ARCN
CL2 = (ARCN * ZMACH)**2.0
CL3 = (ARCN / COS(SWP2))**2.0
IF(ZMACH .LE. 1.0) THEN
   CLACN = CL1 / (2.0 + SQRT(CL3 + 4.0 - CL2))
ELSE
   CLACN = CL1 / (2.0 + SQRT(CL3 + CL2 - 4.0))
ENDIF
ENDIF
CLACN = CLACN / 57.29578
RETURN
END

SUBROUTINE XACCALC(ZMACH,ZCLALFA)

C
C THIS SUBROUTINE IS USED TO DETERMINE THE AERODYNAMIC CENTER OF
C THE WING. THIS INCLUDES MACH NUMBER EFFECTS, WING SWEEP ANGLE, AND
C WING TAPER RATIO. THE EQUATIONS USED FOR THIS ARE TAKEN FROM
C REFERENCE 3, CHAPTER 8, PAGES 484 TO 485. THIS CALCULATES THE AC AS A
C FUNCTION OF THE RATIO OF THE TWO-TO-THREE DIMENSIONAL LIFT-CURVE SLOPES.
C THIS RATIO IS FOUND FROM THE CLASSIC EQUATION FOR TWO-TO-THREE
C DIMENSIONAL LIFT-CURVE SLOPES, REFERENCE 4, EQUATION 5-20.
C THIS WORKS BEST UP TO MACH OF 1, BUT STARTS TO DEVIATE BEYOND THAT POINT.
C
INCLUDE 'STBCM.INC'
C
THE VARIABLE 'A' IS THE POSITION OF THE QUARTER-CHORD POINT OF
C THE WING LOCATED AT THE CENTER OF THE FUSELAGE.
C
BODRAD = BOMAX / 2.
XLECL = XLEWG - (BODRAD * TAN(SWPWOI57.2957B))
XTECL = XLEWG + RDOTWw
A = XLECL + ((XTECL - XLECL)/4.)
C
RAD = 57.29578
PI = 3.1415926
C
CLACLA = 1.0 / (1.0 - ((RAD + ZCLALFA) / (PI + ARWO)))
XA = ((1.0 + 2.0*TRWw)/(1.0 + TRWw)) * ((SPANWw / 2.0) +
$ TAN(SWPWw/RAD)) * CLACLA + 0.3333333
XACWG = XA * A
XLE = XQCBWG - CBARWG/4.
C
RETURN
END

SUBROUTINE XCGCALC(I)

C
C INCLUDE 'STBCM.INC'
C
THIS SUBROUTINE IS CALLED TO CALCULATE THE RELATIVE POSITIONS
AND MOMENTS OF EACH OF THE AIRCRAFT WEIGHTS WHICH ARE
USED TO DETERMINE OVERALL AIRCRAFT C.G.  NOTE THAT PLACEMENT
OF MOST WEIGHTS IS ARBITRARY, BUT CAN BE ADJUSTED BY THE
USERS WITH THE X*** INPUTS IN /STABIN/.

1000 CONTINUE

XBOD = XFBOD * BODL
MBOD = WBODY * XBOD
XCAN = (XQCBCN + XFCAN * CBARCN)
MCAN = WCAND * XCAN
XHT = (XQCBHT + XFHT * CBARHT)
MHT = WHT * XHT
XVT = (XQCBVT + XFVT * CBARVT)
MVT = WVT * XVT
XWING = (XQCBWG + XFWG * CBARWG)
MWG = WWING * XWING
MAIRC = WAIRC * XLEWG
XAPU = XFAPU * BODL
MAPU = WAPU * XAPU
XELT = XFELT * BODL
MELT = WELT * XELT
XEP = XFEP * BODL
MEP = WEP * XEP
XINST = XFINST * BODL
MINST = WINST * XINST
XFS = XFFS * BODL
MFS = WFS * XFS
XHDP = XFHDP * BODL
MHDP = WHDP * XHDP
XSC = XFSC * BODL
MSC = WSC * XSC
XPA = XFPA * BODL
MPA = WPA * XPA
XCARGO = XFCARGO * BODL
MCARGO = XCARGO * WCARGO
XCREW = XFCREW * BODL
MCREW = WCREW * XCREW
XBOMB = XFBOMB * BODL
MBOMB = XBOMB * WZB
XBB2 = XFB2 * BODL
MBB2 = XBB2 * WBB2
XMISS = XFMISS * BODL
MMISS = XMIS * WZM
XAMMUN = XFAMMUN * BODL
MAMMUN = XAMMUN * WZM
XNA = XFINA * BODL
MNA = XNA * WNA

THE WEIGHTS MODULE DOES NOT GIVE OUT THE PIVOT WEIGHTS
DIRECTLY, SO IT IS CALculated BY SUBTRACTING THE DIFFERENT
PARTS FROM THE AIRFRAME WEIGHT.

WPIV = WAF-WBODY-WHT-WL-WNA-WVT-WWING-WCAND
IF (WPIV .LT. 0.) WPIV = 0.
   XPIV = XFPIV * BODL
   MPIV = WPIV * XPIV

   THE SAME GOES FOR THE LIFT FAN AS GOES FOR THE PIVOTS ABOVE.

   WLFTF = WPS - WE - WFS
   IF (WLFTF .LT. 0.) WLFTF = 0.
   XLIFTF = XFLIFTF * BODL
   MLIFTF = WLFTF * XLIFTF

   WLFRT = WFLGFRT * WLC
   XLFRT = XFLGFRT * BODL
   MLFRT = WLFRT * XLFRT
   WLGR = WLG - WLGFRT
   IF (LGFLAG .GT. 0) GO TO 1100
   XLGR = XFLGR * BODL

1100 CONTINUE
   MLGR = XLGR * WLGR

   WZENG = WE / EN
   DO 220 J = 1, EN
      XENG(J) = (XLEPOD(J) * XFENG * POOL)
      MENG(J) = WZENG * XENG(J)
   220 CONTINUE

   FURNISHING WEIGHT IS FOUND BY SUBTRACTING THE KNOWN WEIGHTS FROM
   THE WEIGHTS OF THE FIXED EQUIPMENT.

   WFUR = WFEQ-WAIRC-WELT-WAPU-WPA-WHDP-WINST-WSC
   IF (WFUR .LT. 0.) WFUR = 0.
   XFUR = XFFUR * BODL
   MFUR = XFUR * WFUR

   PAYLOAD IS FOUND LIKE FURNISHINGS

   WPAYL = WPL-WCREW-WAMMUN-WBB2-WBOMB-WMISS-WETANK-WCARGO
   IF (WPAYL .LT. 0.) WPAYL = 0.
   XPAYL = XFPAYL * BODL
   MPAYL = XPAYL * WPAYL

   CALCULATE MOMENTS FOR FUEL
   IF (FUFRAC .GT. 1.0) FUFRAC=1.0
   WFWG = FUFRAC * WFP
   XWGFUEL = (XFFWG * CDARW) + XQCRWG
   WFWG = WFWG * XWGFUEL
   WFFUS = WFP - WFWC
   IF (WFFUS .LT. 0.0) WFFUS=0.0
   XFUSF = (BODL * XFFUS)
   WFFUS = WFFUS + XFUSF

   NOW INCLUDE ANY EXTERNAL TANKS DEPENDING ON THE INPUT IETANK.
XCTIPWG = ROOTWG + TRWG
XLETIP = XLEWG + .25*ROOTWG + SPANWG/2. + TAN(SWPWC/57.29578)
1 - .25*XCTIPWG
GO TO (510, 520, 530, 540, 550, 560), IETANK
C
CONDITION IETANK = 0, SINGLE TANK IN CENTERLINE OF FUS.
XEF1 = (XFEX + BODL)
MEF = XEF1 * (WZET + WZEF)
GO TO 600
C
CONDITION IETANK = 1, TWO TANKS ON THE SIDES OF FUS.
510 XEF1 = (XFEX + BODL)
MEF = XEF1 * (WZET + WZEF)
GO TO 600
C
CONDITION IETANK = 2, TWO TANKS AT C/4 OF WING
520 XEF1 = XQCBWG + CBARWG/4.
MEF = XEF1 * (WZET + WZEF)
GO TO 600
C
CONDITION IETANK = 3, TWO TANKS AT WINGTIPS.
530 XEF1 = (XLETIP + 0.5*XCTIPWG)
MEF = XEF1 * (WZET + WZEF)
GO TO 600
C
CONDITION IETANK = 4, SINGLE FUS. TANK, TWO WING TANKS.
540 WEFF = .5*WZEF + .5*WZET
WEFW = WZEF + WZET - WEFF
XEF1 = (XFEX + BODL)
XEF2 = (XQCBWG + CBARWG/4.)
MEF = XEF1 * WEFF * XEF2 * WEFW
GO TO 600
C
CONDITION IETANK = 5, SINGLE FUS. TANK, TWO TIP TANKS.
550 WEFF = .75*WZEF + .75*WZET
WEFWT = WZET - WZEF - WEFF
XEF1 = (XFEX + BODL)
XEF2 = (XLETIP + .5*XCTIPWG)
MEF = XEF1 * WEFF * XEF2 * WEFWT
GO TO 600
C
CONDITION IETANK = 6, TWO WING TANKS, TWO TIP TANKS.
560 WEFWT = WZEF - WZET - WEFF
XEF1 = (XQCBWG + CBARWG/4.)
XEF2 = (XLETIP + .5*XCTIPWG)
MEF = XEF1 * WEFWT * XEF2 * WEFWT
600 CONTINUE
C
NOW FIND MOMENTS AND WEIGHTS OF FUEL.
WFL = WFFUS + WFWG + WZEF - WZET
MFL = MFWG + MFFUS - MEF
C
DETERMINE THE SUM TOTAL OF THE WEIGHTS AND THE
C.G. OF THE AIRCRAFT.
C
SUMMO = MBOD + MCAN + MHT + MNA + MVT + MWG + MAIRC + MAPL + MELT + MSC + MFIV
1 + MEP + MINS + MHD + MPA + MFUR + MPAYL + MCREW + MACAR + MMAL + MMISS + MCARCO
2 + MBOM + MFL + MLQFRT + MLGR + MLIFTF + MB22 + MREM
C
DO 230 J = 1, EN
SUMMO = SUMMO + MENG(J)

230 CONTINUE

SUMWT = WBODY + WCAND + WHT + WNA + WVT + WWINC + WAPU + WELT + WSC + WPIV
1 + WEP + WINST + WHDP + WPA + WFM + WPL + WIA + WZM + WLC + WLR + WCA
2 + WCREW + WZB + WREM + WFL + WLT + WEN + WBB2

XCG(I) = SUMMO / SUMWT

THIS CALCULATES THE CENTER OF GRAVITY POSITION IN PERCENT MAC.

C
CGBAR(I) = (XCG(I) - XLE) / CBARWG * 100.

IF (ICGPRT.EQ.0) GO TO 800

THIS PRINT FLAG PRINTS THE POSITIONS AND WEIGHTS OF
THE DIFFERENT COMPONENTS USED TO DETERMINE CORRECT PLACEMENT
OF THE AIRCRAFTS C.G.

CALL CQPRINT

800 CONTINUE

THIS WRITE STATEMENT PRINTS A WARNING IF THE C.G. IS CLOSER TO THE
MAIN GEAR THEN THE 15 DEGREE LIMIT. THIS LIMIT IS TAKEN FROM NAVY
SPECIFICATIONS SD-24J, WHICH GIVES A GOOD DESIGN VALUE. THIS
SPECIFICATION WAS TAKEN FROM REFERENCE 5, PAGE 48, FIGURE 4.3.

ZMG = ZFMG * BDMAX
TAN15 = 0.26795
TANB = (XLGR - XCG(I)) / ZMG

IF (TANB .LT. TAN15) THEN
  LGFLAG = 1
  XLGR = XLGR * (0.01 * BODL)
  WRITE (6,700) XCG(I), XLGR
ENDIF

700 FORMAT(//,5X,47HWARNING!!! C.G. IS LESS THEN 15 DEGREES FORWARD,
1/15X,38HOF THE MAIN GEAR IN THE VERTICAL PLANE.5X/,9HMAIN GEAR,
220H MOVED AFT 0.01*BODL,/,5X,6HXCG = ,F9.5,5X,6HXMG = ,F9.5,/)
INCLUDE 'STBCM.INC'

IF (ARHT .LE. 13.) THEN
  A = ARHT
  QCLAFHT = -0.282 + 0.0359*A - 0.0049*A**2 + 0.000345*A**3 - 0.000009561*A**4
ELSE
  QCLAFHT = (2.*3.1415926)/57.29578
ENDIF

FIRST DETERMINE THE DYNAMIC PRESSURE, AND THE LIFT OF THE WING.

VTOR = VROT * 1077.4
QTOR = 0.5 * 0.0017556 * (VTOR)**2

NOW CALCULATE DISTANCE PERAMETERS SPECIFYING DISTANCES FROM
THE WING LEADING EDGE TO THE PARTICULAR COMPONENT.

ZMG = ZFMG * BDMAX
ZT1 = ZFT1 * BDMAX
ZT2 = ZFT2 * BDMAX
ZD = ZFD * BDMAX
T1RAD = GAMAT1 / 57.29578
T2RAD = GAMAT2 / 57.29578
XLEMC = XLCG - XQCBWG
XLECG = XCG(I) - XQCBWG
XLET1 = XQCBWG - (XT1*BODL)
XLET2 = (XT2*BODL) - XQCBWG
XLEACWG = XACWG - XQCBWG
IF(XLEC N .EQ. 0.0) THEN
  XLEC N = XQCBWG - XQCB CN
ENDIF
XLEHT = XQCBHT - XQCBWG

CALCULATE THE LIFT OF THE WING. THIS INCLUDES GROUND EFFECTS
FOR THE LIFT COEFFICIENT. THIS IS JUST MULTIPLICATION OF THE
WING LIFT-CURVE SLOPE MODIFIED FOR GROUND EFFECT,
BY THE ANGLE OF ATTACK OF THE WING, AND THE DYNAMIC PRESSURE.
FIRST MODIFY FOR GROUND EFFECTS

DG = (ZRTWG+ZMG) / (SPANWG/2.)
CALL GRNDEFFECT(DG, AFGDAF)
CLAORD = AFGDAF * CLALFA
LWB = CLAORD * (ALFTO) * QTOR * SWC

CALCULATE THE CHANGE IN LIFT-CURVE SLOPE OF THE TAIL WHILE
IN GROUND EFFECT.

IF(SPADHT .LE. 0.0) GO TO 13
DG = (ZRTHT+ZMG) / (SPANHT/2.)
CALL GRNDEFFECT(DG, AFGDAF)
CLAHTORD = AFGDAF * QCALFHT

CONTINUE

13
CALCULATE THE CHANGE IN LIFT-CURVE SLOPE OF THE CANARD WHILE IN GROUND EFFECT.

IF (SPANCN .LE. 0.0) GO TO 14
DO = (ZRTCN + ZMG) / (SPANCN/2.)
CALL GNDEFFECT(DG, AFGDAF)
CLACNGRD = AFGDAF • CLACN
14 CONTINUE

IF (SPANCN1 .LE. 0.0) GO TO 41
DO = (ZRTCNI + ZMG) / (SPANCN1/2.)
CALL GNDEFFECT(DG, AFGDAF)
CLACNGRD = AFGDAF • CLACN
41 CONTINUE

NOW CALCULATE THE DOWNWASH ANGLE OF THE WING ON THE TAIL.

IF (SHT .EQ. 0.0) GO TO 10
CALL DOWNWASH(CLTO, ALFTO, DWANOLE)
10 CONTINUE

THE CALL TO TRAJDT IS USED TO CALCULATE THE MAXIMUM THRUST AT TAKEOFF ROTATION OF THE AIRCRAFT.

MODN04=4
CALL TRAJDT(MODN04, I CALC, NERROR, I G E O, K G P R N T, IGPLT)
IF (NERROR .GE. 2) RETURN

TN = THRUST•EN

TWZTO = TN

NOW CALCULATE THE THRUST SPLIT, IF ANY.
SHTTEMP = SHT1
SCNTEMP = SCN1
SHT1 = SHT
SCN1 = SCN
IF (IVECT .NE. 0 .AND. TSPLIT .EQ. 0.0) THEN
CALL VECTHRUST(I)
ENDIF
SHT1 = SHTTEMP
SCN1 = SCNTEMP
T1 = TSPLIT • TWZTO
T2 = TWZTO - T1

NOW CALCULATE THE LIFT OF THE HORIZONTAL TAIL AND CANARD, DEPENDING ON WHICH OF THE TWO EXIST. IF BOTH EXIST, THE CANARD IS ASSUMED TO BE USED ONLY FOR TRIM, THE HORIZONTAL TAIL IS THE DESIRED SIZE.

THE METHOD USED HERE IS TAKEN FROM REFERENCE 2, CHAPTER 5 PAGE 373-374. THIS IS MODIFIED TO INCLUDE A CANARD AND ANY THRUST OFFSET DUE TO PLACEMENT OF THE ENGINES.

IF (SCN1 .EQ. 0.0 .AND. SCN .EQ. 0.0) THEN
** THIS IS THE CASE WHERE THERE IS A HORIZONTAL TAIL ONLY **

\[
\begin{align*}
LH1 &= CMWBT0 \cdot QTOR \cdot SWC \cdot CBARWG \\
LH3 &= ZT1 \cdot T1 \\
LH5 &= ZT2 \cdot T2 \\
LH6 &= WGT0 \cdot (XLEMG - XLECG + ZMU \cdot ZMG) \\
LH7 &= LWB \cdot (XLEMCWG - XLEACWG + ZMU \cdot ZMG) \\
LH8 &= D \cdot ZD \\
\text{DENOM} &= XLEHT - XLEMG - ZMU \cdot ZMG \\
LHT &= (LH1 + LH2 + LH3 - LH4 + LHS - LH6 - LH7 - LH8) / \text{DENOM}
\end{align*}
\]

SHT1 = LHT/(CLAHTGRD*QTOR*(ALFT0-DWANGLE+IHT+TAU+ELVDEF))

ELSE
IF (SHT .EQ. 0.) THEN

** THIS IS THE CASE WHERE THERE IS A CANARD BUT NO H. TAIL **

\[
\begin{align*}
LC1 &= CMWBT0 \cdot QTOR \cdot SWC \cdot CBARWG \\
LC3 &= ZT1 \cdot T1 \\
LC5 &= ZT2 \cdot T2 \\
LC6 &= WGT0 \cdot (XLEMG - XLECG + ZMU \cdot ZMG) \\
LC7 &= LWB \cdot (XLEACWG - XLEMG - ZMU \cdot ZMG) \\
LC8 &= D \cdot ZD \\
\text{DENOM} &= XLECN + XLEMG \cdot ZMU, ZMG \\
LCN &= (-LC1 - LC2 - LC3 - LC4 - LC5 - LC6 - LC7 + LC8) / \text{DENOM}
\end{align*}
\]

SCN1 = LCN/(CLACNGRD*QTOR*(ALFT0+ICN+TAU+ELVDEF))

ELSE

** THIS IS THE CASE WITH BOTH CANARD AND HORIZONTAL TAIL **

\[
\begin{align*}
LH1 &= CMWBT0 \cdot QTOR \cdot SWC \cdot CBARWG \\
LH3 &= ZT1 \cdot T1 \\
LH5 &= ZT2 \cdot T2 \\
LH6 &= WGT0 \cdot (XLEMG - XLECG + ZMU \cdot ZMG) \\
LH7 &= LWB \cdot (XLEMCWG - XLEACWG + ZMU \cdot ZMG) \\
LH8 &= D \cdot ZD \\
\text{DENOM} &= XLEHT - XLEMO - ZMU, ZMG \\
LHT &= (LH1 + LH2 + LH3 - LH4 + LHS - LH6 - LH7 - LH8 + LH9) / \text{DENOM}
\end{align*}
\]

SHT1 = LHT/(CLAHTGRD*QTOR*(ALFT0-DWANGLE+IHT+TAU+ELVDEF))

ENDIF
ENDIF
RETURN
END

SUBROUTINE XCGFORWARD(I)
THIS SUBROUTINE CALCULATES THE LONGITUDINAL CONTROL SURFACE
SIZE NEEDED TO MAINTAIN CONTROLLABILITY OF THE AIRCRAFT IN
LANDING CONFIGURATION. THAT IS MAX CL, LOW SPEED, HIGH DRAG,
IN GROUND EFFECT. THIS IS DONE BY A SUMMATION OF MOMENTS ABOUT
THE CENTER OF GRAVITY. WITH THIS EQUATION SET TO ZERO, THE
CENTER OF GRAVITY POSITION IS SOLVED FOR.

INCLUDE 'STBCM.INC'

VTHT = (CLAHTGRD * SHT1 * ETAHT) / SWG
VTCN = (CLACNGRD * SCN1 * ETACN) / SWG

IF (SHT .EQ. 0.0) GO TO 20
CALL DOWNWASH(CLMAX,ALFMAX,DWANOLE)
20 CONTINUE

DETERMINE THE DIFFERENT ANGLES OF ATTACK.

ALFHT = ALFMAX + IHT - DWANOLE
ALFCN = ALFMAX + ICN
IF(SCN .EQ. 0.0) THEN
   ALFHT = ALFHT + TAU * ELVDMAX
ELSE
   IF(SHT .EQ. 0.0) THEN
      ALFCN = ALFCN + TAU * ELVDMAX
      ALFHT = 0.0
   ELSE
      ALFHT = ALFHT + TAU * ELVDMAX
   ENDIF
ENDIF

XAWBZ = XLE + XAWB * CBARWG
CM1 = CLMAX * XAWBZ
CM2 = CMWBMAX * CBARWG
CM3 = VTHT * ALFHT * XQCBBHT
CM4 = VTCN * ALFCN * XQCBCN
CM5B = ZT1 * T1
CM5D = ZT2 * T2
CM5 = (CM5B + CM5D) * CBARWG
DENOCM = CLMAX + (VTHT * ALFHT) + (VTCN * ALFCN)

THIS SOLVES FOR THE FORWARD C.G. LIMIT.

XCGFRWD(I) = (CM1 - CM2 - CM3 - CM4 - CM5) / DENOCM

IF(IDBPRT .NE. 0) THEN
   WRITE(6,10)I-2,XCGFRWD(I)
ENDIF

10 FORMAT(IOX,8HXCGFRWD[,I2,4H] = ,F12.6)
RETURN
SUBROUTINE VECTHRUST(I)

C THIS SUBROUTINE SOLVES FOR ONE OF THREE SPECIFIED VARIABLES IN THE
C SUMMATION OF FORCES ABOUT THE C.G., THE THRUST ANGLE OF THE REAR
C NOZZLE, THE POSITION OF THE FOREWARD THRUST NOZZLE, AND THE THRUST
C SPLIT OF THE FORE AND AFT NOZZLES. THE VALUES CALCULATED ARE THOSE
C NEEDED TO EQUATE THE MOMENTS ABOUT THE C.G. TO ZERO.

C INCLUDE 'STBCM.INC'

WEMP=WGTO-WFTOT-WMISS-WAMMUN-WBOMB-WBB2-WBB1-WETANK
V1 = CLMAX * (((XCG(I) - XACWG) / CBARWG)
V2 = CMWBMAX
V3A = CLAFHT * ALFHT * ETAHT * SHT1 / SWG
V3B = (XQCBHT - XCG(I)) / CBARWG
V3 = V3A * V3B
V4A = CLACN * ALFCN * ETACN * SCN1 / SWG
V4B = (XCC(I) - XQBCCN) / CBARWG
V4 = V4A * V4B
SUMVT = V1 * V2 - V3 ÷ V4
GO TO (100,110,120), IVECT

100 CONTINUE
IF(T1*SIN(TIRAD) + T2*SIN(T2RAD) .LT. WEMP) THEN
   WRITE(6,500) T1*SIN(TIRAD) * T2*SIN(T2RAD), WEMP
GO TO 170
ENDIF
V5 = ZT1 * T1 * COS(TIRAD)
V6 = ((XT2 • BODL) - XCG(I)) * T2 • SIN(T2RAD)
V7 = ZT2 * T2 * COS(T2RAD)
SUMVT = SUMVT + V5 - V6 + V7
XTI = (SUMVT / (T1 * SIN(T1RAD)) * XCG(I)) / BODL
GO TO 170

110 CONTINUE
NUMVT = (SUMVT/TWZTO) - ((XT2*BODL) - XCG(I))*SIN(T2RAD)
$ = ZT2*COS(T2RAD)
DENOV = ZT2*COS(T2RAD) - ((XT2*BODL) - XCG(I))*SIN(T2RAD)
$ - (XCG(I) - (XT1*BODL))*SIN(T1RAD) - ZT1*COS(T1RAD)
TSPLIT = NUMVT / DENOV
GO TO 170

120 CONTINUE
DUMM = SUMVT * (XCG(I) - (XT1*BODL)) * T1 * SIN(T1RAD) - ZT1 * T1 * COS(T1RAD)
GAM21 = 0.0
GAM22 = 1.5707963

150 CONTINUE
DUM1 = DUMM+ZT2*T2*COS(GAM21) - (XT2*BODL-XCG(I))*T2*SIN(GAM21)
DUM2 = DUMM+ZT2*T2*COS(GAM22) - (XT2*BODL-XCG(I))*T2*SIN(GAM22)
IF(ABS(ABS(DUM1)-ABS(DUM2)) .LT. 0.05) THEN
   T2RAD = GAM21
GO TO 170
ENDIF
IF(ABS(DUM1) .LT. ABS(DUM2)) THEN
   GAM22 = GAM22 - (GAM22 - GAM21) / 2.0
ELSE
    GAM21 = GAM21 + (GAM22 - GAM21) / 2.0
ENDIF
GO TO 150
170 CONTINUE
C
500 FORMAT(10X,61HWARN!! VERTICAL THRUST COMPONENT LESS THEN $ AIRCRAFT WEIGHT!,/10X,7HTHRUST=,F15.7,5X,7HWEIGHT=,F15.7)
RETURN
END
C

SUBROUTINE DATATRANS
C
INCLUDE 'STBCM.INC'
C
THIS SUBROUTINE IS USED TO CALL THE DATA TRANSFER SUBROUTINE. IT
IS USED TO DETERMINE THE ANGLE OF ATTACK, COEFFICIENT OF LIFT, AND
PITCHING MOMENT COEFFICIENT FOR TAKEOFF, MAX LIFT CONDITIONS,
AND ZERO ANGLE OF ATTACK CONDITIONS.
C
MACH = 0.35
ALT = 5000.0
CL = 0.54
CD = .54
CMWB = 0.54
ALPHA = AWTOT
ICALC = 2
IAO = 1
CALL STBDT(ICALC,NERR,IGEO,KOPRNT,IOPLT)
ALFTO = ALPHA
CLTO = CL
CDTO = CD
CMWBT0 = CMWB
C
****
CL = 0.54
CD = .54
CMWB = 0.54
ALPHA = 0.54
IAO = 10
CALL STBDT(ICALC,NERR,IGEO,KOPRNT,IOPLT)
ALFMAX = ALPHA
CLMAX = CL
CDMAX = CD
CMWBMAX = CMWB
C
****
CL = 0.0001
CD = .54
CMWB = 0.54
ALPHA = 0.54
IAO = 8
CALL STBDT(ICALC,NERR,IGEO,KOPRNT,IOPLT)
ALFO = ALPHA
CLO = CL
CDO = CD
CMWBO = CMWB

RETURN
END

SUBROUTINE DOWNWASH(CLZ, ALFZ, DWANOLE)

THIS SUBROUTINE CALCULATES THE DOWNWASH ANGLE ON THE TAIL. FIRST
BY THE THEORETICAL METHOD, AND THEN BY POSITION OF THE TAIL RELATIVE
TO THE WING VOTECIES.

INCLUDE 'STBCM.INC'

CALCULATE THE DOWNWASH ANGLE OF THE WING ON THE TAIL.

DWANOLE = ((1.62 * CLZ) / (3.1415926 * ARWG)) * 57.29578

NOW MODIFY THIS CALCULATION FOR THE POSITION OF THE TAIL. THE
METHOD USED IS FROM REFERENCE 1, SECTION 4.4.1.
FIRST IS A CURVE FIT OF FIGURE 4.4.1-55.

ALFRATIO = (ALFZ - ALFO) / (ALFMAX - ALFO)

IF(SWPWG .LE. 0.0) THEN
AREFF = ARWG
SPANEFF = SPANWG
GOTO 1100
ENDIF

IF(ALFRATIO .LT. 0.4) THEN
AREFF = ARWG
SPANEFF = SPANWG
GOTO 1100
ENDIF

IF(ALFRATIO .LT. 0.56 .AND. SWPWG .LT. 45.0) THEN
AREFF = ARWG
SPANEFF = SPANWG
GOTO 1100
ENDIF

IF(ALFRATIO .LT. 0.56) THEN
A1 = -2.5098 + 6.2131 * ALFRATIO
GOTO 1000
ENDIF

IF(ALFRATIO .LT. 0.73 .AND. SWPWG .LT. 30.0) THEN
AREFF = ARWG
SPANEFF = SPANWG
GOTO 1100
ENDIF

IF(ALFRATIO .LT. 0.73) THEN
X1 = 45.0
X2 = 60.0
Y1 = -2.8585 + 5.0852 * ALFRATIO
\[ Y_2 = -2.5098 + 6.2131 \times \text{ALFRATIO} \]

\[ A_1 = \frac{((X_2 - \text{SWPWO}) \times Y_1 - (X_1 - \text{SWPWO}) \times Y_2)}{(X_2 - X_1)} \]

\[ \text{IF}(A_1 < 0.0) \quad A_1 = 0.0 \]

ENDIF

1000 CONTINUE

\[ \text{IF}(\text{TRWG} < 0.50) \quad \text{THEN} \]

\[ X_1 = 0.0 \]

\[ X_2 = 1.0 \]

\[ Y_1 = 0.6 - 0.150 \times A_1 \]

\[ Y_2 = 0.6 - 0.1245 \times A_1 \]

\[ \text{AREFF} = \text{AEFA} \times \text{ARWG} \]

ENDOR

\[ \text{ELSE} \]

\[ X_1 = 0.5 \]

\[ X_2 = 1.0 \]

\[ Y_1 = 0.6 - 0.1245 \times A_1 \]

\[ Y_2 = 0.6 - 1.000 \times A_1 \]

\[ \text{AREFF} = \text{AEFA} \times \text{ARWG} \]

ENDIF

\[ X_1 = 0.0 \]

\[ X_2 = 1.0 \]

\[ Y_1 = -0.20326 \times 2.6606 \times \text{AEFA} - 2.631 \times (\text{AEFA}^{2}) \times 0.80556 \times (\text{AEFA}^{3}) \]

\[ Y_2 = -0.16464 \times 1.0168 \times \text{AEFA} \]

\[ \text{SPANEFB} = \frac{((X_2 - \text{TRWG}) \times Y_1 - (X_1 - \text{TRWG}) \times Y_2)}{(X_2 - X_1)} \]

\[ \text{SPANEFF} = \text{SPANEFB} \times \text{SPANWG} \]

1100 CONTINUE

\[ \text{NOW SOLVE FOR THE VERTICAL POSITION OF THE VORTEX CORE.} \]

\[ \text{THIS ASSUMES TRAILING EDGE SEPERATION. THESE USE EQUATIONS} \]

\[ 4.4.1-A \text{ THROUGH} 4.4.1-F \text{ IN REFERENCE 1, SECTION 4.4.1.} \]

\[ \text{SIG1} = \frac{(0.56 \times \text{ARWG})}{\text{CLZ}} \]

\[ \text{BVRU} = (0.78 \times 0.10 \times (\text{TRWG} - 0.4) \times 0.003 \times \text{SWPWO}) \times \text{SPANEFF} \]

\[ \text{LEFF} = \text{XQCBHT} - \text{XLETIP} \]

\[ \text{BV} = \text{SPANEFF} - \text{SPANEFF} \times \text{SORT}(2.0 \times \text{LEFF}) / (\text{SPANWG} \times \text{SIG1}) \]
\[ HH = ZRTHT - ZRTWC \]
\[ HVERT = HH - LEFF \times ((\text{ALFZI57.29578}) - (0.41 \times \text{CLZ}) / (3.14159 \times \text{AREFF})) \]
\[ S = (\text{SPANEFF} / 2.0) \times \tan(DJHED / 57.29578) \]

NOW CALCULATE THE MODIFIED DOWNWASH ANGLE.

\[ DWANCLE = DWANOLE \times \left(1 + \left(\frac{2.0 \times HVERT}{BV}\right)^2\right) \]

RETURN

END

SUBROUTINE CRDNEFFECT(DG, AFGDAF)

THIS SUBROUTINE DETERMINES THE CHANCES TO CLALPHA FOR TAKEOFF AND LANDING CALCULATIONS. THIS USES FIGURE 5-39 FROM REFERENCE 4, P. 257. THE HEIGHT FROM THE GROUND USED IS THE HEIGHT OF AIRCRAFT WITH THE MAIN LANDING GEAR ON THE GROUND.

INCLUDE 'STBCM.INC'

IF (ARWGC .LE. 6.) THEN
    X1 = 4.
    X2 = 6.
    Y1 = 1.2989 - 0.12851 \times DG + 0.79231 \times DG**2 - 0.4131 \times DG**3 - 0.083452 \times DG**4
    Y2 = 1.2054 - 0.49627 \times DG + 0.54742 \times DG**2 - 0.29063 \times DG**3 - 0.059777 \times DG**4
    AFGDAF = ((X2 - ARWGC) \times Y1 - (X1 - ARWGC) \times Y2) / (X2 - X1)
    GO TO 190
ENDIF

IF (ARWGC .GT. 6. AND. ARWGC .LE. 8.) THEN
    X1 = 6.
    X2 = 8.
    Y1 = 1.2054 - 0.49627 \times DG + 0.54742 \times DG**2 - 0.29063 \times DG**3 - 0.059777 \times DG**4
    Y2 = 1.1751 - 0.5658 \times DG + 0.31668 \times DG**2 + 0.068063 \times DG**3
    AFGDAF = ((X2 - ARWGC) \times Y1 - (X1 - ARWGC) \times Y2) / (X2 - X1)
    GO TO 190
ENDIF

IF (ARWGC .GT. 8. AND. ARWGC .LE. 10.) THEN
    X1 = 8.
    X2 = 10.
    Y1 = 1.1751 - 0.46048 \times DG + 0.5555 \times DG**2 - 0.31668 \times DG**3 - 0.086063 \times DG**4
    Y2 = 1.1498 - 0.51326 \times DG + 0.3014 \times DG**2 - 0.068288 \times DG**3
    AFGDAF = ((X2 - ARWGC) \times Y1 - (X1 - ARWGC) \times Y2) / (X2 - X1)
    GO TO 190
ENDIF

IF (ARWGC .GT. 10.) THEN
    X1 = 10.
    X2 = 12.
    Y1 = 1.1498 - 0.31846 \times DG + 0.4825 \times DG**2 - 0.21725 \times DG**3 - 0.046757 \times DG**4
    Y2 = 1.1188 - 0.51326 \times DG + 0.3014 \times DG**2 - 0.068288 \times DG**3
    AFGDAF = ((X2 - ARWGC) \times Y1 - (X1 - ARWGC) \times Y2) / (X2 - X1)
ENDIF
SUBROUTINE STABOUT

THIS SUBROUTINE PRINTS OUT THE PERTINATE DATA CALCULATED BY THE MODULE.

INCLUDE 'STBCM.INC'

DO 53 I = 3, NPHASE*3
   IF(XCOFRWD(I) .GT. XCOFRWD(I)) THEN
      XCOFRWD(I) = XCOFRWD(I)
   ENDIF
CONTINUE

DO 54 I = 3, NPHASE*3
   IF(XCGAFT(I) .LT. XCGAFT(2)) THEN
      XCGAFT(2) = XCGAFT(I)
   ENDIF
CONTINUE

DO 55 I = 2, NPHASE*2
   SM(I) = ((XCGAFT(2) - XCO(I))/CBARWO)*100.
CONTINUE

WRITE (6,50)
WRITE(6,105) (XAWB-CBARWO)*XLE
IF(SHT1 .NE. 0.) THEN
   WRITE(6,150) SHT1
ENDIF
IF(SCNI .NE. 0.) THEN
   WRITE(6,160) SCNI
ENDIF
WRITE (6,135) XCOFRWD(I)
WRITE (6,140) XCGAFT(2)
DO 10 I = 2, NPHASE*2
   IF (I .LE. 2) THEN
      WRITE(6,59)
   ELSE
      WRITE(6,60) I - 2
   ENDIF
   WRITE(6,120) FNEUTPT(I)
   WRITE(6,100) XCG(I)
   WRITE(6,107) CQBAR(I)
   WRITE(6,110) DCMDCL(I)
   WRITE(6,130) SM(I)
CONTINUE

IF(IVECT .EQ. 0) THEN
   RETURN
END
GO TO 500
ELSE
  WRITE(6,170)T1*SIN(T1RAD),T2*SIN(T2RAD)
  WRITE(6,180)XT1,XT2
  WRITE(6,185)ZT1,ZT2
  WRITE(6,190)T1RAD*57.29578,T2RAD*57.29578
  WRITE(6,195)TSPLIT
ENDIF
500 CONTINUE

C      WRITE(6,200)
C
50  FORMAT(1H1,lOX,21HSTABILITY OUTPUT DATA,//)
59  FORMAT(SX,23HDATA AT END OF TAKE-OFF)
60  FORMAT(SX,20HMISSION PHASE NUMBER,IS)
100 FORMAT(1OX,12HXGC = ,F15.7,4H ft.)
105 FORMAT(1OX,12HXCAC = ,F15.7,4H ft.)
107 FORMAT(1OX,12HCGBAR = ,F15.7,8H % chord)
110 FORMAT(1OX,12HDCMCL = ,F15.7)
120 FORMAT(1OX,12HNEUTPT = ,F15.7)
130 FORMAT(1OX,16HSTATIC MARGIN = ,F15.7,8H % chord)
135 FORMAT(1OX,12HCGFRWD = ,F15.7,4H ft.)
140 FORMAT(1OX,12HAFT CG = ,F15.7,4H ft.)
150 FORMAT(1OX,12HTAIL SIZE = ,F15.7,8H sq. ft.)
160 FORMAT(1OX,14HCANARD SIZE = ,F15.7,8H sq. ft.)
170 FORMAT(1OX,15HORIZONTAL THRUST,/,10X,12HFORE THRUST=,F15.7,
       $5X,12HAFT THRUST =,F15.7)
180 FORMAT(1OX,21HFORE THRUST POSITION=,F15.7,
       $5X,20HAFT THRUST POSITION=,F15.7)
185 FORMAT(1OX,22HVERT FORE THRUST POS.=,F15.7,5X,
       $22HVERT AFT THRUST POS. =,F15.7)
190 FORMAT(1OX,21HFORE THRUST ANGLE = ,F15.7,5X,
       $22HAFT THRUST ANGLE = ,F15.7)
195 FORMAT(1OX,24THRUST SPLIT (FORE/TOT)=,F15.7)
200 FORMAT(1OX,20HEND STABILITY OUTPUT,//)

C RETURN
C
C--------------------------------------------------------------------------------------
C
SUBROUTINE COPRINT

C INCLUDE 'STBCM.INC'
C
C THIS SUBROUTINE PRINTS OUT THE COMPONENT WEIGHTS AND
C POSITIONS WHEN ICOPRT = 1. THIS IS USED TO DETERMINE
C THE CORRECT POSITION OF THE C.G. OF THE AIRCRAFT.
C
WRITE (6,700)
WRITE (6,702)
WRITE (6,705) WBODY,XBDD
WRITE (6,710) WWING,XWING
WRITE (6,715) WCAND,XCAN
WRITE (6,720) WHT,XHT
WRITE (6,725) WVT,XVT
WRITE (6,730) WLCFRW,XLCFRW
WRITE (6,732) WLCR,XLCR
WRITE (6,735) WNA,XNA
WRITE (6,740) WPIV,XPIV
WRITE (6,745) WAIRC,XAPU
WRITE (6,760) WELT,XELT
WRITE (6,765) WEP,XEP
WRITE (6,770) WINST,XINST
WRITE (6,775) WCREW,XCREW
WRITE (6,780) WCARGO,XCARGO
WRITE (6,800) WAMMUN,XAMMUN
WRITE (6,810) WBB2,XBB2
WRITE (6,860) WBOMB,XBOMB
WRITE (6,880) WFFUS,XFUSF
WRITE (6,890) WFWO,XWOFUEL
WRITE (6,900) WFFUS,XFUSF

DO 795 J = 1,EN
WRITE (6,805) WZENG,XENG(J)
795 CONTINUE
WRITE (6,810) WFS,XFS
WRITE (6,815) WLTFW,XLIFTFW
WRITE (6,820) WPAWT,XPAWT
WRITE (6,830) WCREW,XCREW
WRITE (6,835) WCARGO,XCARGO
WRITE (6,840) WAMMUN,XAMMUN
WRITE (6,855) WBB2,XBB2
WRITE (6,860) WBOMB,XBOMB
WRITE (6,880) WFFUS,XFUSF
WRITE (6,890) WFWO,XWOFUEL
WRITE (6,900) WFFUS,XFUSF

IF (IETANK .EQ. 0) THEN
WRITE (6,1050)
WRITE (6,1051) XEF1
WRITE (6,1052) WZET,WZEF
ELSE
GO TO (871,872,873,874,875,876), IETANK
871 WRITE (6,1100)
WRITE (6,1101) XEF1
WRITE (6,1102) WZET,WZEF
GO TO 900
872 WRITE (6,1200)
WRITE (6,1201) XEF1
WRITE (6,1202) WZET,WZEF
GO TO 900
873 WRITE (6,1300)
WRITE (6,1301) XEF1
WRITE (6,1302) WZET,WZEF
GO TO 900
874 WRITE (6,1400)
WRITE (6,1401) XEF1,XEF2
WRITE (6,1402) WEFF,WEFW
GO TO 900
875 WRITE (6,1500)
WRITE (6,1501) XEF1,XEF2
WRITE (6,1502) WEFF,WEFWT
GO TO 900
876 WRITE (6,1600)
WRITE (6,1601) XEF1, XEF2
WRITE (6,1602) WEFW,WEFWT
GO TO 900
900 CONTINUE
ENDIF
WRITE (6,1700)
C
700 FORMAT(/,5X,40HAIRCRAFT COMPONENT WEIGHTS AND POSITIONS,/)  
702 FORMAT(SX,12H COMPONENT,12H WEIGHT,12H POSITION) 
705 FORMAT(SX,12H AIRFRAME, F12.6,F12.6) 
710 FORMAT(SX,12H WING, F12.6,F12.6)  
715 FORMAT(SX,12H CANARD, F12.6,F12.6)  
720 FORMAT(SX,12H  
725 FORMAT(SX,12H VT, F12.6,F12.6)  
730 FORMAT(SX,12H NOSE GEAR, F12.6,F12.6)  
732 FORMAT(SX,12H MAIN GEAR, F12.6,F12.6)  
735 FORMAT(SX,12H NACELLS, F12.6,F12.6)  
740 FORMAT(SX,12H PIVOTS, F12.6,F12.6)  
745 FORMAT(SX,12H AIR COND., F12.6,F12.6)  
750 FORMAT(SX,12H APU, F12.6,F12.6)  
760 FORMAT(SX,12H AVIONICS, F12.6,F12.6)  
765 FORMAT(SX,12H ELECTRICAL, F12.6,F12.6)  
770 FORMAT(SX,12H INSTURMENTS, F12.6,F12.6)  
775 FORMAT(SX,12H HYDRAULICS, F12.6,F12.6)  
780 FORMAT(SX,12H CONTROL SURF, F12.6,F12.6)  
785 FORMAT(SX,12H PA, F12.6,F12.6)  
787 FORMAT(SX,12H FURNISHINGS, F12.6,F12.6)  
805 FORMAT(SX,12H ENGINES, F12.6,F12.6)  
810 FORMAT(SX,12H FUEL SYS., F12.6,F12.6)  
815 FORMAT(SX,12H LIFT FAN, F12.6,F12.6)  
820 FORMAT(SX,12H PAYLOAD, F12.6,F12.6)  
830 FORMAT(SX,12H CREW, F12.6,F12.6)  
835 FORMAT(SX,12H CARGO, F12.6,F12.6)  
840 FORMAT(SX,12H AMMUNITION, F12.6,F12.6)  
855 FORMAT(SX,12H BB2, F12.6,F12.6)  
860 FORMAT(SX,12H BOMBS, F12.6,F12.6)  
865 FORMAT(SX,12H MISSILES, F12.6,F12.6)  
870 FORMAT(SX,25EXTERNAL TANK(S) AND FUEL)  
880 FORMAT(SX,12H FUS. FUEL, F12.3,F12.6)  
890 FORMAT(SX,12H WING FUEL, F12.3,F12.6)  
1050 FORMAT(SX,20HCENTERLINE FUEL TANKS)  
1051 FORMAT(SX,15EXT. TANK POS.=,F12.6)  
1052 FORMAT(SX,8HW. TANK=,F12.6,5X,8HW. FUEL=,F12.6)  
1100 FORMAT(SX,19HTWO FUS. SIDE TANKS)  
1101 FORMAT(SX,15EXT. TANK POS.=,F12.6)  
1102 FORMAT(SX,8HW. TANK=,F12.6,5X,8HW. FUEL=,F12.6)  
1200 FORMAT(SX,18HTWO C/4-WING TANKS)  
1201 FORMAT(SX,15EXT. TANK POS.=,F12.6)  
1202 FORMAT(SX,8HW. TANK=,F12.6,5X,8HW. FUEL=,F12.6)  
1300 FORMAT(SX,17HTWO WINGTIP TANKS)
SUBROUTINE STBDT(ICALC, NERR, ICEO, KCPRNT, IGPLT)

   COMMON/GLOBCM/ARAY(1400), IARAY(100)
   COMMON/ACSN2/INOUT(1600), ILOCAT(30, 4)

   STABILITY AND CONTROL COMMON - MODULE 2.

   COMMON /STBCM/ RA5(134), IA5(2)

   AERODYNAMICS COMMON - MODULE 3.

   COMMON/AEROCM/ RA3(170), IA3(10)

   DATA TRANSFER INTERFACE BETWEEN STABILITY AND AERODYNAMICS.

   OPERATION SEQUENCE.
   1 - TRANSFER DATA FROM STABILITY LOCAL COMMON TO GLOBAL COMMON.
   2 - TRANSFER DATA FROM GLOBAL TO AERODYNAMICS LOCAL COMMON.
   3 - CALL AERODYNAMICS.
   4 - TRANSFER DATA FROM AERODYNAMICS LOCAL COMMON TO GLOBAL.
   5 - TRANSFER DATA FROM GLOBAL COMMON TO STABILITY LOCAL COMMON.

   JCALC=1
   KCALC=2
   NERR = 0

   STEP 1.
   NSTRTR=ILOCAT(5,1)
   NMAXR=ILOCAT(5,2)
   NSTRTI=ILOCAT(5,3)
   NMAXI=ILOCAT(5,4)
   CALL DATAIO(KCALC, NMAXR, NMAXI, INOUT(NSTRTR), INOUT(NSTRTI),
               RA5, IA5, RARRAY, IARAY)
STEP 2.
NSTRTR=ILOCAT(3,1)
NMAXR=ILOCAT(3,2)
NSTRTI=ILOCAT(3,3)
NMAXI=ILOCAT(3,4)
CALL DATAIO(JCALC,NMAXR,NMAXI,INOUT(NSTRTR),INOUT(NST RTI),
  * RA3,IA3,RARAY,IARAY)

STEP 3.
CALL AERO2(ICALC,NERR,ICEO,KCPRNT,IGPLT)

STEP 4.
CALL DATAIO(KCALC,NMAXR,NMAXI,INOUT(NSTRTR),INOUT(NST RTI),
  * RA3,IA3,RARAY,IARAY)

STEP 5.
IF(NERR.GT.1) RETURN
NSTRTR=ILOCAT(5,1)
NMAXR=ILOCAT(5,2)
NSTRTI=ILOCAT(5,3)
NMAXI=ILOCAT(5,4)
CALL DATAIO(JCALC,NMAXR,NMAXI,INOUT(NSTRTR),INOUT(NST RTI),
  * RA5,IA5,RARAY,IARAY)

THE FOLLOWING IS A LIST OF THE REFERENCES
USED IN THIS SUBROUTINE:

1) HOAK, D.E. et al.; "USAF Stability and Control DATCOM";
   Wright Patterson AFB, Ohio, 45433; Revised 1970

2) Roskam, Jan; "Airplane Flight Dynamics and Automatic
   Controls, Part 1"; Published by the author, 519 Boulder
   Ave. Lawrence KA 66044; second printing 1979

3) McCormick, B.W.; "Aerodynamics, Aeronautics, and Flight
   Dynamics"; J. Wiley and Sons, 1979

4) Perkins C.D. and Hage R.E.; "Airplane Performance, Stability
   and Control"; J. Wiley and Sons, 1949

5) Curry, Norman; "Aircraft Landing Gear Design, Principles
   and Practices"; AIAA Education Series; American Institute
   of Aeronautics and Astronautics; 1988

RETURN
END
This contains the common block statements needed to run the stability
and control module 'STBLCON.FOR'. This file is connected with the
module through the 'INCLUDE' statement at the beginning of each
of the subroutines.

COMMON/OVER/ICALC,NERROR,MODNO,IGEO,KGPRNT,ICPLT,IF

COMMON/STBCM/CLALFA,CLAFHT,DEDA,XQCBCN,XQCBHT,XQCBVT,
1 CBARCN,CBARWC,CBARTI,ROOTWC,TRWC,XLEWC,SPANWC,
2 SWPWC,POWL,EN,BOPL,XLEPBO(10),WCAND,WFTOT,FUFRAC,WING,WFS,
3 WLG,WHT,WVT,WETANK,WFEWT,WBDOD,MAPIU,WAPU,WSC,WENC,SNW,WG,SHT,
4 SWT,WCREW,WELT,WEP,WMDP,WINS,WN,WPA,WCARO,WMANIR,WBDOM2,WMISS,
5 ARW,W1M,DUHOT,ZRHOT,AVRT,SWPCN,TRCN,AMTO,
6 ZRRCN,WBLD,BDMAX,FRN,FRAB,ALPHA,CL,THRT,ARHT,TNT(12),WFT(12),
7 WFT0,WFT1,ARCH,CWMB,WGTO,WFIGW,WFB2,WAF,WE,WP,WPS,WSTUM,WFUEL,
8 TWT0,MACH,ALT,SWETWC,SWETHT,SWETCN,SPANHT,SPANCN,
9 STARIN(12),DUMMY,IAO

COMMON/LOCALS/ETAH-1,XFAN,XFHT,XFNA,XFVT,XFWG,XFAPU,XFENG,
1 XFELT,XFEP,XFINST,WFHDP,WFPA,WFFY,WFY,XFCREW,WFPAYL,WFFUR,
2 XFLOFRT,WFLOFRT,XFLGR,XFFS,XFBBM,XFSC,XFPIV,XFLITF,WPIV,WFFR,
3 XEF1,XFMAMN,XFBB2,XFMISS,XFCARGO,XACNG,ETACN,ODENA,XWAB,CLACN,
4 WZET,WZEF,WIFF,WIFF,WIFF,ZWET,WZEF,SSMMO,SHMMO,SHL,WK,WK,
5 IDBPRT,XCG(13),DCMCL(13),SM(13),XCGAFT(14),WFLOFRT,WFGR,WFFG,
6 WFP,WZB,ZM,WZB,IXS,IMS,IAHS,IMPHASE,ICMPT,WIFTF,WFFUS,WFFS,
7 LCN,CELY,CALHT,TZCHT,FINEUPT(13),XFBOD,XFWG,XPAYL,XCREW,WPAYL,
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9 ZMTER,AWRT,XMTR,XWTC,XWTC,ZMTR,XWTC,ZMTR,XWTC,ZMTR,XWTC,
10 TSPLIT,CMAT1,CMAT2,HTH,ICN,ELVDEF,VRD1,ALPHACN,TAU1,T2,
11 SHT1,SCN1,ELDVMAX,IGW,XCGFRT(13),QTR,QCGAR(13),ALFO,CLDA,
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