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

National Aeronautics and Space Administration
Contract Number NCC 2-236
August 1990

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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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January 1990

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, $BR = ((c_b/ c_f)^2 - (t/2\ c_f)^2)^{1/2}$
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
$C_{L\alpha HT}$	- 3-dimensional tail lift-curve slope
$C_{L\alpha CN}$	- 3-dimensional canard lift-curve slope
$C.M._{\alpha}$	- Pitching moment coefficient change due to change in angle of attack
CM_A	- Moment coefficient about point A
CM_{cg}	- Moment coefficient about the Center of Gravity
CM_{mg}	- Moment coefficient about the main landing gear
CM_{TH}	- Contribution of thrust to the pitching moment
CM_{wb}	- Wing-Body pitching moment
D	- Aircraft drag
dC_m/dC_l	- Total pitching moment-curve slope
$dC_m/dC_{l\ CN}$	- Pitching moment-curve slope contribution of the canard

$dC_m/dC_{l\ f}$	- Pitching moment-curve slope contribution of the fuselage
$dC_m/dC_{l\ HT}$	- Pitching moment-curve slope contribution of the tail
$dC_m/dC_{l\ TH}$	- Pitching moment-curve slope contribution of the thrust of the engine(s)
$dC_m/dC_{l\ wing}$	- Pitching moment-curve slope contribution of the wing
$d\varepsilon/d\alpha$	- Change in downwash angle with respect to change in angle of attack
$d\varepsilon/d\alpha_{up}$	- Change in upwash angle with respect to change in angle of attack
$d\delta/d\alpha$	- Change in elevator angle due to change in angle of attack
dg	- 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_{HT}	- Height of aft horizontal tail above ground
h_{HT}	- Height of aft horizontal tail above wing chord plane
i_{CN}	- Canard incidence angle (degrees)
i_{HT}	- Tail incidence angle (degrees)
i_w	- Wing incidence angle (degrees)
K_f	- Fuselage coefficient
L_{CN}	- Canard lift (pounds force)
L_{eff}	- Effective length from wing to horizontal tail (feet)
L_{HT}	- Tail lift (pounds force)
L_w	- Wing Body lift (pounds force)

M_A	- Moment about point A
MAC	- Mean Aerodynamic Chord
M_∞	- Free stream Mach number
N	- Number of engines
N_0	- Neutral Point
R	- Reaction force of aircraft on the ground (pounds force)
S	- Wing surface area (sq. feet)
S_{CN}	- Canard surface area (sq. feet)
S_{HT}	- Tail surface area (sq. feet)
SM	- Static Margin (percent wing chord)
T	- Total engine thrust (pounds force)
T_1	- Thrust of forward thrust vector (pounds force)
T_2	- Thrust of aft thrust vector (pounds force)
t/c	- Elevator thickness-to-chord ratio
V	- Aircraft acceleration
W	- Aircraft gross weight (pounds force)
X_A	- Distance from quarter-chord of wing center to aerodynamic center (feet)
X_{ac}	- Wing aerodynamic center (percent wing chord)
$X_{ac\ cn}$	- Canard aerodynamic center (percent canard chord)
$X_{ac\ ht}$	- Tail aerodynamic center (percent tail chord)
$X_{ac\ wb}$	- Aerodynamic center of wing-body combination (percent wing chord)
X_{CG}	- Distance from nose to aircraft C.G. (feet)
$X_{CG\ aft}$	- Aft C.G. limit (feet)

$X_{CG\ for}$	- Forward C.G. limit (feet)
X_{LE}	- Leading edge of wing Mean Aerodynamic Chord from nose (feet)
$X_{LE\ root}$	- Leading edge of wing root from nose (feet)
X_{mg}	- Distance from C.G. to main gear (feet)
X_{T1}	- Distance from forward thrust vector to C.G. (% fuselage length)
X_{T2}	- Distance from aft thrust vector to C.G. (% fuselage length)
Z_D	- Vertical distance from drag center to main gear (feet)
Z_{mg}	- Vertical height from main gear to C.G. (feet)
Z_{T1}	- Vertical thrust line from forward thrust vector to C.G. (feet)
Z_{T2}	- Vertical thrust line from aft thrust vector to C.G. (feet)
α	- Aircraft angle of attack (degrees)
α_w	- Wing angle of attack (degrees)
δ_e	- Elevator deflection for take off rotation (degrees)
δ_{emax}	- Maximum elevator deflection for landing (degrees)
ΔC_{Lf}	- Change in lift coefficient due to flaps
$\Delta\varepsilon_g$	- Change in downwash due to ground effects (degrees)
ε	- Downwash angle of wing on aft horizontal tail (degrees)
ε_g	- Downwash angle in ground effect (degrees)
ε_v	- Downwash angle in the wing vortex core (degrees)

ε_{up} - Upwash angle of wing on forward canard (degrees)

Λ - Wing sweep angle (degrees)

γ_1 - Forward thrust vector angle (degrees)

γ_2 - Aft thrust vector angle (degrees)

λ - Wing taper ratio

η_{CN} - Canard efficiency

η_{HT} - Tail efficiency

Γ - Wing dihedral angle (degrees)

τ - 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.

$$X_{cg_{fus}} = Xf_{fus} * BODL \tag{2}$$

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 then 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 of 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.

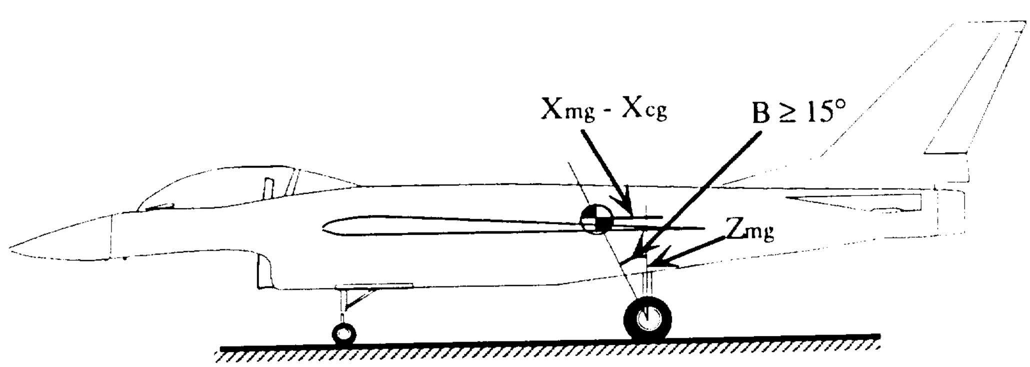

Figure 1: Aft C.G. Limits Relative to Main Landing Gear

1 Curry, Norman S.; Aircraft Landing Gear Design: Principles and Practices, AIAA Education Series; American Institute of Aeronautics and Astronautics, Inc. 370 L'Enfant Promenade, S.W., Washington D.C., 20024; 1988; 43 - 50.

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

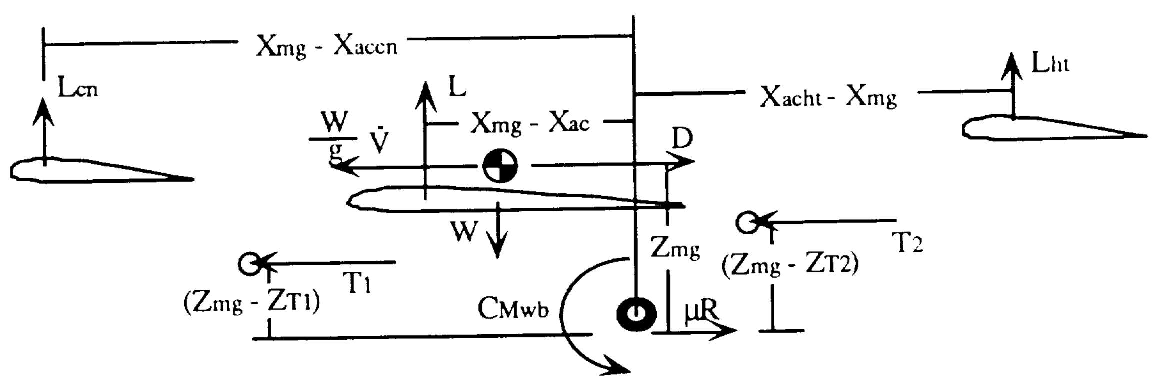

Figure 2: Summation of Moments about the Main Landing Gear

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 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

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[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[3] is seen in Equation 3. This is modified to account for any vertical thrust offset as specified by the designer.

$$M_{mg} = -C_{Mwb} q\, S\, c - \left[Z_{mg} - Z_{T_1}\right] T_1 - \left[Z_{mg} - Z_{T_2}\right] T_2 - \left[X_{mg} - X_{cg}\right] W + \left[X_{mg} - X_{ac}\right] L_{wb} + \left[Z_{mg} - Z_D\right] D + \frac{W}{g} \dot{V} Z_{mg} - \left[X_{ac_{ht}} - X_{mg}\right] L_{ht} \tag{3}$$

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

$$L_{wb} = C_{L_{\alpha\, g}} \alpha_w q\, S \tag{4}$$

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

$$\frac{W}{g} \dot{V} = T - D - \mu R \tag{5}$$

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[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.

$$R = W - L_{wb} - L_{ht} \tag{6}$$

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.

$$L_{ht} = \frac{L_1 + L_2 - L_4 - L_6 + L_7 - L_8}{\Delta} \tag{7}$$

where

$$L_1 = - C_{Mwb}\, q\, S\, c \qquad L_6 = \left[X_{mg} - X_{cg} + \mu\, Z_{mg}\right] W$$

$$L_2 = \left[X_{mg} - \left(X_{cg} - X_{T_1}\right) + \mu Z_{mg}\right] T_1 \qquad L_7 = \left[X_{mg} - X_{ac_{wb}} + \mu\, Z_{mg}\right] L_{wb}$$

$$L_8 = \left[Z_D\right] D$$

$$L_4 = \left[X_{T_2} + X_{cg} - X_{mg} - \mu Z_{mg}\right] T_2 \qquad \Delta = \left[X_{ac_{ht}} - X_{mg} - \mu\, Z_{mg}\right]$$

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.

$$L_{ht} = C_{L\alpha ht\,g}\left(\alpha_w + i_{ht} - i_w + \tau\,\delta_e\right) q\, S_{ht} \; - C_{L\alpha ht}\,\varepsilon_g\, q\, S_{ht} \tag{8}$$

Where the angle of attack of the tail as shown in Figure 3. Rearranging Equation 8,

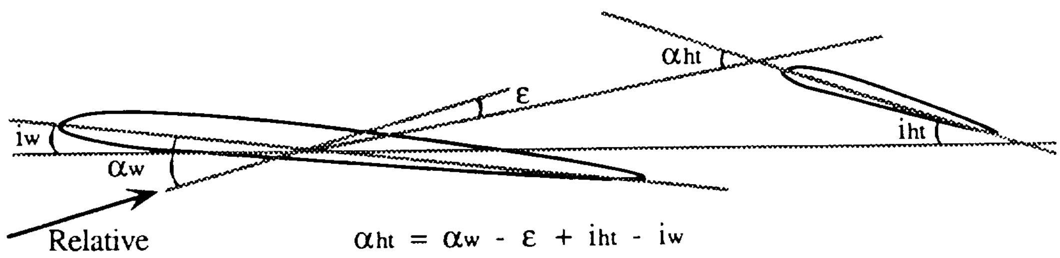


Figure 3: Aft Horizontal Tail Angle of Attack

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

$$S_{ht} = \frac{L_{ht}}{C_{L\alpha ht\,g}\left(\alpha_w + i_{ht} - i_w + \tau\,\delta_e\right) q \; - C_{L\alpha ht}\,\varepsilon_g\, q} \tag{9}$$

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.

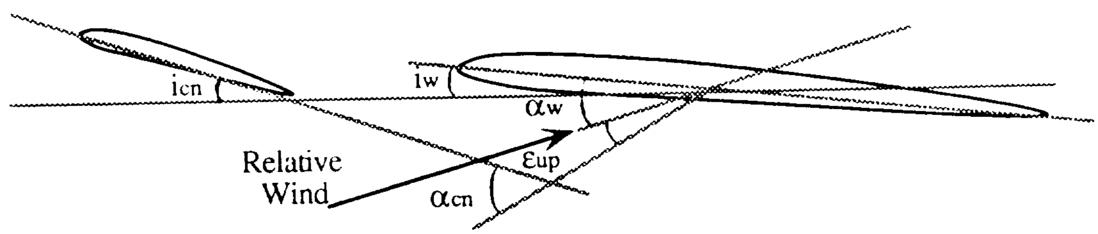
$$L_{ht} = \frac{L_1 + L_2 - L_4 - L_6 + L_7 - L_8 + L_9}{\Delta} \tag{10}$$

where

$$L_9 = \left[X_{mg} - X_{ac_{cn}} + \mu\, Z_{mg}\right] C_{L\alpha cn\,g}\left(\alpha_w\, \varepsilon_{up} + i_{cn} - i_w\right)$$

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



$$\alpha_{cn} = \alpha_w + \varepsilon_{up} + i_{cn} - i_w$$

Figure 4: Forward Canard Angle of Attack

about the point of ground contact with a forward canard and any thrust offset.

$$M_{mg} = -C_{M_{wb}} q S c - \left[Z_{mg} - Z_{T_1}\right] T_1 - \left[Z_{mg} - Z_{T_2}\right] T_2 - \left[X_{mg} - X_{cg}\right] W - \left[X_{ac_{wb}} - X_{mg}\right] L_{wb} + \left[Z_{mg} - Z_D\right] D + \frac{W}{g} \dot{V} Z_{mg} + \left[X_{mg} - X_{ac_{cn}}\right] L_{cn} \tag{11}$$

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

$$L_{cn} = \frac{-L_1 - L_2 + L_4 + L_6 + L_7 + L_8}{\Delta} \tag{12}$$

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 Δ 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

$$S_{cn} = \frac{L_{cn}}{C_{L\alpha\ g}\left(\alpha_w + \varepsilon_{up} + i_{cn} - i_w + \tau\,\delta_c\right) q} \tag{13}$$

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

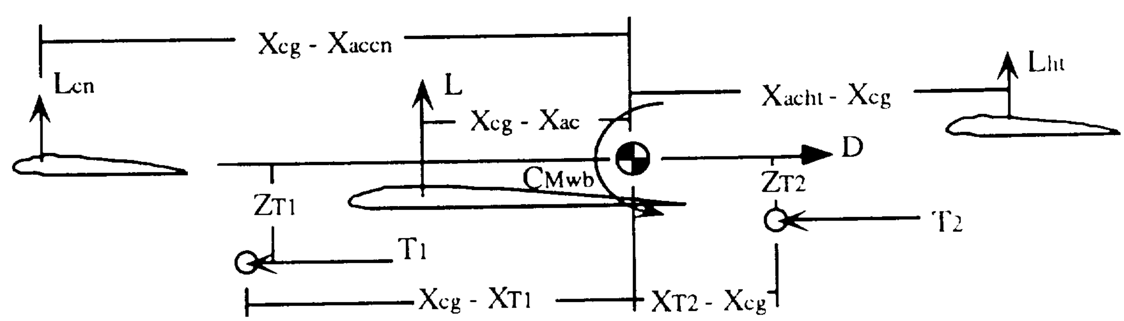


Figure 5: Summation of Moments about the Center of Gravity

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.

$$C_{M_{cg}} = C_L \frac{X_{cg} - X_{ac}}{c} - C_{M_{wb}} - C_{L\alpha_{HT}}\left(\alpha_w - \varepsilon + i_{HT} - i_w + \tau\,\delta_e\right) \frac{S_{HT}}{S} \frac{\left(X_{ac_{HT}} - X_{cg}\right)}{c} \eta_{HT}$$
$$+ C_{L\alpha_{CN}}\left(\alpha_w + \varepsilon_{up} + i_{CN} - i_w + \tau\,\delta_e\right) \frac{S_{CN}}{S} \frac{\left(X_{cg} - X_{ac_{CN}}\right)}{c} \eta_{CN} + \frac{\left(T_1 Z_{T1} + T_2 Z_{T2}\right)}{q\,S\,c} N \qquad (14)$$

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.

$$Xcg_{forc} = \frac{C_L X_{ac} + c\, C_{M_{wb}} + C_{L\alpha_{HT}}\left(\alpha_W - \varepsilon + i_{HT} - i_w + \tau\,\delta_e\right) \frac{S_{HT}}{S} X_{ac_{HT}} \eta_{HT}}{\Delta}$$
$$+ \frac{C_{L\alpha_{CN}}\left(\alpha_W + \varepsilon_{up} + i_{CN} - i_w + \tau\,\delta_e\right) \frac{S_{CN}}{S} X_{ac_{CN}} \eta_{CN} - \frac{\left(T_1 Z_{T1} + T_2 Z_{T2}\right)}{q\,S} N}{\Delta} \qquad (15)$$

where

$$\Delta = C_L + C_{L_{HT}}\left(\alpha_w - \varepsilon + i_{HT} - i_w + \tau\,\delta_e\right) \frac{S_{HT}}{S} \eta_{HT}$$
$$+ C_{L_{CN}}\left(\alpha_w + \varepsilon_{up} + i_{CN} - i_w + \tau\,\delta_e\right) \frac{S_{CN}}{S} \eta_{CN}$$

In Equation 15, the C_L is the maximum lift coefficient, and δ_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

$$\frac{dC_{M\,cg}}{d\alpha} = 0 \tag{16}$$

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

$$\begin{aligned} C_{M\alpha} = {} & C_{L\alpha}\left(X_{cg} - X_{ac}\right) - C_{L\alpha_{HT}}\eta_{HT}\frac{S_{HT}}{S}\left(X_{ac_{HT}} - X_{cg}\right)\left(1 - \frac{d\varepsilon}{d\alpha} + \tau\frac{d\delta_{ht}}{d\alpha}\right) \\ & + C_{L\alpha_{CN}}\eta_{CN}\frac{S_{CN}}{S}\left(X_{cg} - X_{ac_{CN}}\right)\left(1 - \frac{d\varepsilon}{d\alpha_{up}} + \tau\frac{d\delta_{cn}}{d\alpha}\right) \end{aligned} \tag{17}$$

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 $C_{M\alpha}$ 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

$$N_0 = \frac{\overline{X_{ac_{wb}}} + \overline{X_{ac_{HT}}}\,\frac{C_{L\alpha_{HT}}}{C_{L\alpha}}\frac{S_{HT}}{S}\eta_{HT}\left(1 - \frac{d\varepsilon}{d\alpha} + \tau\frac{d\delta}{d\alpha}\right)}{\Delta} + \frac{\overline{X_{ac_{CN}}}\,\frac{C_{L\alpha_{CN}}}{C_{L\alpha}}\frac{S_{CN}}{S}\eta_{CN}\left(1 - \frac{d\varepsilon}{d\alpha_{up}} + \tau\frac{d\delta}{d\alpha}\right)}{\Delta} \tag{18}$$

where

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

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.

$$Xcg_{aft} = X_{LE} + MAC\ N_0 \tag{19}$$

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 ($dC_{M_{cg}}/dC_L$) or the static margin (SM) must be determined. The aircraft is stable if it has a negative value for the dC_M/dC_L, the larger the negative value, the more stable the aircraft. The equation used in solving for the 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.)

$$\begin{aligned} C_{M_{cg}} = {} & C_L \frac{X_{cg} - X_{ac}}{c} - C_{M_{wb}} - C_{L\alpha_{HT}}\left(\alpha_W - \varepsilon + i_{HT} - i_w + \tau\,\delta_e\right) \frac{S_{HT}}{S} \frac{(X_{ac_{HT}} - X_{cg})}{c} \eta_{HT} \\ & + C_{L\alpha_{CN}}\left(\alpha_W + \varepsilon_{up} + i_{CN} - i_w + \tau\,\delta_e\right) \frac{S_{CN}}{S} \frac{(X_{cg} - X_{ac_{CN}})}{c} \eta_{CN} + \frac{(T_1 Z_{T1} + T_2 Z_{T2})}{q\,S\,c} N \end{aligned} \tag{20}$$

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

$$\begin{aligned} \frac{dC_M}{dC_L} = {} & \frac{X_{cg} - X_{ac}}{c} - \frac{dC_{M_{wb}}}{dC_L} - \frac{C_{L\alpha_{HT}}}{C_{L\alpha}} \frac{S_{HT}}{S} \frac{(X_{ac_{HT}} - X_{cg})}{c} \eta_{HT}\left(1 - \frac{d\varepsilon}{d\alpha}\right) \\ & + \frac{C_{L\alpha_{CN}}}{C_{L\alpha}} \frac{S_{CN}}{S} \frac{(X_{cg} - X_{ac_{CN}})}{c} \eta_{CN}\left(1 - \frac{d\varepsilon}{d\alpha}\right)_{up} + \frac{(T_1 Z_{T1} + T_2 Z_{T2})}{W\,c} N \end{aligned} \tag{21}$$

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[4]

$$\frac{dCm}{dC_{L\ wing}} = \frac{Xcg - Xac}{c} \tag{22}$$

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_{Mwb}/dC_L, is estimated using the equation[5]

$$\frac{dC_{Mwb}}{dC_L} = \frac{K_f\ BDMAX^2\ BODL}{S\ c\ C_{L\alpha}} \tag{23}$$

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[6]

$$\frac{dCm}{dC_{L\ HT}} = \frac{-a_{HT}}{a}\ \frac{S_{HT}}{S}\ \frac{X_{acHT} - X_{cg}}{c}\ \eta_{HT}\left(1 - \frac{d\varepsilon}{d\alpha}\right) \tag{24}$$

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4 Perkins, C.D. and Hage, R.E., Airplane Performance, Stability and Control, J. Wiley and Sons, 1949. 216-218

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

$$\frac{dCm}{dC_{L\,CN}} = \frac{a_{CN}}{a}\,\frac{S_{CN}}{S}\,\frac{X_{CN}}{c}\,\eta_{CN}\left(1 - \frac{d\varepsilon}{d\alpha}\right)_{up} \tag{25}$$

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.

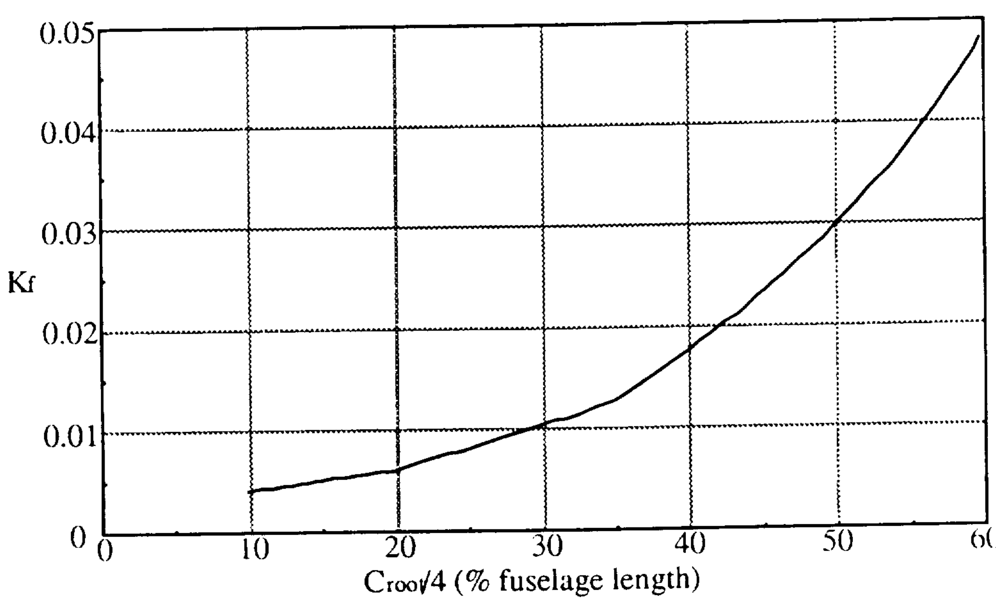

Figure 6: Fuselage Stability Coefficient (K_f)
vs.
Position of Wing Root Quarter-Chord ($C_{root}/4$)
(Reproduced from Reference 4)

The contribution of engine thrust is calculated using Equation 26, which assumes that the lift equals the weight of the aircraft.

$$\frac{dC_{M_{Tn}}}{dC_L} = \left(\frac{T_1\, Z_{T1} + T_2\, Z_{T2}}{W\;c}\right) N \tag{26}$$

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 then 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

$$SM = N_0 - \frac{X_{cg} - X_{LE}}{C} \tag{27}$$

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_{eff}), the effective wing span (b_{eff}), and the tail height above or below the trailing wing vortex. This method assumes a large wing span to horizontal tail span ratio ($b/b_{ht} \geq 1.5$), wing trailing edge vortex separation, and subsonic flow.

First AR_{eff} and b_{eff} are determined. These are determined from Figure 7, dependent on the angle-of-attack and geometry of the wing. Once the AR_{eff} and the b_{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.

$$b_v = b_{eff} - \left(b_{eff} - b_{v_{ru}}\right)\left(\frac{2\,L_{eff}}{b\,\xi_{ru}}\right)^{1/2} \tag{28}$$

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[7] Hoak, D.E. et al; USAF Stability and Control Datcom; Wright Patterson AFB Ohio, 45433; Revised 1970; sect. 4.4.1

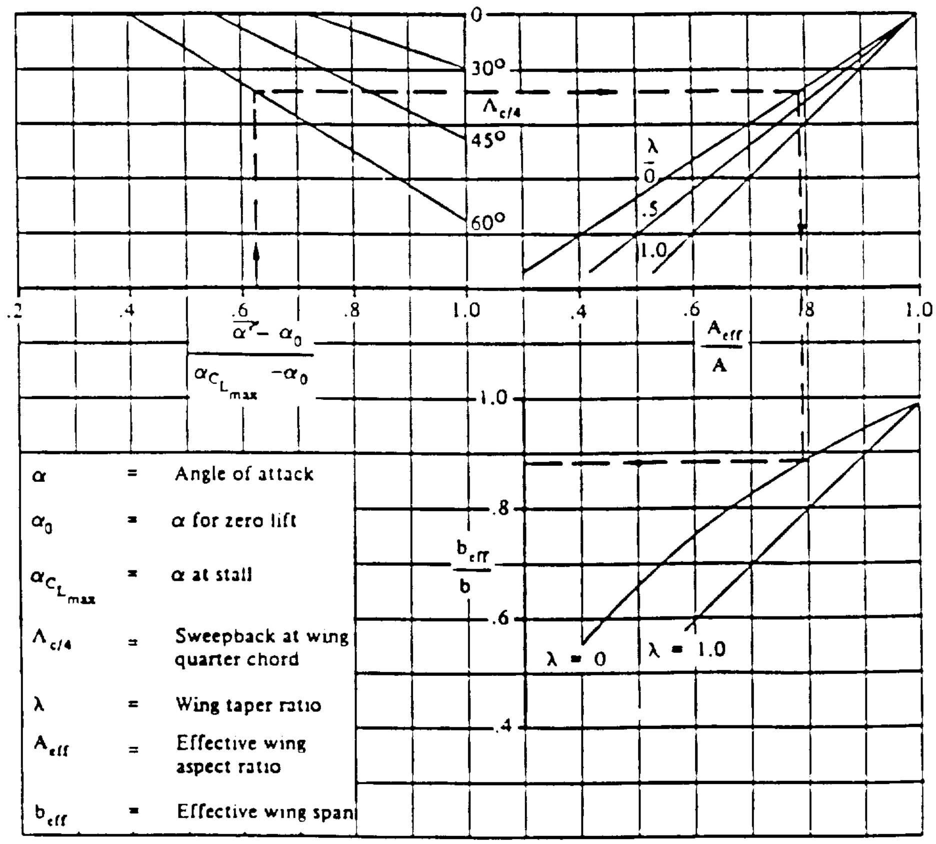

Figure 7: Effective Wing Aspect Ratio and Span
- Low Speeds
(Reproduced From Reference 1)

where L_{eff} is the distance from the wing tip trailing edge to the horizontal tail quarter-chord and ξ_{ru} is a dummy variable. The following are used in solving for b_v.

$$b_{v_{ru}} = \left[0.78 + 0.10(\lambda - 0.4) + 0.003\, \Lambda_{c/4}\right] b_{eff} \tag{29}$$

and

$$\xi_{ru} = \frac{0.56\, AR_{eff}}{C_L} \tag{30}$$

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.

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

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

$$\frac{\varepsilon}{\varepsilon_v} = \frac{1}{1 + \left(\frac{2\, a}{b_v}\right)^2} \tag{32}$$

where the downwash at the vortex core is determined from

$$\varepsilon_v = \frac{1.62\, C_L}{\pi\, AR} \tag{33}$$

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_{root}), and the distance the canard is ahead of the wing quarter-chord. The upwash effects are determined using Figure 8[8]. This solution is accurate only for Mach numbers less then one, and as such currently limits the analysis to subsonic missions.

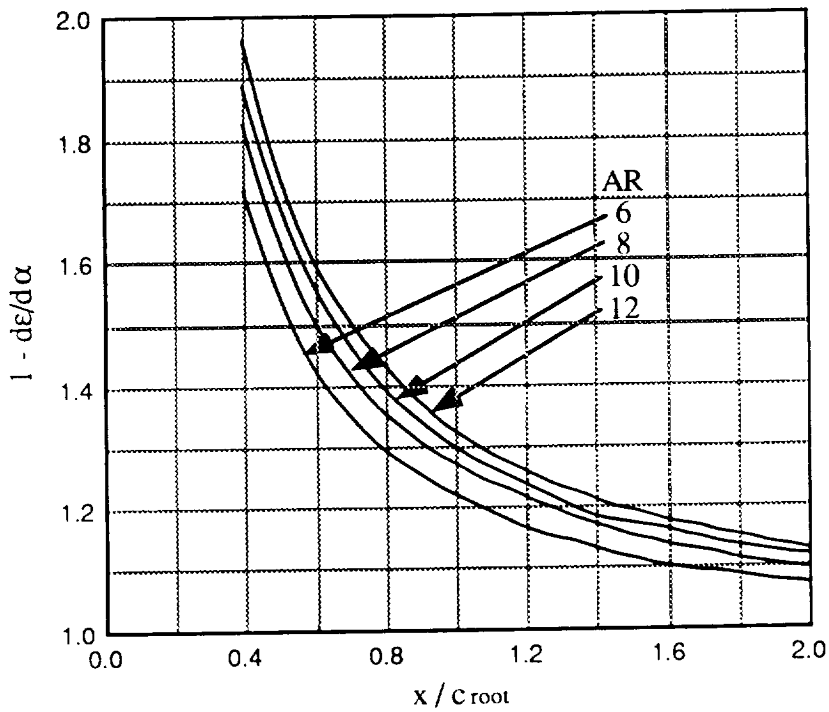

Figure 8: Upwash Effects in Front of the Wing(1 - d ε/dα)
vs.
Forward Position in Percent Wing Quarter-Chord (x/c_{root})
as a Function of Wing Aspect Ratio (AR)

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

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8 McCormick, B.W., 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[9]. Once the change in the lift curve slope is determined, the wing lift in ground effect is calculated using

$$L_{wb} = C_{L_{\alpha\, g}} \alpha_w q\, S \tag{34}$$

the horizontal tail lift is determined using

$$L_{ht} = C_{L_{\alpha ht\, g}} \left(\alpha_w + i_{ht} - i_w + \tau\, \delta_c \right) q\, S_{ht} - C_{L_{\alpha ht}} \varepsilon_g\, q\, S_{ht} \tag{35}$$

and the canard lift is found using

$$L_{cn} = C_{L_{\alpha cn\, g}} \left(\alpha_w + \varepsilon_{up} + i_{cn} - i_w + \tau\, \delta_c \right) q\, S_{cn} \tag{36}$$

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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9 Perkins, C.D. and Hage, R.E.; Airplane Performance, Stability and Control; J. Wiley and Sons, 1949; 257.

geometric and aerodynamic parameters[10]. 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,

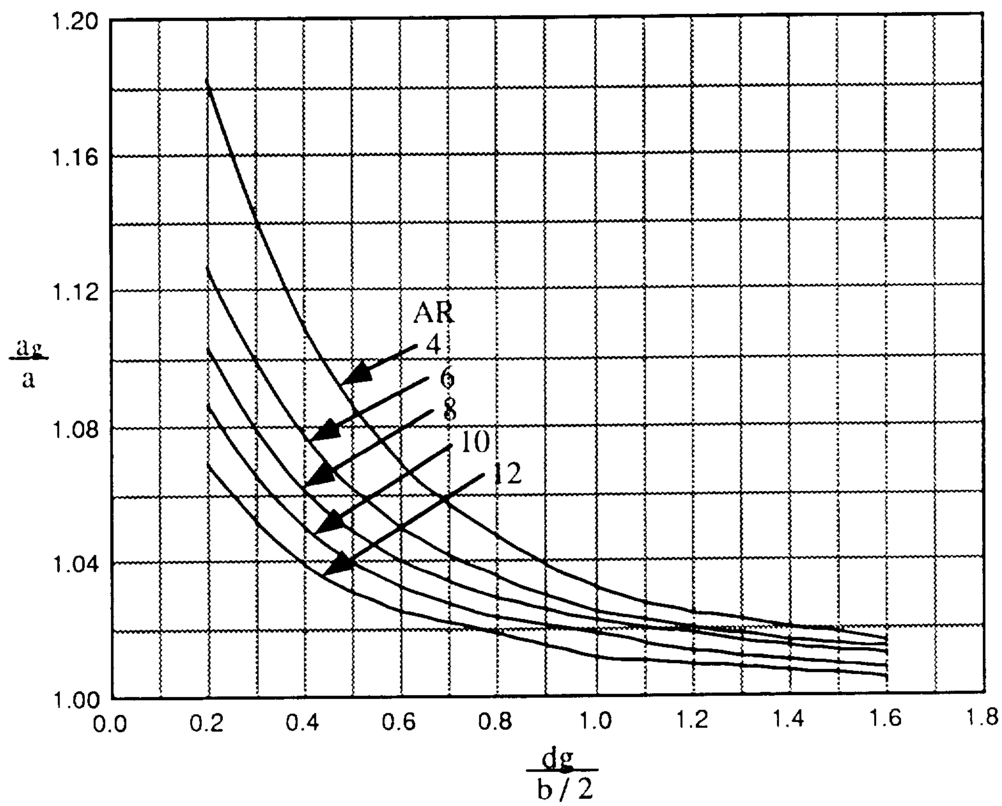

Figure 9: Ratio of Lift Curve Slopes in and out of Ground Effects (a_g/a) vs. Height above Ground in Semi-Spans (dg/b/2) as a Function of Aspect Ratio (AR)

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

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[10] Hoak, D.E. et al; USAF Stability and Control Datcom; Wright Patterson AFB Ohio, 45433; Revised 1970; sect. 4.7.1

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.

$$b_{eff} = \left[\frac{C_{Lw} + \Delta C_{L_f}}{\dfrac{C_{Lw}}{b_w^{'}} + \dfrac{\Delta C_{L_f}}{b_f^{'}}} \right] \tag{37}$$

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.

$$\Delta\varepsilon_g = \varepsilon \left[\frac{b_{eff}^2 + 4\left(H_{HT} - H\right)^2}{b_{eff}^2 + 4\left(H_{HT} + H\right)^2} \right] \tag{38}$$

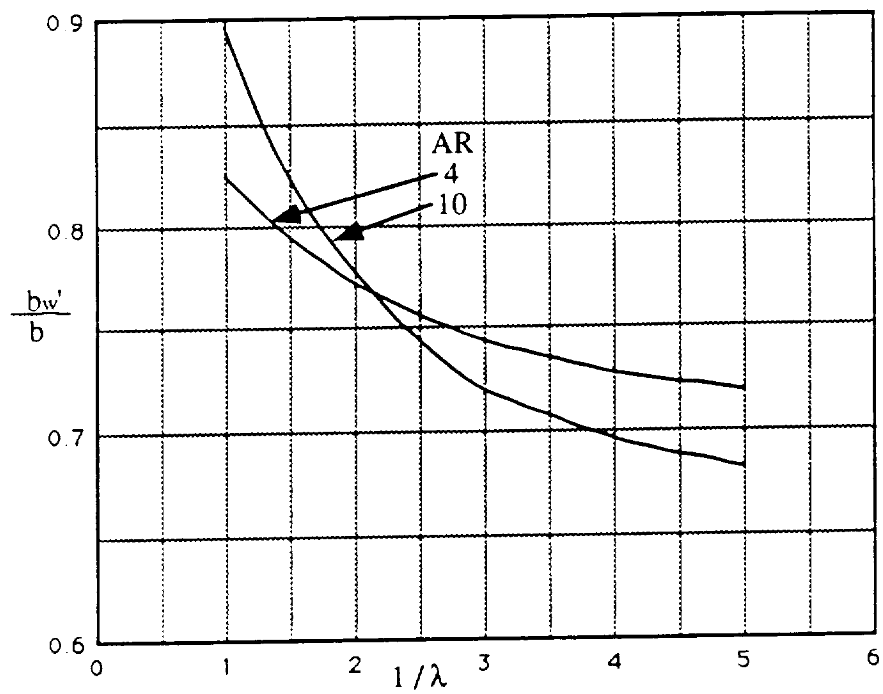

Figure 10: Effective Span of Wing in Presence of Ground
(Reproduced from reference 1)

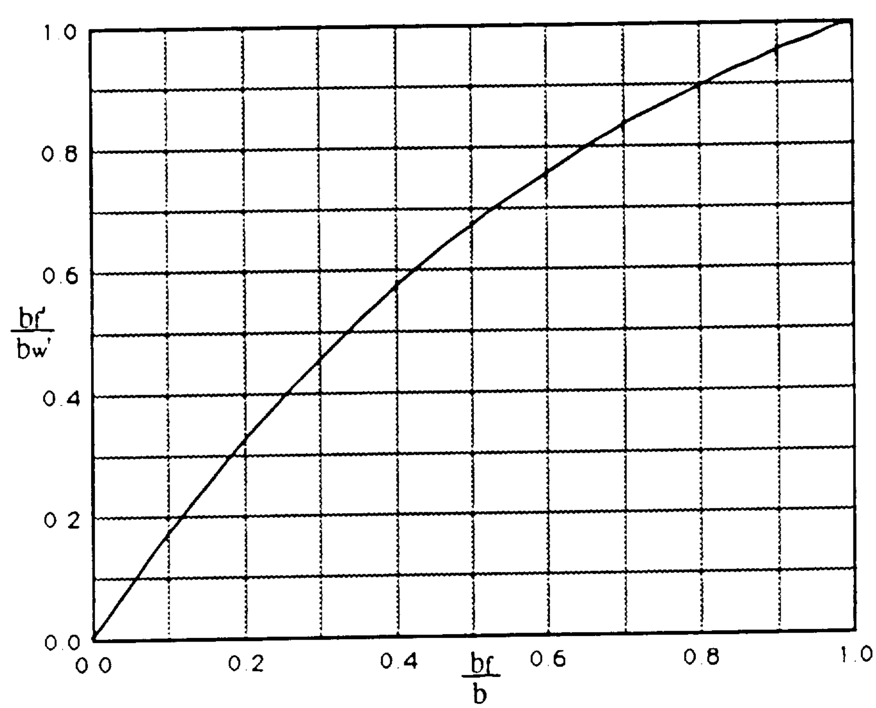

Figure 11: Effective Span of Flaps in Presence of Ground
(Reproduced from reference 1)

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 where 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 take off 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 effect 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

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

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

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

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

	Production	Computer Model	% Difference
Horizontal CanardSize	66.7	64.09	3.91
Main Gear Position	34.8	36.14	3.71
MAC	17.15	17.94	4.40
Quarter-chord Position	35.3	32.64	7.54
Half-chord Position	39.64	37.13	6.33
Forward C.G.	*	31.40	
Aft C.G.	*	36.05	
Quarter-chord of Horizontal Tail	-	-	-

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 cananrd size is in feet squared.

*: The author was unable to obtain information on the forward or aft C.G. limits of the Production aircraft

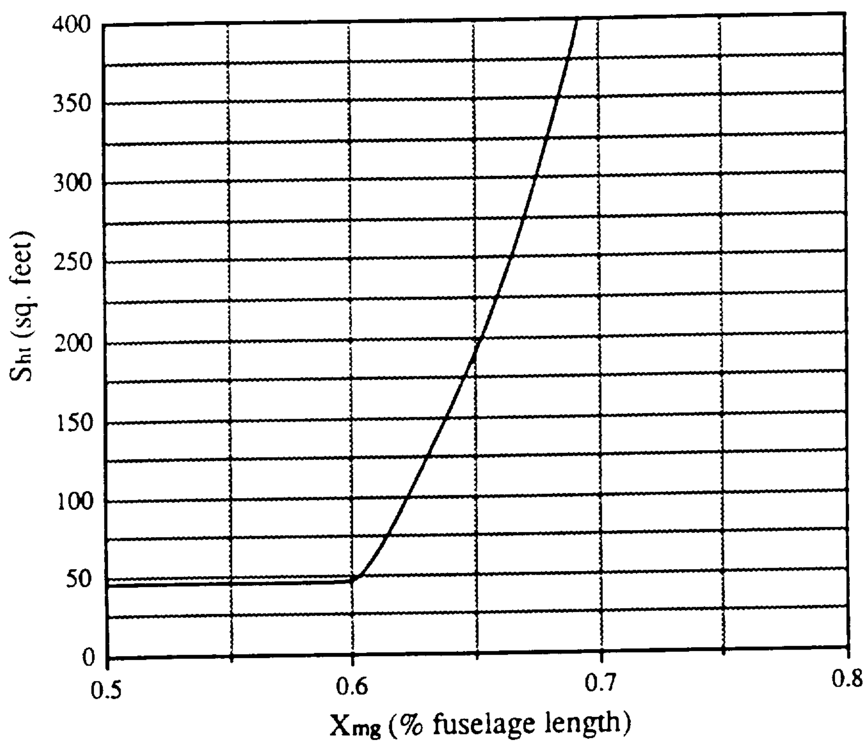

Figure 12: Horizontal Tail Size (S_{ht})
vs.
Main Landing Gear Position (X_{mg})

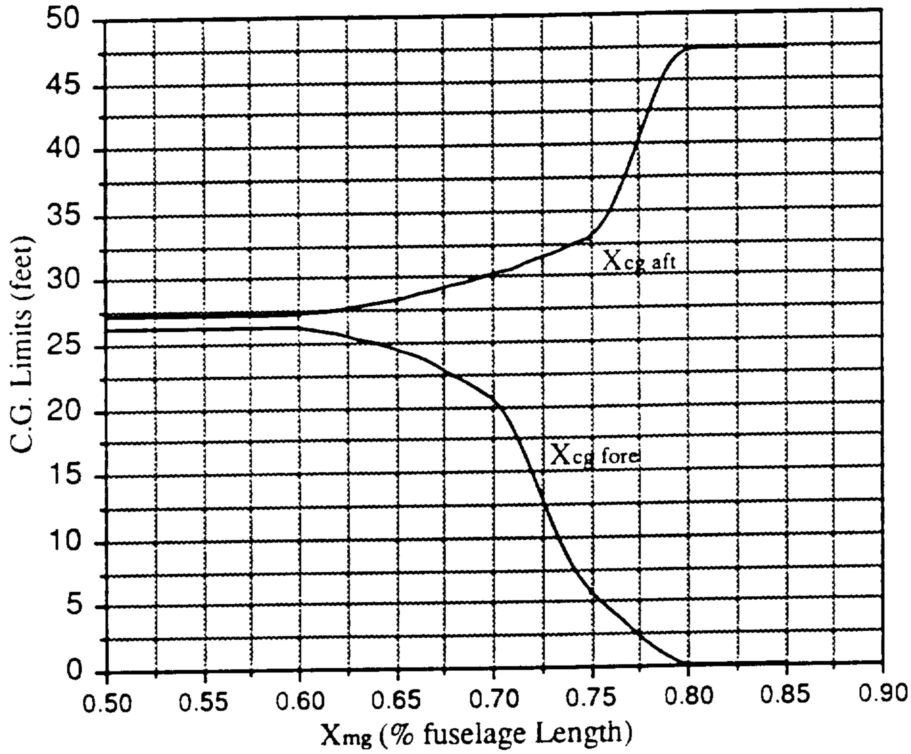

Figure 13: Forward and Aft C.G. Limits ($X_{cg\ fore}$ and $X_{cg\ aft}$)
vs.
Main Landing Gear Position (X_{mg})

CHAPTER 8

Vectored Thrust Analysis

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 (γ_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.

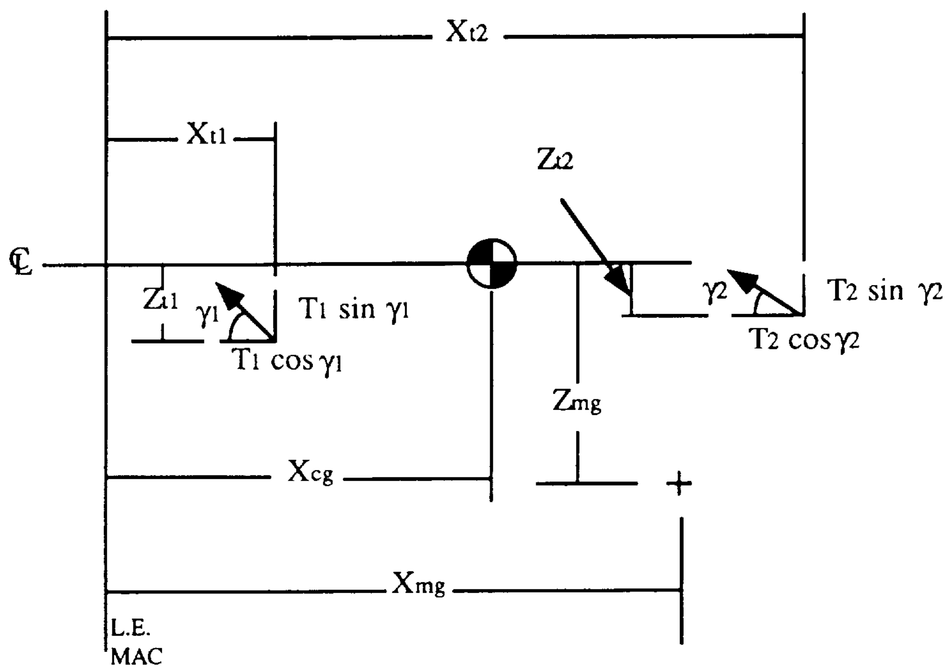

Figure 14: Thrust Vector Components and Positions

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

$$C_{M_{cg}} = C_L \frac{(X_{cg} - X_{ac})}{c} - C_{M_{wb}} - C_{L_{\alpha HT}} \alpha_{HT} \frac{S_{HT}}{S} \frac{(X_{ac_{HT}} - X_{cg})}{c} \eta_{HT} \tag{39}$$

$$+ C_{L_{\alpha CN}} \alpha_{CN} \frac{S_{CN}}{S} \frac{(X_{ac_{CN}} - X_{cg})}{c} \eta_{CN} + [X_{cg} - X_{T1}] \frac{T_1 \sin\gamma_1}{q\,S\,c}$$

$$+ Z_{T1} \frac{T_1 \cos\gamma_1}{q\,S\,c} - [X_{T2} - X_{cg}] \frac{T_2 \sin\gamma_2}{q\,S\,c} + Z_{T2} \frac{T_2 \cos\gamma_2}{q\,S\,c}$$

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 straight forward 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.

$$X_{T1} = \frac{\text{SUMVT}\ q\ S\ c}{T_1 \sin\gamma_1} + X_{cg} \tag{40}$$

where SUMVT is given by

$$\begin{aligned}\text{SUMVT} = {} & C_L \frac{(X_{cg} - X_{ac})}{c} - C_{M_{wb}} - C_{L_{\alpha HT}} \alpha_{HT} \frac{S_{HT}}{S} \frac{(X_{ac_{HT}} - X_{cg})}{c} \eta_{HT} \\ & + C_{L_{\alpha CN}} \alpha_{CN} \frac{S_{CN}}{S} \frac{(X_{ac_{CN}} - X_{cg})}{c} \eta_{CN} + Z_{T1} \frac{T_1 \cos\gamma_1}{q\ S\ c} \\ & - [X_{T2} - X_{cg}] \frac{T_2 \sin\gamma_2}{q\ S\ c} + Z_{T2} \frac{T_2 \cos\gamma_2}{q\ S\ c}\end{aligned}$$

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

$$\text{TSPLIT} = \frac{T_1}{\text{THRUST}} \tag{41}$$

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).

$$T_2 = \mathrm{THRUST} - T_1 = \mathrm{THRUST}\left(1 - \mathrm{TSPLIT}\right) \tag{42}$$

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

$$\mathrm{TSPLIT} = \frac{\dfrac{\mathrm{SUMVT}}{\mathrm{THRUST}} - X_{T2}\dfrac{\sin\gamma_2}{q\,S\,c} + Z_{T2}\dfrac{\cos\gamma_2}{q\,S\,c}}{\Delta} \tag{43}$$

where

$$\mathrm{SUMVT} = C_L\frac{\left(X_{cg} - X_{ac}\right)}{c} - C_{M_{wb}} - C_{L_{\alpha HT}}\alpha_{HT}\frac{S_{HT}}{S}\frac{\left(X_{ac_{HT}} - X_{cg}\right)}{c}\eta_{HT}$$

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

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 (γ_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 can not 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_{Mcg} with γ_2 set to 0 degrees, and with γ_2 set to 90 degrees. The two solutions are compared, and the one that has the largest magnitude for C_{Mcg} 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_{Mcg} 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 γ_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.

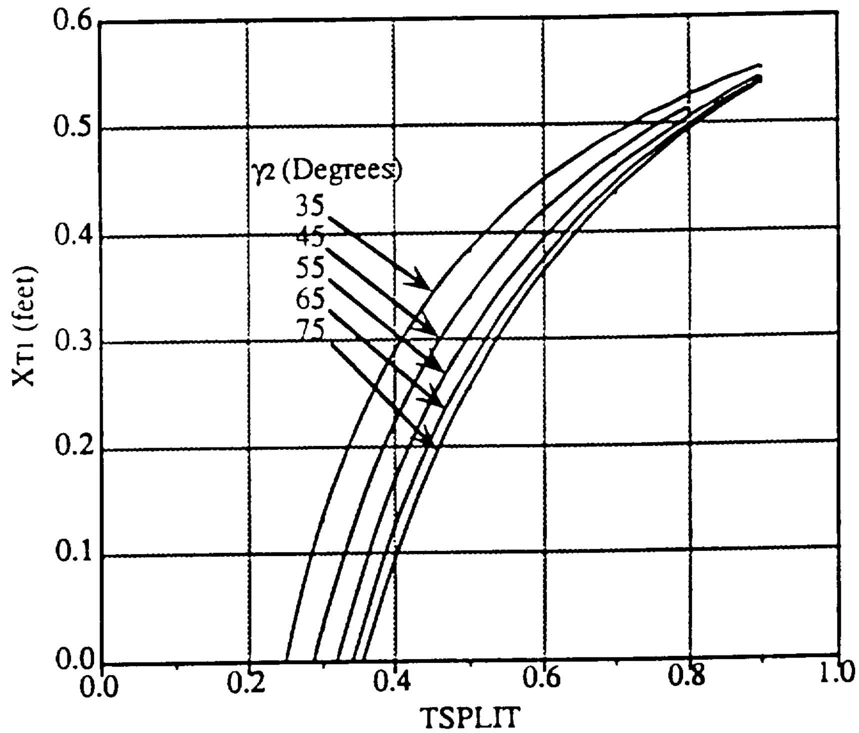

Figure 15: Forward Thrust Vector Position (X_{T1})
vs.
Thrust Split (TSPLIT) for Various
Aft Thrust Angles (γ_2)

* This graph is for an F-16A modified with
with a forward nozzle set at $\gamma = 90$ 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[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 derivitive with respect to the angle-of-

$$M_A = q \int_{-\frac{b}{2}}^{\frac{b}{2}} c^2 C_{Mac}\, dy - q \int_{-\frac{b}{2}}^{\frac{b}{2}} c\, C_l\, y \tan\Lambda\, dy \tag{44}$$

[11] McCormick, B.W., Aerodynamics, Aeronautics, and Flight Dynamics, J. Wiley and Sons, 1979. pg. 484 - 485.

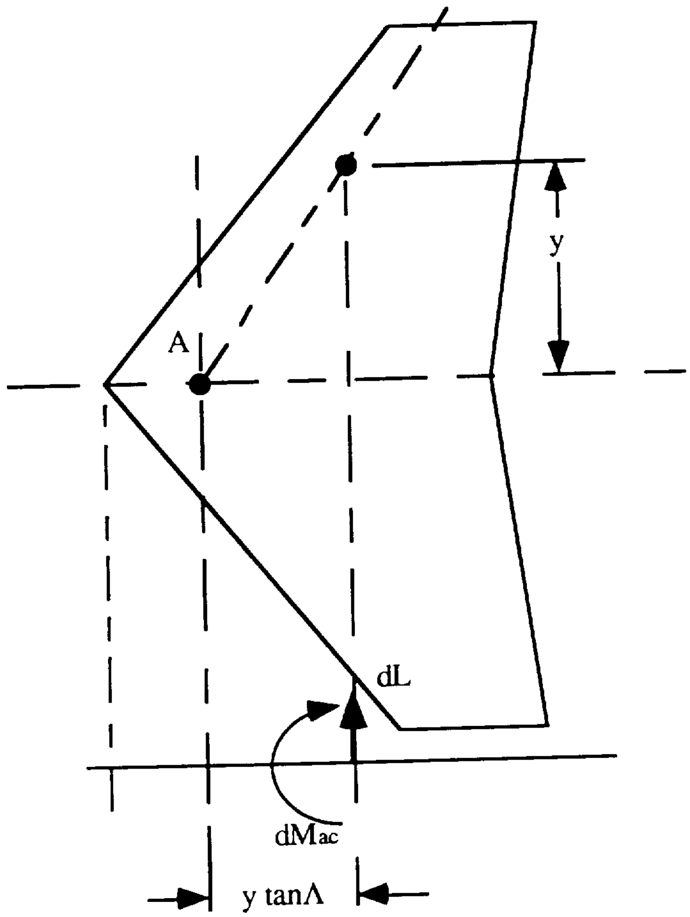

Figure 16:
Calculation of Wing Aerodynamic Center
by Moments about Root Quarter-Chord, point A.

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

$$\frac{d\,C_{M_A}}{d\,\alpha} = -\int_{\frac{-b}{2}}^{\frac{b}{2}} \frac{C_{l\,\alpha}}{S}\, y \tan\Lambda\, dy \tag{45}$$

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.

$$M_{ac} = M_A + L X_A \tag{46}$$

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.

$$\frac{d\, C_{M_A}}{d\, \alpha} = C_{L\alpha} \frac{X_A}{c} \tag{47}$$

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

$$X_A = \frac{1}{C_{L\alpha} S} \int_{-\frac{b}{2}}^{\frac{b}{2}} c\, C_{l\,\alpha}\, y \tan \Lambda\, dy \tag{48}$$

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.

$$X_{ac} = \left(\frac{1 + 2\lambda}{1 + \lambda}\right) \frac{1}{3} \left(\frac{b}{2} \tan \Lambda\right) \frac{C_{l\alpha}}{C_{L\alpha}} + \left(X_{LE_{root}} + \frac{c_{root}}{4}\right) \tag{49}$$

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 48[12].

$$C_{L\alpha} = \frac{C_{l\alpha}}{1 + \frac{57.29\, C_{l\alpha}}{e\,\pi\, AR}} \tag{50}$$

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 equation[13], Equation 51. This gives the three-dimensional lift curve

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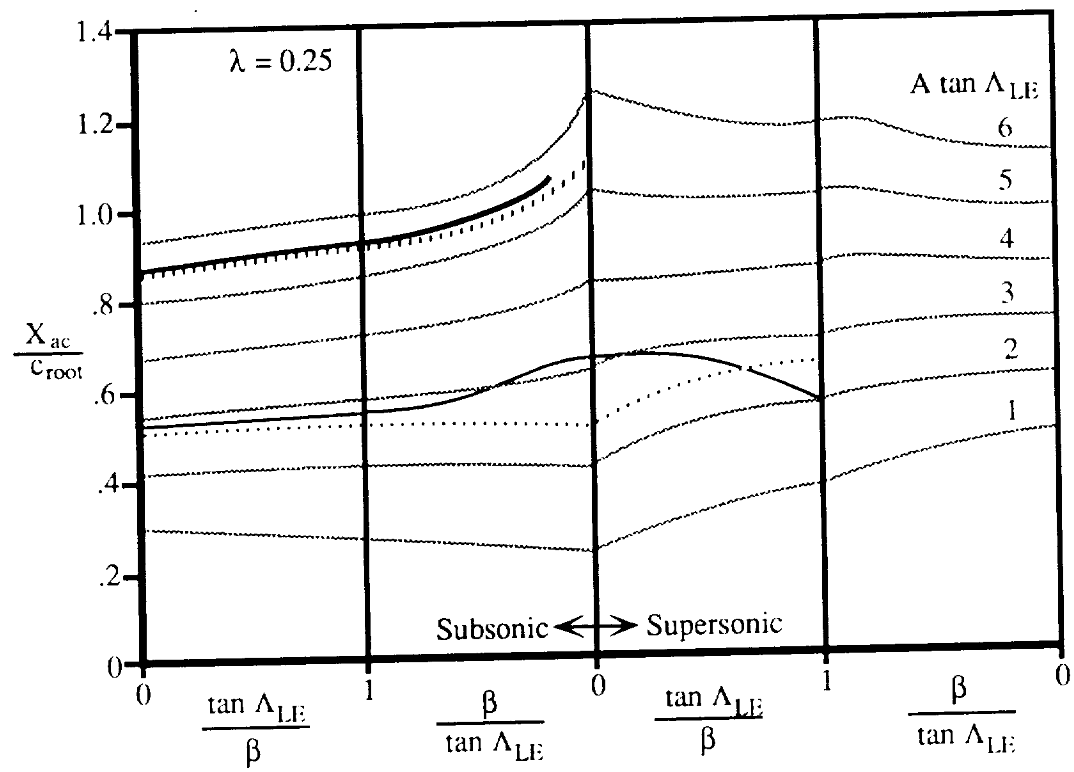
[12] Perkins, C.D. and Hage, R.E., Airplane Performance, Stability and Control, J. Wiley and Sons, 1949. pg. 220.

[13] McCormick, B.W., Aerodynamics, Aeronautics, and Flight Dynamics, J. Wiley and Sons, 1979. pg. 137.



— Numerical Results
..... Graphical Results
B727-200

— Numerical Results
..... Graphical Results
F-16A

Figure 17: Numerical and Graphical Aerodynamic Center Comparisons

(Graph Reproduced From Reference 1)

$$C_{L\alpha} = C_{l\alpha} \frac{AR}{\left(\frac{C_{l\alpha}}{\pi}\right) + \sqrt{\left(\frac{C_{l\alpha}}{\pi}\right)^2 + AR^2}} \tag{51}$$

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 Pradtl-Glauert compressibility factor. The compressibility factor depends on the subsonic or supersonic flow. The final three-dimensional lift curve slope equation is

$$C_{L\alpha} = \frac{C_{l\alpha}\, AR}{\left(\frac{C_{l\alpha}}{\pi}\right) + \sqrt{\left(\frac{AR}{\cos\Lambda}\right)^2 + \left(\frac{C_{l\alpha}}{\pi}\right)^2 - (AR\, M_\infty)^2}} \tag{52}$$

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

$$F_e = 1 - \tau \frac{d\delta}{d\alpha} = 1 - \tau \frac{b_1}{b_2} \tag{53}$$

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 of the designer.

$$b_1 = -0.55\, k_{1(ce/c)}\, k_{1(t/c)}\, k_{1(BR)}\, k_{1(1/A)} \tag{54}$$

$$b_2 = -0.89\, k_{2(ce/c)}\, k_{2(t/c)}\, k_{2(BR)}\, k_{2(1/A)} \tag{55}$$

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

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15 ibid., 495 - 508.

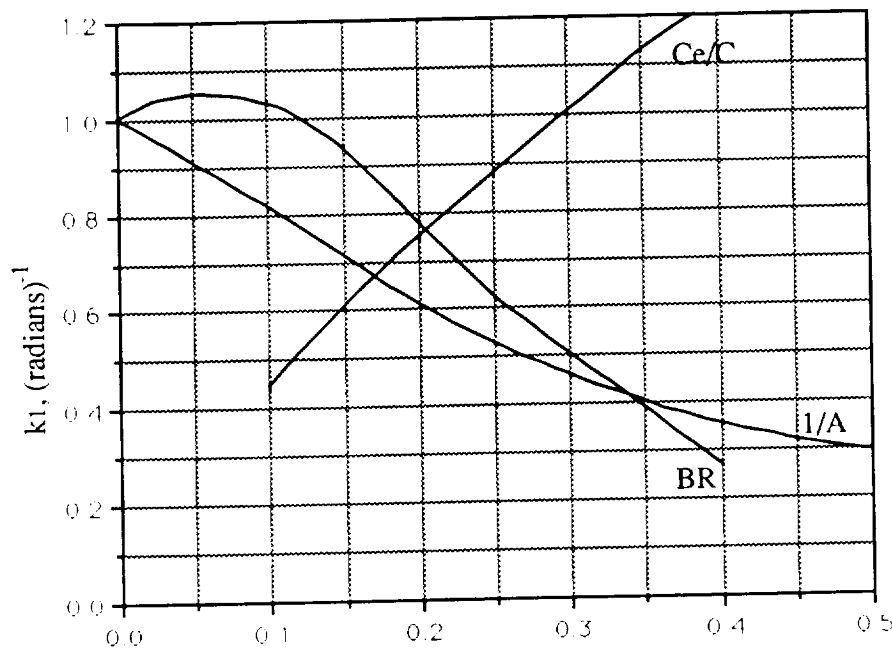

Figure 18: Curves Used in Determining the k_1 Coefficients to Calculate Stick-Free Effects. BR, 1/A, Ce/C

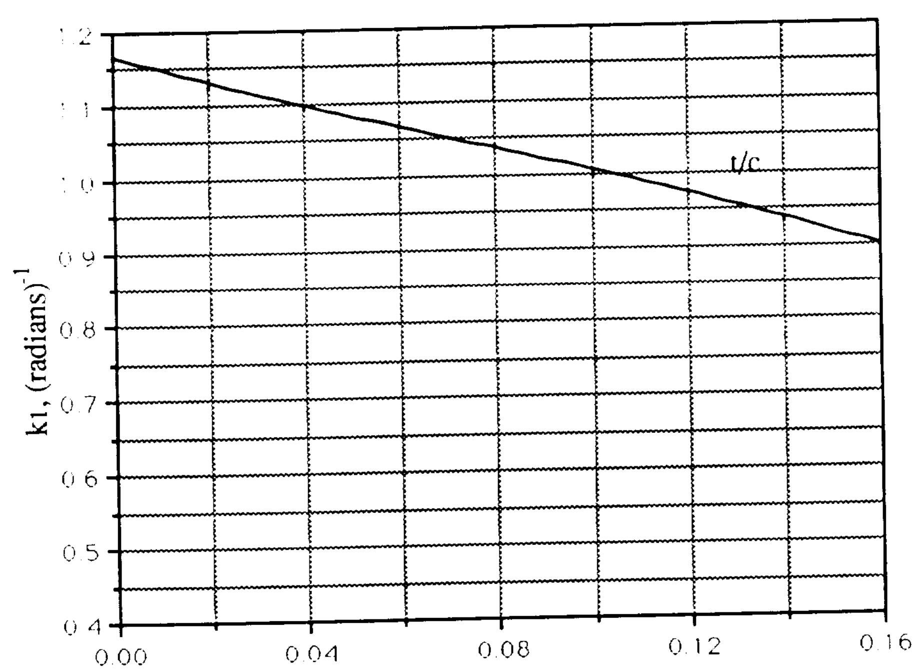

Figure 19: Curve Used in Determining the k_1 Coefficients to Calculate Stick-Free Effects. t/c

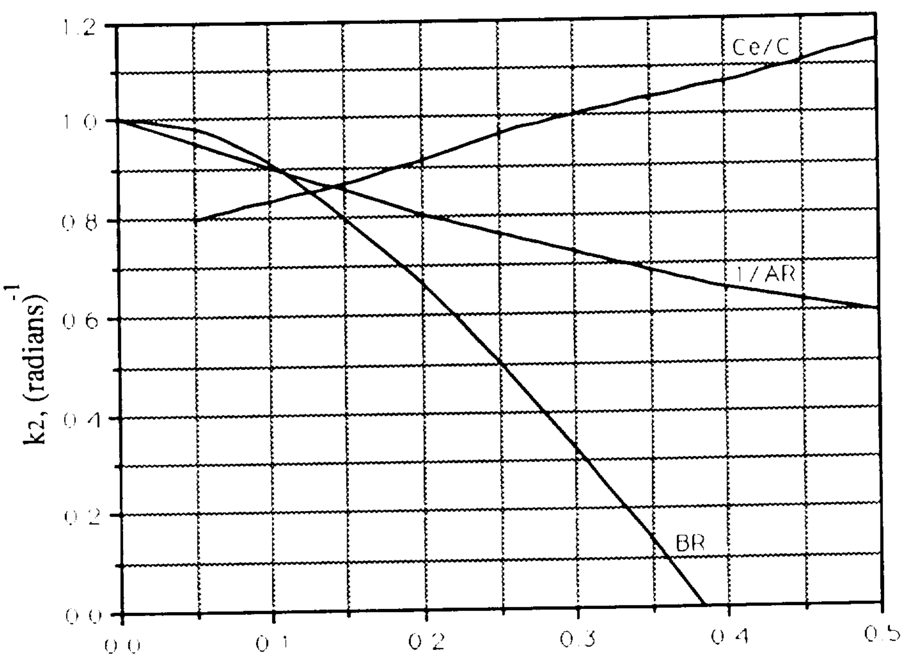

Figure 20: Curves Used in Determining the k2 Coefficients to Calculate Stick-Free Effects. BR, 1/A, Ce/C

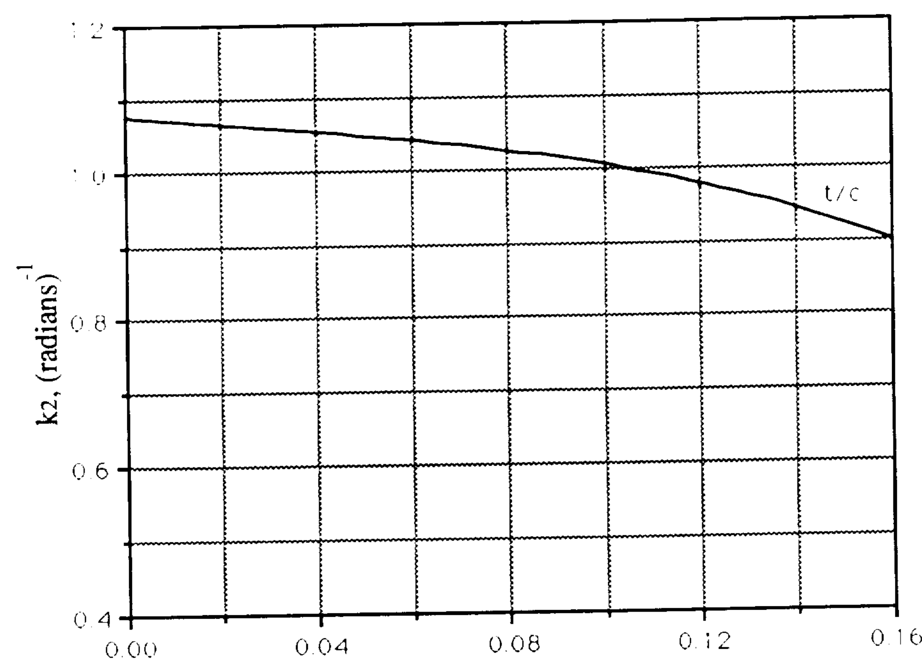

Figure 21: Curve Used in Determining the k2 Coefficient to Calculate Stick-Free Effects. t/c

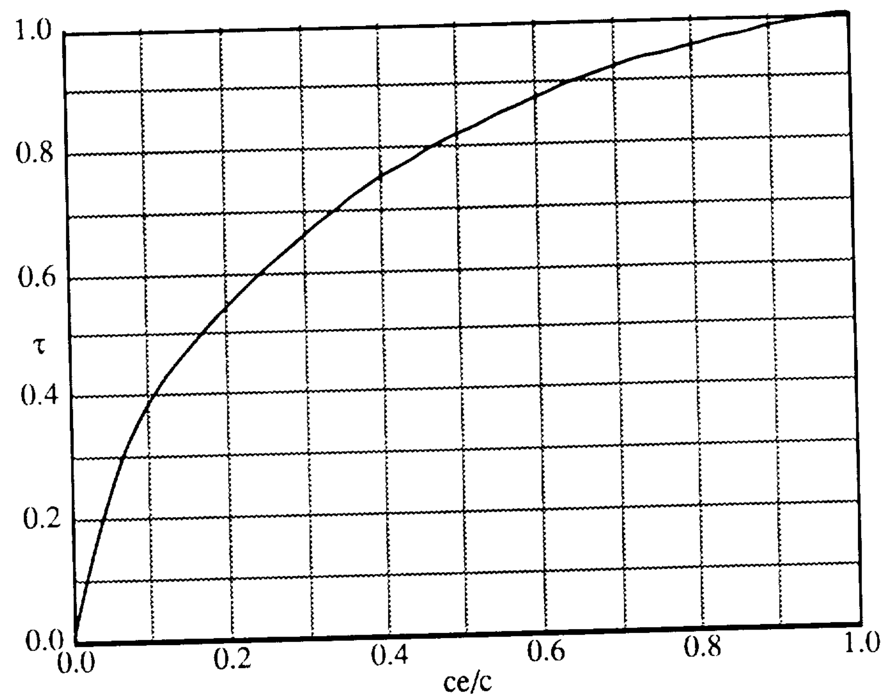

Figure 22: Control Surface Effeciency Factor (τ)
vs.
Percent Elevator Chord (c_e/c)

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.

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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 '$STABIN' and then the desired inputs. Finally, at the end of the inputs, a '$END' 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, ........
$END
```

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.

ELVDEF	-10.00	Amount of elevator deflection needed for takeoff rotation. (degrees)
ELVDMAX	-15.00	Maximum amount of elevator deflection needed for controllability at low speed. (degrees)
GAMAT1	0.00	Forward thrust nozzle rotation angle from the horizontal. (degrees)
GAMAT2	0.00	Aft thrust nozzle rotation angle from the horizontal. (degrees)
ICN	0.00	Canard incidence angle. (degrees)
IHT	0.00	Horizontal tail incidence angle. (degrees)
SCN1	0.0	Area of canard when both a canard and an aft tail are being used (feet^2).
SPANCN1	0.0	Span of the canard when both a canard and an aft tail are being used (feet).
TRCN1	0.0	Taper ratio of the canard when both a canard and an aft tail are being used.
TSPLIT	0.00	Percent thrust split, front-to-rear.
TZCHT	0.10	Elevator thickness-to-chord ratio.
VROT	0.30	Mach number at take off rotation.
WLGFRT	0.30	Weight ratio of front-to-rear landing gear for C.G. calculations.
XFAMMUN	0.30	C.G. position of the ammunition in percent BODL (fuselage length).
XFAPU	0.95	C.G. position of A.P.U. in percent BODL.
XFBB2	0.50	C.G. position of advanced weapons in percent BODL.
XFBOD	0.50	C.G. position of body in percent BODL.

XFBOMB	0.50	C.G. position of bombs in percent BODL.
XFCAN	0.20	C.G. position of the canard in percent of the Mean Aerodynamic Chord from the leading edge.
XFCARGO	0.50	C.G. position of cargo in percent BODL.
XFCREW	0.10	C.G. position of crew in percent BODL.
XFELT	0.05	C.G. position of Avionics in percent BODL.
XFENG	0.50	C.G. position of the engine in percent of PODL (engine pod length).
XFEP	0.50	C.G. position of electrical system in percent BODL.
XFEX	0.50	C.G. position of fuselage external tank in percent BODL.
XFFFUS	0.50	C.G. position of fuselage fuel in percent BODL.
XFFS	0.60	C.G. position of fuel system in percent BODL.
XFFUR	0.50	C.G. position of furnishings in percent BODL.
XFFWG	0.50	C.G. position of the fuel in the wing in percent BODL.
XFHDP	0.50	C.G. position of the hydraulics and pneumatics in percent BODL.
XFHT	0.20	C.G. position of the horizontal tail in percent of the Mean Aerodynamic Chord from the leading edge.
XFINST	0.15	C.G. position of the instruments in percent BODL.
XLECN	0.0	Leading edge position of canard when both a canard and an aft tail are used (feet).
XFLIFTF	0.50	C.G. position of the lift-fan in percent BODL.
XFLGFRT	0.10	C.G. position of the nose gear in percent BODL.
XFLGR	0.55	C.G. position of the main gear in percent BODL.
XFMISS	0.50	C.G. position of the missiles in percent BODL.
XFNA	0.98	C.G. position of the nacelles in percent PODL.

XFPA	0.55	C.G. position of passenger accommodations in percent BODL.
XFPAYL	0.50	C.G. position of any payload in percent BODL.
XFPIV	0.65	C.G. position of the control surface pivots in percent BODL.
XFSC	0.70	C.G. position of control surfaces in percent BODL.
XFVT	0.20	C.G. position of the vertical tail in percent of the Mean Aerodynamic Chord from the leading edge.
XFWG	0.15	C.G. position of the wing in percent of the Mean Aerodynamic Chord from the leading edge.
XT1	0.30	Horizontal position of the forward thrust vector in percent BODL.
XT2	0.60	Horizontal position of the aft thrust vector in percent BODL.
ZFD	0.00	Vertical distance of the aircraft centerline from the C.G. in percent BDMAX (maximum fuselage length).
ZFMG	2.00	Vertical distance of the main gear from the C.G. in percent BDMAX.
ZFT1	0.00	Vertical distance of the forward thrust vector from the C.G. in percent BDMAX.
ZFT2	0.00	Vertical distance of the aft thrust vector from the C.G. in percent BDMAX.
ZMU	0.02	Friction coefficient of the runway on the main gear.
ZRTCN1	0.0	Height of the canard quarter-chord above the aircraft centerline (feet).

---Integer Format----------------------

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

APPENDIX B

Sample Output

AIRCRAFT COMPONENT WEIGHTS AND POSITIONS

COMPONENT	WIEGHT	POSITION
AIRFRAME	17022.544922	42.939999
WING	17129.294922	64.515587
CANARD	0.000000	0.000000
HT	1581.655640	114.010048
VT	1310.330078	107.414925
NOSE GEAR	1223.618774	11.300000
MAIN GEAR	2855.110352	70.512001
NACELLS	1503.169800	45.200001
PIVOTS	0.001953	45.200001
AIR COND.	309.482300	44.365093
APU	605.342590	70.059998
AVIONICS	1441.599487	1.130000
ELECTRICAL	1902.595215	22.600000
INSTURMENTS	796.524597	5.650000
HYDRAULICS	563.688660	56.500000
CONTROL SURF	2981.909424	56.500000
PA	0.000000	45.200001
FURNISHINGS	450.000000	36.160000
ENGINES	6954.871582	91.914490
FUEL SYS.	1682.630249	33.900002
LIFT FAN	0.000610	56.500000
PAYLOAD	60.000000	22.600000
CREW	510.000000	11.300000
CARGO	0.000000	33.900002
AMMUNITION	0.000000	33.900002
BB2	0.000000	56.500000
BOMBS	0.000000	56.500000
MISSILES	0.000000	53.110001
FUS. FUEL	0.000	56.500000
WING FUEL	16347.398	64.235992

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 DCMDCLW = -0.00342405
DCMDCLF = 0.00338223 DCMDCLCN = 0.00000000
DCMDCLJ = 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 =	0.4101450
XCG =	64.0528641 ft.
CGBAR =	33.6899948 % chord
DCMDCL =	0.0047093
STATIC MARGIN =	7.3245034 % chord
MISSION PHASE NUMBER	1
XAC =	64.0994263 ft.
NEUTPT =	0.4538571
XCG =	64.0515594 ft.
CGBAR =	33.6806602 % chord
DCMDCL =	-0.0539999
STATIC MARGIN =	7.3338361 % chord
MISSION PHASE NUMBER	2
XAC =	64.4846954 ft.
NEUTPT =	0.4740317
XCG =	64.0514832 ft.
CGBAR =	33.6801147 % chord
DCMDCL =	-0.0784065
STATIC MARGIN =	7.3343816 % chord
MISSION PHASE NUMBER	3
XAC =	64.1274567 ft.
NEUTPT =	0.4552816
XCG =	64.0369263 ft.
CGBAR =	33.5759850 % chord
DCMDCL =	-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
DCMDCL =	-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
DCMDCL =	-0.0627594
STATIC MARGIN =	7.5302525 % chord

END STABILITY OUTPUT

B727-200 Final Stability Output

APPENDIX C

Production Aircraft Layouts

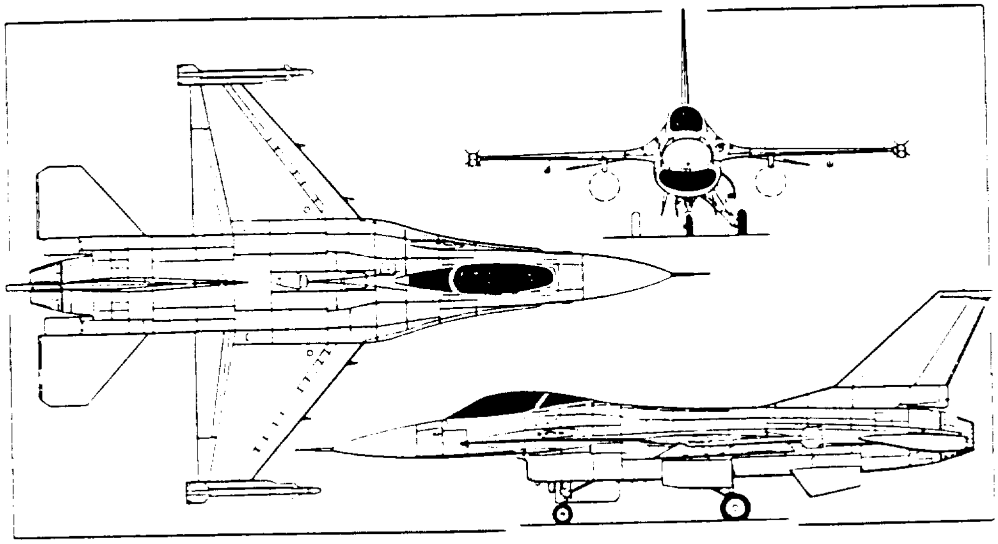

Figure 23: F-16A Geometric Layout

(Reproduced from Reference 9)

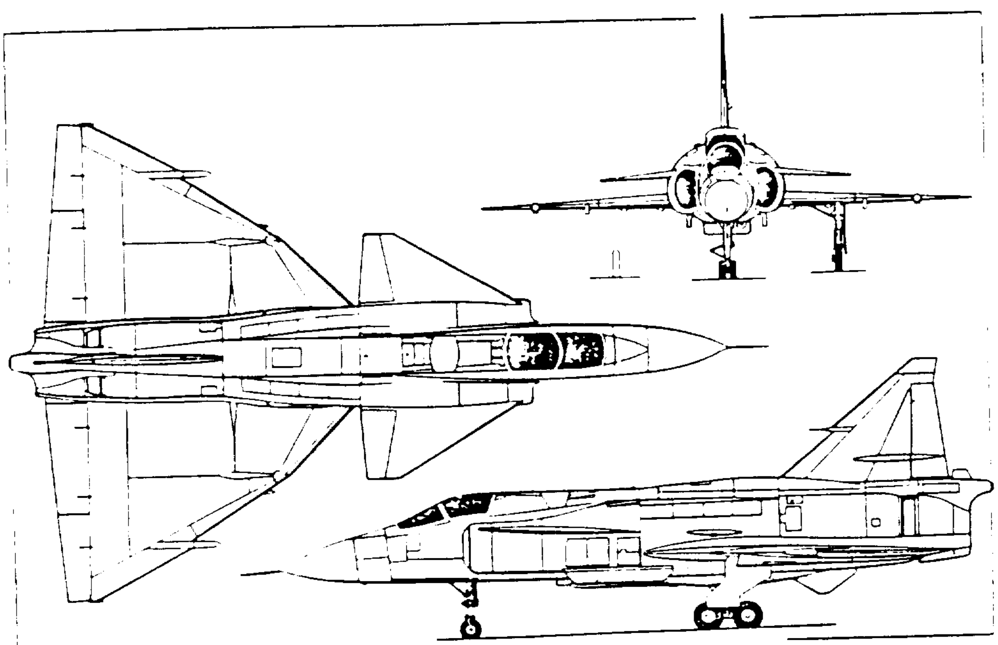

Figure 24: Saab J-37 Viggen Geometric Layout

(Reproduced from Reference 9)

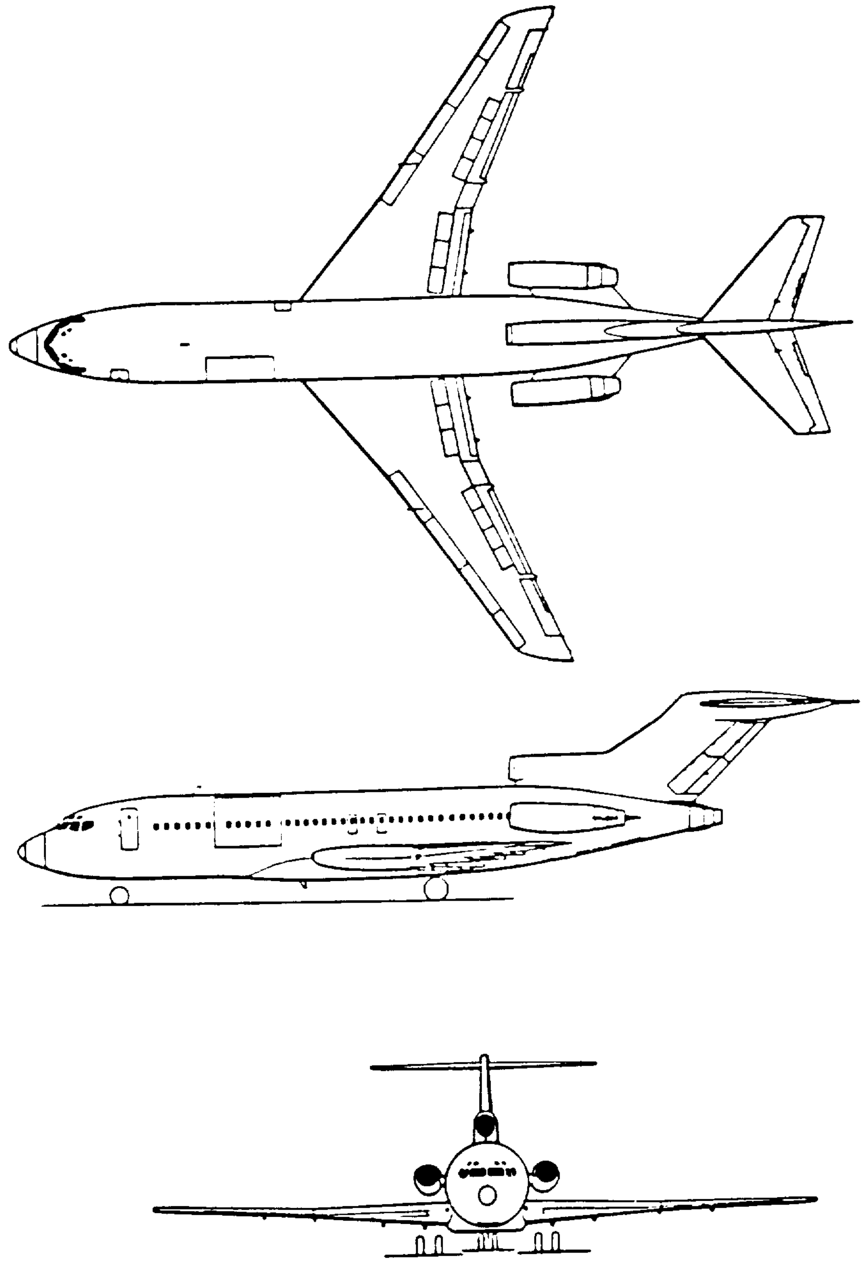

Figure 25: B727-200 Geometric Layout

(Reproduced from Reference 2)

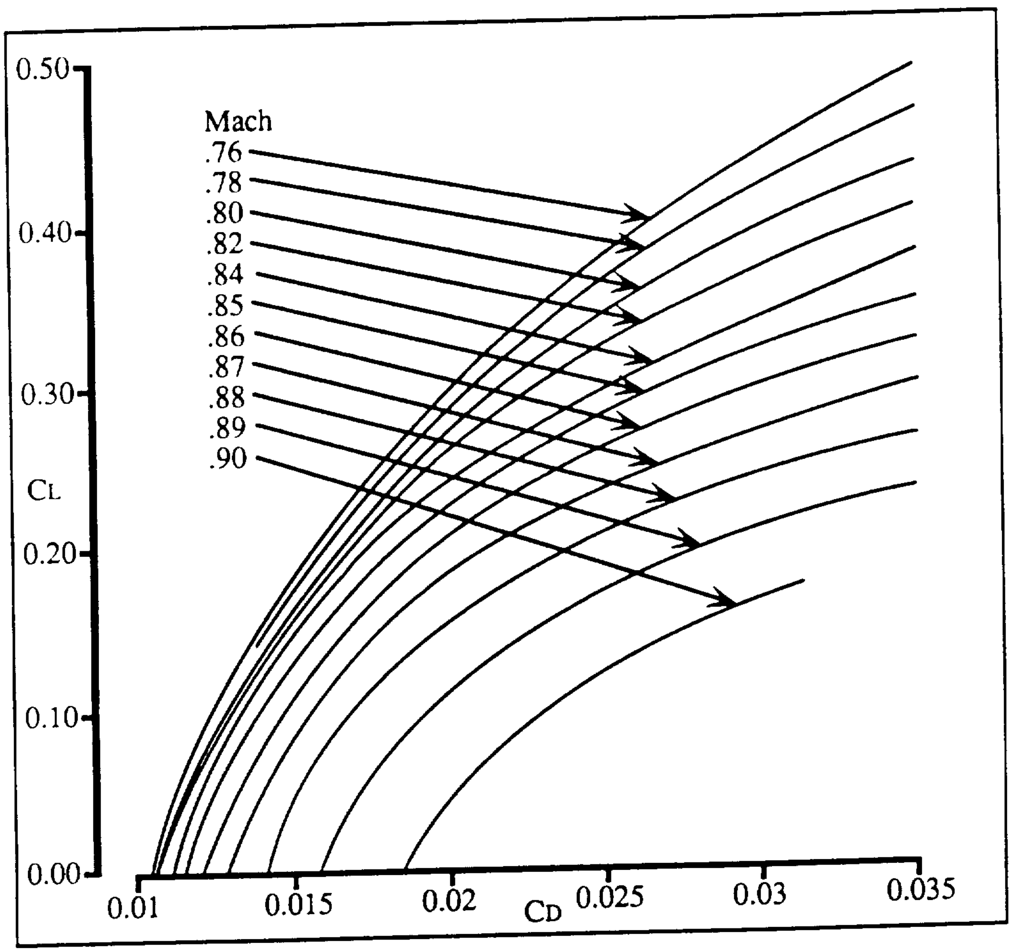

Figure 26: B727-200 Aerodynamic Data

(Reproduced from Reference 7)

APPENDIX D

Program Listing

```
      SUBROUTINE STBLTY
C
C  THIS SUBROUTINE IS USED IN CONJUCTION WITH NASA AMES AIRCRAFT DESIGN
C   PROGRAM 'ACSYNT'.  THIS PROGRAM CALCULATES THE SIZE OF THE HORIZONTAL
C   CONTROL SURFACE FOR EITHER AN AFT TAIL OR A FORWARD CANARD.  IT CAN
C   ALSO HANDLE AN AIRCRAFT WITH BOTH CONTROLS, BUT IT ONLY SIZES THE AFT
C   TAIL, THE SIZE OF THE CANARD IS AN INPUT FOR THE USER.
C  THIS SUBROUTINE ALSO CALCULATES THE CENTER OF GRAVITY OF THE AIRCRAFT,
C   THE CENTER OF GRAVITY SHIFT DURING THE TRAJECTORY MISSION, AND THE
C   FORWARD AND AFT CENTER OF GRAVITY LIMITS IMPOSED BY THE SIZE OF THE
C   HORIZONTAL CONTROL SURFACE.
C  IT ALSO DETERMINES THE PITCHING MOMENT CURVE SLOPE (DCM/DCL) OF THE
C   AIRCRAFT FOR EACH PHASE IN THE MISSION.
C  THE REFERENCES SITED THROUGHOUT THE PROGRAM ARE LISTED AT THE END
C   OF THE LISTING.
C                                             STEPHEN SWANSON  4/89
C
C  ****************************************************************************
C  *   THE FILE 'STBCM.INC' MUST BE IN THE SAME DIRECTORY AS THIS MODULE      *
C  *    FOR IT TO WORK.  THE FILE CONTAINS ALL THE COMMON BLOCK STATEMENTS    *
C  *    NEEDED TO RUN THIS MODULE.  THE 'INCLUDE' STATEMENT IS USED TO        *
C  *    INCLUDE THE FILE IN WITH THE MODULE DURING COMPILATION.   S.M.S. 1/89 *
C  *                                                                          *
C  *   THE FOLLOWING IS A LISTING OF THE STBCM.INC FILE......                 *
C  *    *     *     *     *     *     *     *     *     *     *     *         *
C  * THIS CONTAINS THE COMMON BLOCK STATEMENTS NEEDED TO RUN THE STABILITY    *
C  *  AND CONTROL MODULE 'STBLCON.FOR'.  THIS FILE IS CONNECTED WITH THE      *
C  *  MODULE THROUGH THE 'INCLUDE' STATEMENT FOUND AT THE BEGINING OF EACH    *
C  *  OF THE SUBROUTINES.                                                     *
C  *                                                                          *
C  *    COMMON/OVER/ICALC,NERROR,MODNO,IGEO,KGPRNT,IGPLT,IF                   *
C  *                                                                          *
C  *    COMMON/STBCM/CLALFA,CLAFHT,DEDA,XQCBCN,XQCBWG,XQCBHT,XQCBVT,          *
C  *  1  CBARCN,CBARWG,CBARHT,CBARVT,ROOTWG,TRWG,XLEWG,SPANWG,                *
C  *  2  SWPWG,PODL,EN,BODL,XLEPOD(10),WCAND,WFTOT,FUFRAC,WWING,WFS,         *
C  *  3  WLG,WHT,WVT,WETANK,WFEXT,WBODY,WAIRC,WAPU,WSC,WENG,SCN,SWG,SHT,     *
C  *  4  SVT,WCREW,WELT,WEP,WHDP,WINST,WNA,WPA,WCARGO,WAMMUN,WBOMB,WMISS,    *
C  *  5  ARWG,DIHED,ZRTWG,ZRTHT,ARVT,SWPCN,TRCN,AMTO,                        *
C  *  6  ZRTCN,VOLB,BDMAX,FRN,FRAB,ALPHA,CL,THRUST,ARHT,TNT(12),WFT(12),     *
C  *  7  WFTO,WTOT,ARCN,CMWB,WGTO,WFEQ,WBB2,WAF,WE,WPL,WPS,WTSUM,WFUEL,      *
C  *  8  TWTO,MACH,ALT,SWETWG,SWETHT,SWETCN,SPANHT,SPANCN,                   *
C  *  9  STARTM(12),DUMMY,IAO                                                *
C  *                                                                          *
C  *    COMMON/LOCALS/ETAHT,XFCAN,XFHT,XFNA,XFVT,XFWG,XFAPU,XFENG,            *
C  *  1  XFELT,XFEP,XFINST,XFHDP,XFPA,XFPAYL,XFFFUS,XFCREW,WPAYL,XFFUR,      *
C  *  2  XFLGFRT,WFLGFRT,XFLGR,XFFS,XFBOMB,XFSC,XFPIV,XFLIFTF,WPIV,WFUR,     *
C  *  2  XEF1,XFAMMUN,XFBB2,XFMISS,XFCARGO,XACWG,ETACN,ODEDA,XAWB,CLACN,     *
C  *  3  WZET,WZEF,WEFF,WEFW,XEF2,WEFWT,WZENG,SUMMO,SUMWT,XLE,LWB,LTTWB,     *
C  *  3  IDBPRT,XCG(13),DCMDCL(13),SM(13),XCGAFT(13),WLGFRT,WLGR,WFWG,       *
C  *  4  WFP,WZB,WZM,WZA,IBS,IMS,IAS,NPHASE,ICGPRT,WLFTF,WFFUS,LHT,LCN,      *
C  *  5  CELV,CBALHT,TZCHT,FNEUTPT(13),XFBOD,XFFWG,XFEX,XLETIP,IETANK,       *
C  *  6  ZMTOR,AWTOT,ZMU,XT1,XT2,ZFMG,ZMG,ZFT1,ZT1,ZFT2,ZT2,ZFD,ZD,T1,       *
C  *  7  TSPLIT,GAMAT1,GAMAT2,IHT,ICN,ELVDEF,VROT,ALPHACN,TAU,T2,            *
C  *  8  SHT1,SCN1,ELVDMAX,IWG,XCGFRWD(13),QTOR,CGBAR(13),ALFTO,CLTO,        *
```

```
C  *  8  CDTO,CMWBTO,ALFMAX,CDMAX,CMWBMAX,ALFO,CLO,CDO,CMWBO,XLECN,           *
C  *  9  T1RAD,T2RAD,XLECG,XLET1,XLET2,CLMAX,CLAHTGRD,CLACNGRD,IVECT,        *
C  *  1  SPANCN1,ZRTCN1,ARCN1,TRCN1,ALFHT,ALFCN,DCMTMUL,TWZTO                *
C  *                                                                         *
C  *    COMMON/XCGS/XBOD,XWING,XCAN,XHT,XVT,XNA,XPIV,XAIRC,XAPU,            *
C  *  1  XELT,XEP,XINST,XHDP,XSC,XPA,XENG(10),XFS,XLIFTF,XFUR,              *
C  *  2 XPAYL,XCREW,XCARGO,XAMMUN,XBB2,XBOMB,XMISS,XETANK,                  *
C  *  3  XFUSF,XWGFUEL,XLGFRT,XLGR,LFLAG                                    *
C  *                                                                         *
C  *     REAL XACWG,DCMDCLF,DCMDCLT,DCMDCLW,DCMCLCN,KF,CSC,TC,MBOD,          *
C  *  1  MCAN,MWG,MFS,MLG,MHT,MVT,METANK,MFEXT,MAIRC,MAPU,MSC,MENG(10),     *
C  *  1  LWB,LH1,LH2,LH3,LH4,LH5,LH6,LH7,LH8,LH9,LC1,LC2,LC3,LC4,LC5,       *
C  *  2  LC6,LC7,LC8,LHT,LCN,ZMTOR,ZMU,LTTWB,IWG,IHT,ICN,LEFF,MACH,         *
C  *  2 MCREW,MELT,MEP,MHDP,MINST,MNA,MPA,MCARGO,MAMMUN,MBOMB,MMISS,        *
C  *  3  MPAYL,MPIV,MLIFTF,K11,K12,K13,K14,K21,K22,K23,K24                  *
C  ****************************************************************************
C
      INCLUDE 'STBCM.INC'
C
        IF (ICALC.GT.1.) GO TO 10
C     READ IN THE INPUT PARAMETERS.
        CALL STABINPT
C
 10     CONTINUE
C
        IF (ICALC.NE.2.) GO TO 20
C     DO ALL THE NECESSARY CALCULATIONS.
        CALL STABCALC1
C
 20     CONTINUE
C
        IF (ICALC.NE.3.) GO TO 30
C     DO A FINAL PRINTOUT OF THE DATA.
        CALL STABOUT
C
 30     CONTINUE
C
        RETURN
        END
C
C
C   ***********************************************************************
C
        SUBROUTINE STABINPT
C
C   THIS SUBROUTINE READS IN THE INPUT FROM THE INPUT FILE CREATED BY
C THE USER IN NAMELIST 'STABIN'   IT ALSO INITILIZES ALL VALUES SO
C THE USER DOES NOT NEED TO SPECIFIY ALL VALUES, ONLY THE ONES DESIRED.
C
        INCLUDE 'STBCM.INC'
C
        NAMELIST/STABIN/ETAHT,XFCAN,XFHT,XFNA,XFVT,XFWG,XFAPU,XFBB2,
     1  XFPIV,XFELT,XFEP,XFINST,XFHDP,XFPA,XFFFUS,XFBOD,XFFWG,XFCARGO,
     2  XFLGFRT,WFLGFRT,XFLGR,XFFUR,XFMISS,XFAMMUN,XFPAYL,XFEX,XFFS,
```

```
     2  XFSC,XFBOMB,XFENG,XFLIFTF,XFCREW,IDBPRT,CELV,CBALHT,TZCHT,
     3  ETACN,AWTOT,ZMU,XT1,XT2,ZFMG,ZFT1,ZFT2,ZFD,TSPLIT,GAMAT1,
     4  GAMAT2,IHT,ICN,ELVDEF,ELVDMAX,VROT,SCN1,XLECN,
     5  SPANCN1,ZRTCN1,TRCN1,IETANK,IBS,IMS,IAS,NPHASE,ICGPRT,IVECT
C
C ***************************************************************************
C     THE FOLLOWING IS A LIST OF THE INPUTS AND WHAT THEY STAND FOR...
C       ETAHT   - AFT HORIZONTAL TAIL EFFICIENCY
C       ETACN   - FORWARD CANCARD EFFICIENCY
C       XFBOD   - FUSSELAGE BODY C.G. POSITION IN PERCENT BODL
C       XFCAN   - FORWARD CANARD C.G. POSITION IN PERCENT MAC
C       XFHT    - AFT HORIZONTAL TAIL C.G. POSITION IN PERCENT MAC
C       XFNA    - NACELL(S) C.G. POSITION IN PERCENT BODL
C       XFVT    - VERTICAL TAIL C.G. POSITION IN PERCENT MAC
C       XFWG    - WING C.G. POSITION IN PERCENT MAC
C       XFLGFRT - NOSE GEAR C.G. POSITION IN PERCENT BODL
C       WFLGFRT - PERCENT NOSE GEAR OF TOTAL LANDING GEAR WEIGHT
C       XFLGR   - MAIN LANDING GEAR C.G. POSITION IN PERCENT BODL
C       XFPIV   - CONTROL PIVOTS C.G. POSITION IN PERCENT BODL
C       XFFUR   - FURNISHINGS C.G. POSITION IN PERCENT BODL
C       XFAPU   - APU C.G. POSITION IN PERCENT BODL
C       XFELT   - AVIONICS C.G. POSITION IN PERCENT BODL
C       XFEP    - ELECTRICAL SYSTEMS C.G. POSITION IN PERCENT BODL
C       XFINST  - INSTRUMENTS C.G. POSITION IN PERCENT BODL
C       XFFS    - FUEL SYSTEM C.G. POSITION IN PERCENT BODL
C       XFHDP   - HYDRAULIC AND PNEUMATIC C.G. POSITIONS IN PERCENT BODL
C       XFSC    - SURFACE CONTROLS C.G. POSITION IN PERCENT BODL
C       XFPA    - PASSENGER ACCOMADATIONS C.G. POSITION IN PERCENT BODL
C       XFPAYL  - PAYLOAD C.G. POSITION IN PERCENT BODL
C       XFCARGO - CARGO C.G. POSITION IN PERCENT BODL
C       XFFWG   - FUEL IN WING C.G. POSITION IN PERCENT BODL
C       XFFFUS  - FUEL IN FUSELAGE C.G. POSITION IN PERCENT BODL
C       XFEX    - EXTERNAL TANKS C.G. POSITION IN PERCENT BODL
C       XFCREW  - CREW C.G. POSITION IN PERCENT BODL
C       XFENG   - ENGINE(S) C.G. POSITION IN PERCENT PODLength
C       XFLIFTF - LIFT FAN C.G. POSITION IN PERCENT BODL
C       XFBOMB  - BOMBS C.G. POSITION IN PERCENT BODL
C       XFBB2   - ADVANCED WEAPONS SYSTEMS C.G. POSITION IN PERCENT BODL
C       XFMISS  - MISSILES C.G. POSITION IN PERCENT BODL
C       XFAMMUN - AMMUNITION C.G. POSITION IN PERCENT BODL
C       IDBPRT  - DEBUG PRINT FLAG, CREATES MORE OUTPUT
C       CELV    - PERCENT OF STABILIZER WHICH IS MOVABLE ELEVATOR
C       CBALHT  - PERCENT OF ELEVATOR WHICH IS FORWARD OF THE HINGE LINE
C       TZCHT   - THICKNESS TO CHORD RATIO OF THE ELEVATOR
C       AWTOT   - ANGLE OF ATTACK OF THE WING AT TAKE OFF ROTATION
C       ZMU     - FRICTION COEFFICIENT OF RUNWAY ON TIRES
C       XT1     - HORIZONTAL POSITION OF FORWARD THRUST VECTOR PERCENT BODL
C       XT2     - HORIZONTAL POSITION OF AFT THRUST VECTOR IN PERCENT BODL
C       ZFMG    - VERT. DIST. FROM FUSS. CENTERLINE TO MAIN GEAR, % BDMAX
C       ZFT1    - VERT. DIST. FROM FUSS. CENTERLINE TO FORE THRUST, %BDMAX
C       ZFT2    - VERT. DIST. FROM FUSS. CENTERLINE TO AFT THRUST, %BDMAX
C       ZFD     - VERT. DIST. FROM FUSS. CENTERLINE TO DRAG VECTOR,%BDMAX
C       TSPLIT  - FORWARD THRUST TO TOTAL THRUST RATIO IN PERCENT
C       GAMAT1  - FORWARD THRUST VECTOR ANGLE, DEGREES
```

```
C       GAMAT2  - AFT THRUST VECTOR ANGLE, DEGREES
C       SCN1    - 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       ZRTCN1  - VERTICAL CANARD HEIGHT USED WITH AFT TAIL, FEET
C       TRCN1   - TAPER RATIO OF CANARD USED WITH AFT TAIL
C       IHT     - AFT TAIL INCEDENCE ANGLE, DEGREES
C       ICN     - FORWARD CANARD INCEDENCE 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. O=TANK ON FUSS. CENTERLINE,1=TWO
C                 FUSS. SIDE MOUNTED TANKS, 2=TWO MAC MOUNTED TANKS,
C                 3=TWO WING-TIP TANKS, 4=ONE CENTERLINE TANK AND TWO
C                 MAC MOUNTED TANKS, 5=CENTERLINE TANK AND WING-TIP TANKS,
C                 6=TWO MAC TANKS AND TWO WING-TIP TANKS.
C       IBS     - INDICATOR TO REMOVE BOMB WEIGHT ON A SPECIFIED MISSON PHASE
C       IMS     - INDICATOR TO REMOVE MISSILE 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 VECTOR
C                 POSTION, 2=SOLVE FOR THRUST SPLIT, 3=SOLVE FOR AFT
C                 THRUST VECTOR ANGLE.
C   ********************************************************************************
C
C       INITIALLZE THE STABIN INPUT VALUES
C
        ETAHT = 0.90
        ETACN = 0.90
        XFBOD = .50
        XFCAN = .20
        XFHT = .20
        XFNA = .40
        XFVT = .20
        XFWG = .15
        XFLGFRT = .10
        WFLGFRT = .3
        XFLGR = .55
        XFPIV = .65
        XFFUR = .50
        XFAPU = .95
        XFELT = .05
        XFEP = .50
        XFINST = .15
        XFFS = .60
        XFHDP = .50
        XFSC = .70
        XFPA = .55
        XFPAYL = .50
        XFCARGO = .50
        XFFWG = .50
```

```
      XFFFUS = .50
      XFEX = .50
      XFCREW = .20
      XFENG = .50
      XFLIFTF = .50
      XFBOMB = .50
      XFBB2 = .50
      XFMISS = .47
      XFAMMUN = .30
      IDBPRT = 0
      CELV = 0.30
      CBALHT = 0.0
      TZCHT = 0.10
      AWTOT = 2.0
      ZMU = .02
      XT1 = .30
      XT2 = .60
      ZFMG = 1.0
      ZFT1 = 0.0
      ZFT2 = 0.0
      ZFD = 0.0
      TSPLIT = 0.0
      GAMAT1 = 0.0
      GAMAT2 = 0.0
      SCN1 = 0.0
      XLECN = 0.0
      SPANCN1 = 0.0
      ZRTCN1 = 0.0
      TRCN1 = 0.0
      IHT = 0.0
      ICN = 0.0
      ELVDEF = -10.0
      ELVDMAX = -15.0
      VROT = 0.3
      IETANK = 0
      IBS = 0
      IMS = 0
      IAS = 0
      NPHASE = 1
      ICGPRT = 0
      IVECT = 0
C
C     READ IN THE VALUES FROM THE NAMELIST
C
      IF (ICALC .GT. 1.) GO TO 290
      READ(5,40) TITLE
      READ(5,STABIN)
C
C   OUTPUT THE INPUTS TO THE OUTPUT FILE SO THE USER CAN DOUBLE
C   CHECK THAT THE VALUES INPUT ARE THOSE DESIRED.
C
     WRITE(6,300)
     WRITE(6,310)ETAHT,XFBOD,XFCAN
     WRITE(6,315)ETACN,WFLGFRT
```

```
      WRITE(6,320)XFHT,XFNA,XFVT
      WRITE(6,330)XFWG,XFLGFRT,XFLGR
      WRITE(6,340)XFPIV,XFFUR,XFAPU
      WRITE(6,350)XFELT,XFEP,XFINST
      WRITE(6,360)XFFS,XFHDP,XFSC
      WRITE(6,370)XFPA,XFPAYL,XFFWG
      WRITE(6,380)XFFFUS,XFEX,XFCREW
      WRITE(6,390)XFENG,XFLIFTF
      WRITE(6,400)XFAMMUN,XFBOMB,XFMISS
      WRITE(6,405)XFCARGO,XFBB2
      WRITE(6,410)CELV,CBALHT,TZCHT
      WRITE(6,411)AWTOT,ZMU
      WRITE(6,412)XT1,XT2,ZFMG
      WRITE(6,413)ZFT1,ZFT2,ZFD
      WRITE(6,414)TSPLIT,GAMAT1,GAMAT2
      WRITE(6,415)IHT,ICN,ELVDEF
      WRITE(6,416)VROT,SPANCN1,ZRTCN1
      WRITE(6,417)SCN1,XLECN,ELVDMAX
      WRITE(6,418)TRCN1,IVECT
      WRITE(6,420)IETANK,IBS,IMS,IAS
      WRITE(6,430)NPHASE,IDBPRT,ICGPRT
 290    CONTINUE
C
  40  FORMAT(20A4)
 300  FORMAT(//,1H1,15X,28HSTABILITY AND CONTROL INPUTS,//)
 310  FORMAT(10X,7HETAHT= ,F10.5,5X,8HXFBOD=  ,F10.5,5X,7HXFCAN= ,F10.5)
 315  FORMAT(10X,7HETACN= ,F10.5,5X,8HWFLGFRT=,F10.5)
 320  FORMAT(10X,7HXFHT=  ,F10.5,5X,8HXFNA=   ,F10.5,5X,7HXFVT=  ,F10.5)
 330  FORMAT(10X,7HXFWG=  ,F10.5,5X,8HXFLGFRT=,F10.5,5X,7HXFLGR= ,F10.5)
 340  FORMAT(10X,7HXFPIV= ,F10.5,5X,8HXFFUR=  ,F10.5,5X,7HXFAPU= ,F10.5)
 350  FORMAT(10X,7HXFELT= ,F10.5,5X,8HXFEP=   ,F10.5,5X,7HXFINST=,F10.5)
 360  FORMAT(10X,7HXFFS=  ,F10.5,5X,8HXFHDP=  ,F10.5,5X,7HXFSC=  ,F10.5)
 370  FORMAT(10X,7HXFPA=  ,F10.5,5X,8HXFPAYL= ,F10.5,5X,7HXFFWG= ,F10.5)
 380  FORMAT(10X,7HXFFFUS=,F10.5,5X,8HXFEX=   ,F10.5,5X,7HXFCREW=,F10.5)
 390  FORMAT(10X,7HXFENG= ,F10.5,5X,8HXFLIFTF=,F10.5)
 400  FORMAT(10X,7HXFAMUN=,F10.5,5X,8HXFBOMB= ,F10.5,5X,7HXFMISS=,F10.5)
 405  FORMAT(10X,7HXFCARG=,F10.5,5X,8HXFBB2 = ,F10.5)
 410  FORMAT(10X,7HCELV=  ,F10.5,5X,8HCBALHT= ,F10.5,5X,7HTZCHT= ,F10.5)
 411  FORMAT(10X,8HAWTOT = ,F10.5,5X,7HZMU  = ,F10.5)
 412  FORMAT(10X,7HXT1 =  ,F10.5,5X,8HXT2 =   ,F10.5,5X,7HZFMG = ,F10.5)
 413  FORMAT(10X,7HZFT1 = ,F10.5,5X,8HZFT2 =  ,F10.5,5X,7HZFD =  ,F10.5)
 414  FORMAT(10X,7HTSPLIT=,F10.5,5X,8HGAMAT1= ,F10.5,5X,7HGAMAT2=,F10.5)
 415  FORMAT(10X,7HIHT =  ,F10.5,5X,8HICN =   ,F10.5,5X,7HELVDEF=,F10.5)
 416  FORMAT(10X,7HVROT = ,F10.5,5X,8HSPANCN1=,F10.5,5X,7HZRTCN1=,F10.5)
 417  FORMAT(10X,7HSCN1 = ,F10.5,5X,8HXLECN = ,F10.5,5X,7HELVDMX=,F10.5)
 418  FORMAT(10X,7HTRCN1 =,F10.5,5X,8HIVECT = ,I3)
 420  FORMAT(10X,7HIETANK=,I3,5X,5HIBS= ,I3,5X,5HIMS= ,I3,5X,5HIAS= ,I3)
 430  FORMAT(10X,7HNPHASE=,I3,5X,8HIDBPRT= ,I3,5X,8HICGPRT= ,I3)
C
C
        RETURN
        END
C
C    ***********************************************************************
```

```
C
      SUBROUTINE STABCALC1
C
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
      LGFLAG = 0
  112 CONTINUE
      I = 1
      IPHASE = 0
      XCGFRWD(1) = 0
      XCGAFT(2) = BODL
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
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
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 THEN 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 - WFTO
```

```
        ENDIF
      ELSE
        WFP = WFP - WFTO
      ENDIF
C
      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
      IF(IPHASE.GE.IBS) WZB = 0.
      IF(IPHASE.GE.IWS) WZM = 0.
      IF(IPHASE.GE.IAS) WZA = 0.
C
      I = IPHASE + 2
C
      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
          GO TO 112
        ENDIF
C
  100 CONTINUE
C
C      THIS SET OF INSTRUCTIONS ACTIVATES THE VECTORED THRUST ANALYSIS
C      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(/,10X,35HWARNING!! C.G. IS FORWARD OF LIMITS,/,15X,
     139H CONTROL SURFACE INCREASED BY 5 SQ. FT.,/,15X,6HXCG = ,F12.6,
     25X,10HXCGFRWD = ,F12.6)
       RETURN
       END
C
C   ******************************************************************
C
       SUBROUTINE TAILSIZE(I,IPHASE)
C
C      THIS SUBROUTINE IS USED TO CALL THOSE ROUTINES NEEDED TO SIZE THE
C      HORIZONTAL TAIL AT TAKEOFF.
C
       INCLUDE 'STBCM.INC'
C
C      DATATRANS SUBROUTINE TRANSFERS OVER THE NECESSARY AERODYNAMICS
C       PARAMETERS.
C
       CALL DATATRANS
C
C      THIS SUBROUTINE DOES ALL THE SECONDARY AERODYNAMICS CALCULATIONS.
C
       CALL STABCALC3(I,IPHASE)
C
C      THIS SUBROUTINE DETERMINES THE TAIL SIZE NEEDED FOR ROTATION.
C
       CALL TOROTATE(I,IPHASE)
C
       RETURN
       END
C
C   ******************************************************************
C
         SUBROUTINE STABCALC2(I,IPHASE,CGFLAG)
C
C    THIS SUBROUTINE IS USED TO CALCULATE ANY PARAMETERS NEEDED
C      BY THE SUBROUTINES WHICH ARE NOT CALCULATED BY THE REST OF ACSYNT.
C
       INCLUDE 'STBCM.INC'
C
```

```
C     THIS SERIES OF INSTRUCTIONS SETS THE MACH NUMBER EITHER TO THE
C      TAKE OFF MACH NUMBER OR THE MACH NUMBER AT THE START OF A MISSION
C      PHASE.  IT THEN CALLS AERODYNAMICS TO GET THE NEW CLALPHA OF THE
C      WING AND THEN CALLS THE AERODYNAMIC CENTER SUBROUTINE.
C
      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,IGEO,KGPRNT,IGPLT)
      ZCLALFA = CLALFA
      ZMACH = MACH
C
C    NOW CALCULATE THE AERODYNAMIC CENTER INCORPERATING MACH EFFECTS,
C     SWEEP ANGLE EFFECTS, AND TAPER EFFECTS.
C
      CALL XACCALC(ZMACH,ZCLALFA)
C
C    NOW CALCULATE THE CENTER OF GRAVITY OF THE AIRCRAFT.
C
      CALL XCGCALC(I)
C
 200  CONTINUE
C
C    THE FOLLOWING SERIES OF COMMANDS CALCULATE THE PITCHING MOMENT
C     CURVE SLOPE FOR THE AIRCRAFT.  THESE CALCULATIONS ARE BASED ON
C     EQUATIONS TAKEN FROM REFERENCE 4, CHAPTER 5.  THEY INCLUDE THE
C     TAIL, WING, CANARD, FUSELAGE, AND JET(S).
C
C       **TAIL CONTRIBUTION TO AIRCRAFT STABILITY
C
        CSC=CLALFA*SWG*CBARWG
        DCMDCL1=-CLAFHT*SHT1*(XQCBHT-XCG(I))*ETAHT*(1.-DEDA)/CSC
C
C     THIS ACCOUNTS FOR STICK FREE EFFECTS, DETERMINED IN STABCALC3.
C
        DCMDCLT=DCMDCL1*DCMTMUL
C
C       **CALCULATE THE PITCHING MOMENT SLOPE CONTRIBUTION DUE TO
C         THE CANARD. NOTE THAT THERE ARE TWO SETS OF COMMANDS FOR
C         THE CANARD, ONE FOR A CANARD ALONE, AND ONE FOR A CANARD
C         AND AFT TAIL CONFIGURATION.
C
      IF(XLECN .EQ. 0.0) THEN
        VOLCN = (SCN1/SWG)*((XCG(I)-XQCBCN)/CBARWG)
      ELSE
        VOLCN = (SCN1/SWG)*((XCG(I)-XLECN)/CBARWG)
      ENDIF
```

```
      DCMCLCN=(CLACN/CLALFA)*VOLCN*ETACN*ODEDA
C
C        COMPUTE THE STICK FREE DCMDCL OF THE CANARD IF THERE IS
C        NO HOROZONTAL STABALIZER, OTHERWISE THE H.T. IS ASSUMED
C        TO BE FREE AND THE CANARD IS USED ONLY FOR TRIM.
C
      IF (SCN .GT. O. .AND. SHT .LE. O.) THEN
        DCMCMUL = DCMTMUL
        DCMCLCN = DCMCLCN * DCMCMUL
      ELSE
        DCMCMUL = 1.
      ENDIF
C
C        **WING CONTRIBUTION TO AIRCRAFT STABILITY
C
        DCMDCLW=((XCG(I)-XLE)/CBARWG)-((XACWG-XLE)/CBARWG)
C
C        **FUSALAGE CONTRIBUTION TO STABILITY
C
C        THIS USES THE EASY METHOD FROM REFERENCE 4 WHICH CURVE-FITS
C         THE GRAPH OF KF (FIGURE 5-16) TO DETERMINE THE FUSSELAGE EFFECTS.
C
        KF=((XLEWG+ROOTWG/4.)/BODL)*100.
        KF=.0023+.0002062*KF-.000005762*KF**2+.0000002499*KF**3
      CLARAD = CLALFA * 57.29578
        DCMDCLF=(KF*(BDMAX**2)*BODL)/(SWG*CBARWG*CLARAD)
C
C     CALCULATE THE AERODYNAMIC CENTER FOR THE WING AND FUSSELAGE BY
C     ADDING THE EFFECTS OF THE FUSELAGE TO THE AERODYNAMIC CENTER
C     OF THE WING.
C
      XAWB = ((XACWG-XLE)/CBARWG) + DCMDCLF
C
C        **STABILITY EFFECTS DUE TO THE JET(S)
C
C        1; EFFECTS DUE TO THRUST
        TN1 = TNT(IPHASE)*TSPLIT
        TN2 = TNT(IPHASE) - TN1
        DDT1 = (TN1 * ZT1) / (SUMWT * CBARWG)
        DDT2 = (TN2 * ZT2) / (SUMWT * CBARWG)
        DDCMTH = DDT1 + DDT2
C        2; EFFECTS DUE TO NORMAL FORCE (THERE ARE NOT ANY YET.)
        DDCMNF = 0.
C        TOTAL JET EFFECTS
        DCMDCLJ = DDCMTH + DDCMNF
C
C                **OVERALL MOMENT-CURVE SLOPE
C
C   THIS IS JUST A SUMMATION OF ALL THE PARTS.
C
        DCMDCL(I)=DCMDCLW+DCMCLCN+DCMDCLF+DCMDCLT+DCMDCLJ
C
C                **AIRCRAFT STICK-FREE NEUTRAL POINT
C
```

```
C    THIS USES AN EQUATION FROM REFERENCE 2 (CHAPTER 5-EQN. 5.145)
C     MODIFIED TO INCLUDE A CANARD.  ASSUMES THRUST EFFECTS AND
C     FUSELAGE EFFECTS DO NOT CHANGE WITH ANGLE OF ATTACK.
C
      CLHR = (CLAFHT/CLALFA)*(SHT1/SWG)*ETAHT
      CLCR = (CLACN/CLALFA)*(SCN1/SWG)*ETACN
      IF (SHT .NE. 0.0) THEN
      FNEUTPT(I)=(XAWB+CLHR*(XQCBHT/CBARWG)*(1.-DEDA)*DCMTMUL)/(1.
     $ +CLHR*(1.-DEDA)*DCMTMUL)
      ELSE
      FNEUTPT(I)=(XAWB+CLCR*(XQCBCN/CBARWG)*ODEDA*DCMCMUL)/(1.+
     $ CLCR*ODEDA*DCMCMUL)
      ENDIF
C
C                 **CG AFT LIMIT
C
C     THE AFT C.G. LIMIT IS THE NEUTRAL POINT OF THE AIRCRAFT.
C
        XCGAFT(I+1) = XLE + (FNEUTPT(I) * CBARWG)
C
C                 **STATIC MARGIN
C
C    THE STATIC MARGIN IS SIMPLY THE DIFFERENCE BETWEEN THE
C     NEUTRAL POINT AND THE CENTER OF GRAVITY.
C
        SM(I)= ((XCGAFT(I+1) - XCG(I))/CBARWG)*100.
C
C    CHECK TO SEE IF THE C.G. FOR EACH PHASE IS AFT OF THE AFT C.G.
C     LIMIT.  IF IT IS, THEN THE TAIL SIZE IS INCREASED, AND THE
C     ENTIRE SUBROUTINE RERUN.
C
      IF(XCGAFT(I+1) .LT. XCG(I))THEN
         WRITE(6,300)XCG(I),XCGAFT(I+1)
         IF(SCN .EQ. 0.0)THEN
            SHT1 = SHT1 + 5.0
            CGFLAG=1
            RETURN
         ELSE
            SCN1 = SCN1 + 5.0
            CGFLAG=1
            RETURN
         ENDIF
      ENDIF
C
        IF (IDBPRT.EQ.0) GO TO 400
C
C    THIS IS THE WRITE STATEMENTS TO BE USED WHEN COMPLETE DEBUGING
C     INFORMATION IS DESIRED.  THIS IS STARTED BY SETTING IDBPRT TO 1.
C
      IF(IPHASE .LE. 0)THEN
        WRITE(6,59)
      ELSE
        WRITE(6,60) IPHASE
      ENDIF
```

```
      WRITE(6,100) XCG(I)
      WRITE(6,105) (XAWB*CBARWG)+XLE
      WRITE(6,110) DCMDCL(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) DCMDCLT,DCMDCLW
      WRITE(6,425) DCMDCLF,DCMCLCN
      WRITE(6,430) DCMDCLJ
      WRITE(6,440) LWB,LHT,LCN
      WRITE(6,450) SHT1,SCN1
C
 400  CONTINUE
  59  FORMAT(5X,19HDATA AFTER TAKE-OFF)
  60  FORMAT(5X,20HMISSION PHASE NUMBER,I5)
 100  FORMAT(10X,6HXCG = F15.7)
 105  FORMAT(10X,6HXAC = F15.7)
 110  FORMAT(10X,9HDCMDCL = F15.7)
 120  FORMAT(10X,9HNEUTPT = F15.7)
 130  FORMAT(10X,16HSTATIC MARGIN = F15.7)
 140  FORMAT(10X,9HAFT CG = F15.7)
 410  FORMAT(5X,8HSUMMO = ,F15.6,5X,8HSUMWT = ,F15.6)
 420  FORMAT(5X,10HDCMDCLT = ,F15.8,5X,10HDCMDCLW = ,F15.8)
 425  FORMAT(5X,10HDCMDCLF = ,F15.8,5X,10HDCMDCLCN =,F15.8)
 430  FORMAT(5X,10HDCMDCLJ = ,F15.8)
 440  FORMAT(5X,6HLWB = ,F15.8,5X,6HLHT = ,F15.8,5X,6HLCN = ,F15.8)
 450  FORMAT(5X,6HSHT = ,F15.8,5X,6HSCN = ,F15.8)
 300  FORMAT(10X,31HWARNING!! C.G. IS AFT OF LIMITS,/,15X,
     144H CONTROL SURFACE SIZE INCREASED BY 5 SQ. FT.,/,15X,6HXCG = ,
     2F15.7,5X,8HAFTCG = ,F15.7)
      RETURN
      END
C
C     ******************************************************************
C
      SUBROUTINE STABCALC3(I,IPHASE)
C
C     THIS SUBROUTINE DETERMINES THE AERODYNAMIC CENTER AND C.G. DURING
C      TAKEOFF, IT DETERMINES THE STICK FREE EFFECTS, IT DETERMINES
C      THE UPWASH EFFECTS ON THE CANARD, AND THE LIFT-CURVE SLOPE
C      OF THE CANARD.
C
      INCLUDE 'STBCM.INC'
C
C         DETERMINE THE AERODYNAMIC CENTER OF THE WING,
C         USING THE GEOMETRIC SHAPE AND THE 2-D AND 3-D
C         LIFT COEFFICIENTS.  THIS USES THE INPUT OF THE TAKEOFF MACH
C         NUMBER.
C
      IF(AMTO .LE. 0.0)THEN
         AMTO = VROT
      ENDIF
C     NEED TO TRANSFER LIFT-CURVE SLOPE DATA FROM AERODYNAMICS MODULE
```

```
C        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
C
      CALL XACCALC(ZMACH,ZCLALFA)
C
C    **** DETERMINE THE CENTER-OF-GRAVITY FOR THE AIRCRAFT ****
C
      CALL XCGCALC(I)
C
C     ** DETERMINE THE STICK FREE MUTIPLIER FOR THE HORIZONTAL TAIL.
C       THIS WAS DONE BY CURVE FITTING TO DETERMINE THE TAIL
C       HINGE MOMENTS DUE TO ALPHA AND ELEVATOR DEFLECTION.
C       THE METHOD USED IS FROM REFERENCE 3, CHAPTER 8. EQNS 8.42 AND
C       8.46  THE GRAPHS CURVE FIT ARE FIGURES 8.11, AND 8.12.
C       INPUTS INCLUDE THE ELEVATOR SPAN, BALLANCE SPAN,
C       AS WELL AS THE AR AND t/c OF THE TAIL.
C
      IF (SCN .GT. 0.0 .AND. SHT .LE. 0.0) THEN
                CZELV = CELV*CBARCN
                CZBALHT = CBALHT * CELV
                CHT = CBARCN
                ARTAIL = ARCN
                THT = TZCHT * CBARCN
        ELSE
                CZELV = CELV*CBARHT
                CZBALHT = CBALHT * CELV
                CHT = CBARHT
                ARTAIL = ARHT
                THT = TZCHT * CBARHT
      ENDIF
C        NEED TO DETERMINE THE BALLANCE RATIO OF THE ELEVATOR.  IF IT IS
C        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 = TZCHT
C
C HERE WE ARE CURVE-FITTING THE TWO GRAPHS TO GET THE VALUES FOR
C  K1 AND K2, BY WHICH WE THEN GET BG1 AND BG2.
C  THESE ARE GRAPHS ARE FROM REFERENCE 3, FIGURES 8.11 AND 8.12
C
        K11=.1027+3.5772*CEC-1.8662*CEC**2+.174*CEC**3-.3587*CEC**4
        K12=1.1692-2.5003*TC+23.4944*TC**2-202.1809*TC**3+546.3254*TC**4
```

```
      K13=1.+2.8032*BR-31.5474*BR**2+72.8412*BR**3-56.8704*BR**4
      K14=1.-2.2462*A1A+1.2067*A1A**2-1.378*A1A**3+1.1478*A1A**4
      BG1=-.55*K11*K12*K13*K14
C
    K21=.7504+.8193*CEC-.0829*CEC**2+.2082*CEC**3-.1449*CEC**4
    K22=1.+.15*BR-12.4349*BR**2+18.7076*BR**3-12.8205*BR**4
    K23=1.-1.0762*A1A+.41303*A1A**2+.41321*A1A**3-.37067*A1A**4
    K24=1.0748-.7949*TC+9.1283*TC**2-105.7365*TC**3+233.0985*TC**4
      BG2=-.89*K21*K22*K23*K24
C
C   TAU IS DETERMINED FROM REFERENCE 3, CHAPTER 3, FIGURE 3.32.
C    THIS IS KNOWN AS THE ELEVATOR EFFECTIVNESS FACTOR.
C
      TAU=4.1065*CEC-8.9175*CEC**2+9.7178*CEC**3-3.9307*CEC**4
C
C    HERE WE CALCULATE THE 'FREE ELEVATOR FACTOR' (DCMTMUL), WHICH WILL
C     BE  MULTIPLIED TO THE ELEVATOR ANGLE OF ATTACK TO DETERMINE STICK
C     FREE EFFECTS.
C
      DCMTMUL=1- (TAU * (BG1/BG2))
C
C     **CANARD CONTRIBUTION TO AIRCRAFT STABILITY
C
C     FIRST DETERMINE THE UPWASH ON THE CANARD DUE TO THE WING.
C     THIS IS FOUND BY CURVE FITTING FIGURE 8.22c FROM REFERENCE 3.
C
    CO = ROOTWG
    IF(XLECN .EQ. 0.0) THEN
      XFOR = (XLEWG + ROOTWG/4.) - XQCBCN
    ELSE
      XFOR = XLECN
    ENDIF
    XC = XFOR/CO
    IF (XC .GE. 5.)THEN
      ODEDA = 1.
      GO TO 190
    ENDIF
    IF (ARWG .LE. 8.) THEN
    X1 = 6.
    X2 = 8.
    Y1=2.6538-3.3623*XC+2.9508*XC**2.-1.2856*XC**3.
   $    +.2706*XC**4.-.0219*XC**5.
    Y2=2.8419-3.6494*XC+3.1799*XC**2.-1.3833*XC**3.
   $    +.2916*XC**4.-.0236*XC**5.
    ODEDA=((X2-ARWG)*Y1-(X1-ARWG)*Y2)/(X2-X1)
    GO TO 190
    ENDIF
    IF (ARWG .GT. 8. .AND. ARWG .LE. 10.) THEN
    X1 = 8.
    X2 = 10.
    Y1=2.8419-3.6494*XC+3.1799*XC**2.-1.3833*XC**3.
   $    +.2916*XC**4.-.0236*XC**5.
    Y2=2.9405-3.7399*XC+3.1866*XC**2.-1.3592*XC**3.
   $    +.2821*XC**4-.0226*XC**5
```

```
      ODEDA=((X2-ARWG)*Y1-(X1-ARWG)*Y2)/(X2-X1)
      GO TO 190
      ENDIF
      IF (ARWG .GT. 10.) THEN
      X1 = 10.
      X2 = 12.
      Y1=2.9405-3.7399*XC+3.1866*XC**2.-1.3592*XC**3.
     $    +.2821*XC**4.-.0226*XC**5.
      Y2=3.1183-4.1221*XC+3.5627*XC**2.-1.5362*XC**3.
     $    +.3217*XC**4.-.0259*XC**5.
      ODEDA=((X2-ARWG)*Y1-(X1-ARWG)*Y2)/(X2-X1)
      ENDIF
 190  CONTINUE
C
C        CALCULATE LIFT CURVE SLOPE FOR THE CANARD.  THIS ASSUMES
C        A THEORETICAL 2*PI 2-D LIFT-CURVE SLOPE, AND A LINEARLY TAPERED
C        CANARD PLANFORM.   THIS USED EQN 5.85 FROM REFERENCE 3 TO
C        DETERMINE THE LIFT-CURVE SLOPE, AND EQNS 5.89 a AND b TO DETERMINE THE
C        SWEEP AT THE ONE-HALF CHORD OF THE WING.
C
C     HERE IS CALCULATED THE SWEEP AT THE ONE-HALF CHORD POINT.
C
      IF (SCN .NE. 0.0 .OR. SCN1 .NE. 0.0) THEN
         IF(ARCN .EQ. 0.0) ARCN = (SPANCN1*SPANCN1) / 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 = 0.785398
               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
C
C    HERE IS CALCULATED THE LIFT-CURVE SLOPE
C
            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
C
        RETURN
        END
C
C  ***********************************************************************
C
      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
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 = BDMAX / 2.
      XLECL = XLEWG - (BODRAD * TAN(SWPWG/57.29578))
      XTECL = XLEWG + ROOTWG
      A = XLECL + ((XTECL - XLECL)/4.)
C
      RAD = 57.29578
      PI = 3.1415926
C
      CLACLA = 1.0 / (1.0 - ((RAD * ZCLALFA) / (PI * ARWG)))
      XA = ((1.0 + 2.0*TRWG)/(1.0 + TRWG)) * ((SPANWG / 2.0) *
     $ TAN(SWPWG/RAD)) * CLACLA * 0.3333333
      XACWG = XA + A
      XLE = XQCBWG - CBARWG/4.
C
      RETURN
      END
C
C  ***********************************************************************
C
      SUBROUTINE XCGCALC(I)
C
      INCLUDE 'STBCM.INC'
C
```

```
C        THIS SUBROUTINE IS CALLED TO CALCULATE THE RELATIVE POSITIONS
C        AND MOMENTS OF EACH OF THE AIRCRAFT WEIGHTS WHICH ARE
C        USED TO DETERMINE OVERALL AIRCRAFT C.G.   NOTE THAT PLACEMENT
C        OF MOST WEIGHTS IS ARBITRARY, BUT CAN BE ADJUSTED BY THE
C        USERS WITH THE XF*** INPUTS IN /STABIN/.
C
 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 = XFBB2 * BODL
        MBB2 = XBB2 * WBB2
      XMISS = XFMISS * BODL
        MMISS = XMISS * WZM
      XAMMUN = XFAMMUN * BODL
        MAMMUN = XAMMUN * WZA
      XNA = XFNA * BODL
        MNA = XNA * WNA
C
C        THE WEIGHTS MODULE DOES NOT GIVE OUT THE PIVOT WEIGHTS
C         DIRRECTLY, SO IT IS CALCULATED BY SUBTRACTING THE DIFFERENT
C         PARTS FROM THE AIRFRAME WIEGHT.
C
      WPIV = WAF-WBODY-WHT-WLG-WNA-WVT-WWING-WCAND
```

```
      IF (WPIV .LT. 0.) WPIV = 0.
        XPIV = XFPIV * BODL
        MPIV = WPIV * XPIV
C
C         THE SAME GOES FOR THE LIFT FAN AS GOES FOR THE PIVOTS ABOVE.
C
      WLFTF = WPS - WE - WFS
      IF (WLFTF .LT. 0.) WLFTF = 0.
        XLIFTF = XFLIFTF * BODL
        MLIFTF = WLFTF * XLIFTF
C
      WLGFRT = WFLGFRT * WLG
        XLGFRT = XFLGFRT * BODL
        MLGFRT = WLGFRT * XLGFRT
      WLGR = WLG - WLGFRT
      IF(LGFLAG .GT. 0) GO TO 1100
        XLGR = XFLGR * BODL
 1100  CONTINUE
        MLGR = XLGR * WLGR
C
        WZENG = WE / EN
        DO 220 J = 1 , EN
                XENG(J) = (XLEPOD(J) + XFENG * PODL)
                MENG(J) = WZENG * XENG(J)
  220   CONTINUE
C
C     FURNISHING WEIGHT IS FOUND BY SUBTRACTING THE KNOWN WEIGHTS FROM
C      THE WEIGHTS OF THE FIXED EQUIPMENT.
C
      WFUR = WFEQ-WAIRC-WELT-WAPU-WEP-WPA-WHDP-WINST-WSC
        IF (WFUR .LT. 0.) WFUR = 0.
        XFUR = XFFUR * BODL
        MFUR = XFUR * WFUR
C
C         PAYLOAD IS FOUND LIKE FURNISHINGS
C
      WPAYL = WPL-WCREW-WAMMUN-WBB2-WBOMB-WMISS-WETANK-WCARGO
        IF (WPAYL .LT. 0.) WPAYL = 0.
        XPAYL = XFPAYL * BODL
        MPAYL = XPAYL * WPAYL
C
C
C          CALCULATE MOMENTS FOR FUEL
      IF(FUFRAC .GT. 1.0) FUFRAC=1.0
        WFWG = FUFRAC * WFF
      XWGFUEL = (XFFWG * CBARWG + XQCBWG)
        MFWG = WFWG * XWGFUEL
        WFFUS = WFP - WFWG
      IF(WFFUS .LT. 0.0) WFFUS=0.0
      XFUSF = (BODL * XFFFUS)
        MFFUS = WFFUS * XFUSF
C
C          NOW INCLUDE ANY EXTERNAL TANKS DEPENDING ON THE INPUT IETANK.
C
```

```
      XCTIPWG = ROOTWG*TRWG
      XLETIP = XLEWG+.25*ROOTWG+SPANWG/2.*TAN(SWPWG/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 WEFW = .75*WZEF+.75*WZET
      WEFWT = WZEF+WZET-WEFW
      XEF1 = (XQCBWG+CBARWG/4.)
      XEF2 = (XLETIP+.5*XCTIPWG)
      MEF = XEF1*WEFW+XEF2*WEFWT
  600   CONTINUE
C
C       NOW FIND MOMENTS AND WEIGHTS OF FUEL.
        WFL = WFFUS+WFWG+WZEF+WZET
        MFL = MFWG+MFFUS+MEF
C
C       DETERMINE THE SUM TOTAL OF THE WEIGHTS AND THE
C       C.G. OF THE AIRCRAFT.
C
        SUMMO = MBOD+MCAN+MHT+MNA+MVT+MWG+MAIRC+MAPU+MELT+MSC+MPIV
     1   +MEP+MINST+MHDP+MPA+MFUR+MPAYL+MCREW+MAMMUN+MMISS+MCARGO
     2   +MBOMB+MFL+MLGFRT+MLGR+MLIFTF+MBB2+MREM
C
        DO 230 J = 1 , EN
```

```
                SUMMO = SUMMO + MENG(J)
  230   CONTINUE
C
        SUMWT = WBODY+WCAND+WHT+WNA+WVT+WWING+WAIRC+WAPU+WELT+WSC+WPIV
     1   +WEP+WINST+WHDP+WPA+WFUR+WPAYL+WZA+WZM+WLGFRT+WLGR+WCARGO
     2   +WCREW+WZB+WREMM+WFL+WLFTF+WE+WBB2
C
        XCG(I) = SUMMO / SUMWT
C
C     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
C
C       THIS PRINT FLAG PRINTS THE POSITIONS AND WEIGHTS OF
C       THE DIFFERENT COMPONENTS USED TO DETERMINE CORRECT PLACEMENT
C       OF THE AIRCRAFTS C.G.
C
      CALL CGPRINT
C
  800   CONTINUE
C
C     THIS WRITE STATEMENT PRINTS A WARNING IF THE C.G. IS CLOSER TO THE
C MAIN GEAR THEN THE 15 DEGREE LIMIT.  THIS LIMIT IS TAKEN FROM NAVY
C SPECIFICATIONS SD-24J, WHICH GIVES A GOOD DESIGN VALUE.  THIS
C SPECIFICATION WAS TAKEN FROM REFERENCE 5, PAGE 48, FIGURE 4.3.
C
      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
         GO TO 1000
      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,/)
C
      RETURN
      END
C
C   ****************************************************************
C
      SUBROUTINE TOROTATE(I,IPHASE)
C
C     THIS SUBROUTINE DETERMINES THE SIZE OF THE HORIZONTAL
C     TAIL OR CANARD, NEEDED TO ROTATE THE AIRCRAFT ON TAKEOFF.
C     THIS USES AN ALTITUDE OF 10,000 FT FOR TAKEOFF DENSITY.
C      THE METHOD USED HERE IS TAKEN FROM REFERENCE 2, CHAPTER 5 PAGE
C      373 - 374.  THIS IS MODIFIED TO INCLUDE A CANARD AND ANY THRUST
C      OFFSET DUE TO PLACEMENT OF THE ENGINES.
```

```
C
      INCLUDE 'STBCM.INC'
C
C
      IF (ARHT .LE. 13.) THEN
        A = ARHT
        QCLAFHT=-.0282+.0359*A-.0049*A**2+.000345*A**3-.000009561*A**4
      ELSE
        QCLAFHT = (2.*3.1415926)/57.29578
      ENDIF
C
C     FIRST DETERMINE THE DYNAMIC PRESSURE, AND THE LIFT OF THE WING.
C
      VTOR = VROT * 1077.4
      QTOR = 0.5 * .0017556 * (VTOR**2.)
C
C     NOW CALCULATE DISTANCE PERAMETERS SPECIFYING DISTANCES FROM
C     THE WING LEADING EDGE TO THE PARTICULAR COMPONENT.
C
      ZMG = ZFMG * BDMAX
      ZT1 = ZFT1 * BDMAX
      ZT2 = ZFT2 * BDMAX
      ZD = ZFD * BDMAX
      T1RAD = GAMAT1 / 57.29578
      T2RAD = GAMAT2 / 57.29578
      XLEMG = XLGR - XQCBWG
      XLECG = XCG(I) - XQCBWG
      XLET1 = XQCBWG - (XT1*BODL)
      XLET2 = (XT2*BODL) - XQCBWG
      XLEACWG = XACWG - XQCBWG
      IF(XLECN .EQ. 0.0) THEN
        XLECN = XQCBWG - XQCBCN
      ENDIF
      XLEHT = XQCBHT - XQCBWG
C
C     CALCULATE THE LIFT OF THE WING.  THIS INCLUDES GROUND EFFECTS
C     FOR THE LIFT COEFFICIENT.  THIS IS JUST MULTIPLICATION OF THE
C     WING LIFT-CURVE SLOPE MODIFIED FOR GROUND EFFECT,
C     BY THE ANGLE OF ATTACK OF THE WING, AND THE DYNAMIC PRESSURE.
C     FIRST MODIFY FOR GROUND EFFECTS
C
      DG = (ZRTWG+ZMG) / (SPANWG/2.)
      CALL GRNDEFFECT(DG,AFGDAF)
      CLAGRD = AFGDAF * CLALFA
      LWB = CLAGRD * (ALFTO) * QTOR * SWG
C
C     CALCULATE THE CHANGE IN LIFT-CURVE SLOPE OF THE TAIL WHILE
C     IN GROUND EFFECT.
C
      IF(SPANHT .LE. 0.0) GO TO 13
      DG = (ZRTHT+ZMG) / (SPANHT/2.)
      CALL GRNDEFFECT(DG,AFGDAF)
      CLAHTGRD = AFGDAF * QCLAFHT
   13 CONTINUE
```

```
C
C
C     CALCULATE THE CHANGE IN LIFT-CURVE SLOPE OF THE CANARD WHILE
C     IN GROUND EFFECT.
C
      IF(SPANCN .LE. 0.0) GO TO 14
      DG = (ZRTCN+ZMG) / (SPANCN/2.)
      CALL GRNDEFFECT(DG,AFGDAF)
      CLACNGRD = AFGDAF * CLACN
   14 CONTINUE
      IF(SPANCN1 .LE. 0.0) GO TO 41
      DG = (ZRTCN1+ZMG) / (SPANCN1/2.)
      CALL GRNDEFFECT(DG,AFGDAF)
      CLACNGRD = AFGDAF * CLACN
   41 CONTINUE
C
C     NOW CALCULATE THE DOWNWASH ANGLE OF THE WING ON THE TAIL.
C
      IF (SHT .EQ. 0.0) GO TO 10
      CALL DOWNWASH(CLTO,ALFTO,DWANGLE)
   10 CONTINUE
C
C       THE CALL TO TRAJDT IS USED TO CALCULATE THE MAXIMUM THRUST AT
C       TAKEOFF ROTATION OF THE AIRCRAFT.
C
        MODNO4=4
      CALL TRAJDT(MODNO4,ICALC,NERROR,IGEO,KGPRNT,IGPLT)
      IF(NERROR.GE.2) RETURN
      TN = THRUST*EN
C
       TWZTO=TN
C
C     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
C
C     NOW CALCULATE THE LIFT OF THE HORIZONTAL TAIL AND CANARD
C     DEPENDING ON WHICH OF THE TWO EXIST.  IF BOTH EXIST, THE
C     CANARD IS ASSUMED TO BE USED ONLY FOR TRIM, THE HORIZONTAL TAIL
C     IS THE DESIRED SIZE.
C      THE METHOD USED HERE IS TAKEN FROM REFERENCE 2, CHAPTER 5 PAGE
C      373 - 374.  THIS IS MODIFIED TO INCLUDE A CANARD AND ANY THRUST
C      OFFSET DUE TO PLACEMENT OF THE ENGINES.
C
      IF (SCN1 .EQ. 0.0 .AND. SCN .EQ. 0.0) THEN
```

```
C     ** THIS IS THE CASE WHERE THERE IS A HORIZONTAL TAIL ONLY **
        LH1 = CMWBTO * QTOR * SWG * CBARWG
        LH3 = ZT1 * T1
        LH5 = ZT2 * T2
        LH6 = WGTO * (XLEMG - XLECG + ZMU * ZMG)
        LH7 = LWB * (XLEMG - XLEACWG + ZMU * ZMG)
        LH8 = D * ZD
        DENOM = XLEHT - XLEMG - ZMU * ZMG
        LHT = (LH1 + LH2 + LH3 - LH4 + LH5 - LH6 + LH7 - LH8) / DENOM
C
        SHT1=LHT/(CLAHTGRD*QTOR*(ALFTO-DWANGLE+IHT+TAU*ELVDEF))
      ELSE
      IF (SHT .EQ. 0.) THEN
C
C     ** THIS IS THE CASE WHERE THERE IS A CANARD BUT NO H. TAIL **
C
        LC1 = CMWBTO * QTOR * SWG * CBARWG
        LC3 = ZT1 * T1
        LC5 = ZT2 * T2
        LC6 = WGTO * (XLEMG - XLECG + ZMU * ZMG)
        LC7 = LWB * (XLEACWG - XLEMG - ZMU * ZMG)
        LC8 = D * ZD
        DENOM = XLECN + XLEMG + ZMU * ZMG
        LCN = (-LC1 - LC2 - LC3 + LC4 - LC5 + LC6 + LC7 + LC8) / DENOM
C
        SCN1 = LCN/(CLACNGRD*QTOR*(ALFTO+ICN+TAU*ELVDEF))
      ELSE
C
C       ** THIS IS THE CASE WITH BOTH CANARD AND HORIZONTAL TAIL **
C
        LH1 = CMWBTO * QTOR * SWG * CBARWG
        LH3 = ZT1 * T1
        LH5 = ZT2 * T2
        LH6 = WGTO * (XLEMG - XLECG + ZMU * ZMG)
        LH7 = LWB * (XLEMG - XLEACWG + ZMU * ZMG)
        LH8 = D * ZD
        LCN = CLACNGRD * QTOR * SCN1 * (ALFTO + ICN)
        LH9 = LCN * (XLECN + XLEMG + ZMU * ZMG)
        DENOM = XLEHT - XLEMG - ZMU * ZMG
        LHT = (LH1 + LH2 + LH3 - LH4 + LH5 - LH6 + LH7 - LH8
     $  + LH9) / DENOM
C
        SHT1=LHT/(CLAHTGRD*QTOR*(ALFTO-DWANGLE+IHT+TAU*ELVDEF))
      ENDIF
      ENDIF
C
C
      RETURN
      END
C
C   ***********************************************************************
C
      SUBROUTINE XCGFORWARD(I)
C
```

```
C     THIS SUBROUTINE CALCULATES THE LONGITUDINAL CONTROL SURFACE
C     SIZE NEEDED TO MAINTAIN CONTROLABILITY OF THE AIRCRAFT IN
C     LANDING CONFIGURATION.  THAT IS MAX CL, LOW SPEED, HIGH DRAG,
C     IN GROUND EFFECT.  THIS IS DONE BY A SUMMATION OF MOMENTS ABOUT
C     THE CENTER OF GRAVITY.  WITH THIS EQUATION SET TO ZERO, THE
C     CENTER OF GRAVITY POSITION IS SOLVED FOR.
C
      INCLUDE 'STBCM.INC'
C
      VTHT = (CLAHTGRD * SHT1 * ETAHT)/ SWG
      VTCN = (CLACNGRD * SCN1 * ETACN)/ SWG
C
      IF (SHT .EQ. 0.0) GO TO 20
      CALL DOWNWASH(CLMAX,ALFMAX,DWANGLE)
   20 CONTINUE
C
C     DETERMINE THE DIFFERENT ANGLES OF ATTACK.
C
      ALFHT = ALFMAX + IHT - DWANGLE
      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
C
      XAWBZ = XLE + XAWB * CBARWG
      CM1 = CLMAX * XAWBZ
      CM2 = CMWBMAX * CBARWG
      CM3 = VTHT * ALFHT * XQCBHT
      CM4 = VTCN * ALFCN * XQCBCN
         CM5B = ZT1 * T1
         CM5D = ZT2 * T2
      CM5 = (CM5B + CM5D) * CBARWG
      DENOCM = CLMAX + (VTHT * ALFHT) + (VTCN * ALFCN)
C
C     THIS SOLVES FOR THE FORWARD C.G. LIMIT.
C
      XCGFRWD(I) = (CM1 - CM2 + CM3 + CM4 - CM5) / DENOCM
C
      IF(IDBPRT .NE. 0)THEN
         WRITE(6,10)I-2,XCGFRWD(I)
      ENDIF
C
   10 FORMAT(10X,8HXCGFRWD[,I2,4H] = ,F12.6)
      RETURN
      END
C
C **********************************************************************
```

```
C
      SUBROUTINE VECTHRUST(I)
C
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 POSTION 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'
C
      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 = (XCG(I) - XQCBCN) / CBARWG
      V4 = V4A * V4B
      SUMVT = V1 + V2 - V3 + V4
C
      GO TO (100,110,120), IVECT
 100  CONTINUE
      IF(T1*SIN(T1RAD) + T2*SIN(T2RAD) .LT. WEMP) THEN
         WRITE(6,500) T1*SIN(T1RAD)+T2*SIN(T2RAD),WEMP
      GO TO 170
      ENDIF
      V5 = ZT1 * T1 * COS(T1RAD)
      V6 = ((XT2 * BODL) - XCG(I)) * T2 * SIN(T2RAD)
      V7 = ZT2 * T2 * COS(T2RAD)
      SUMVTT = SUMVT + V5 - V6 + V7
      XT1 = (SUMVTT / (T1 * SIN(T1RAD)) + XCG(I)) / BODL
      GO TO 170
 110  CONTINUE
      NUMVT = (SUMVT/TWZTO)-((XT2*BODL)-XCG(I))*SIN(T2RAD)
     $ +ZT2*COS(T2RAD)
      DENOVT = ZT2*COS(T2RAD)-((XT2*BODL)-XCG(I))*SIN(T2RAD)
     $ -(XCG(I)-(XT1*BODL))*SIN(T1RAD)-ZT1*COS(T1RAD)
      TSPLIT = NUMVT / DENOVT
      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,61HWARNIG!! VERTICAL THRUST COMPONENT LESS THEN
     $ AIRCRAFT WEIGHT!,/,10X,7HTHRUST=,F15.7,5X,7HWEIGHT=,F15.7)
      RETURN
      END
C
C ***********************************************************************
C
      SUBROUTINE DATATRANS
C
      INCLUDE 'STBCM.INC'
C
C     THIS SUBROUTINE IS USED TO CALL THE DATA TRANSFER SUBROUTINE.  IT
C     IS USED TO DETERMINE THE ANGLE OF ATTACK, COEFFICIENT OF LIFT,AND
C     PITCHING MOMENT COEFFICIENT FOR TAKEOFF, MAX LIFT CONDITIONS,
C     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,KGPRNT,IGPLT)
      ALFTO = ALPHA
      CLTO = CL
      CDTO = CD
      CMWBTO = CMWB
C  %%%%%
      CL = 0.54
      CD = .54
      CMWB = 0.54
      ALPHA = 0.54
      IAO = 10
      CALL STBDT(ICALC,NERR,IGEO,KGPRNT,IGPLT)
      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,KGPRNT,IGPLT)
      ALFO = ALPHA
```

```
      CLO = CL
      CDO = CD
      CMWBO = CMWB
C
      RETURN
      END
C
C ***************************************************************************
C
      SUBROUTINE DOWNWASH(CLZ,ALFZ,DWANGLE)
C
C     THIS SUBROUTINE CALCULATES THE DOWNWASH ANGLE ON THE TAIL.  FIRST
C BY THE THEORETICAL METHOD, AND THEN BY POSITION OF THE TAIL RELATIVE
C TO THE WING VOTECIES.
C
      INCLUDE 'STBCM.INC'
C
C     CALCULATE THE DOWNWASH ANGLE OF THE WING ON THE TAIL.
C
      DWANGLE = ((1.62 * CLZ) / (3.1415926 * ARWG)) * 57.29578
C
C     NOW MODIFY THIS CALCULATION FOR THE POSITION OF THE TAIL.  THE
C     METHOD USED IS FROM REFERENCE 1, SECTION 4.4.1.
C     FIRST IS A CURVE FIT OF FIGURE 4.4.1-55.
C
      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
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```
         Y2 = -2.5098 + 6.2131*ALFRATIO
         A1 = ((X2-SWPWG)*Y1-(X1-SWPWG)*Y2)/(X2-X1)
         IF(A1 .LT. 0.0) A1 = 0.0
         GOTO 1000
      ENDIF
      IF(ALFRATIO .GE. 0.73) THEN
         IF(SWPWG .LE. 45.0) THEN
            X1 = 30.0
            X2 = 45.0
            Y1 = -2.6464 + 3.6316*ALFRATIO
            Y2 = -2.5098 + 6.2131*ALFRATIO
            A1 = ((X2-SWPWG)*Y1-(X1-SWPWG)*Y2)/(X2-X1)
            IF(A1 .LT. 0.0) A1 = 0.0
         ELSE
            X1 = 45.0
            X2 = 60.0
            Y1 = -2.8585 + 5.0852*ALFRATIO
            Y2 = -2.5098 + 6.2131*ALFRATIO
            A1 = ((X2-SWPWG)*Y1-(X1-SWPWG)*Y2)/(X2-X1)
            IF(A1 .LT. 0.0) A1 = 0.0
         ENDIF
      ENDIF
 1000 CONTINUE
      IF(TRWG .LT. 0.50) THEN
         X1 = 0.0
         X2 = 0.5
         Y1 = 1.0 - 0.150*A1
         Y2 = 1.0 - .1245*A1
         AEFA = ((X2 - TRWG)*Y1-(X1-TRWG)*Y2)/(X2-X1)
         AREFF = AEFA * ARWG
      ELSE
         X1 = 0.5
         X2 = 1.0
         Y1 = 1.0 - .1245*A1
         Y2 = 1.0 - .1000*A1
         AEFA = ((X2-TRWG)*Y1-(X1-TRWG)*Y2)/(X2-X1)
         AREFF = AEFA * ARWG
      ENDIF
      X1 = 0.0
      X2 = 1.0
      Y1 = -.20326+2.6606*AEFA-2.2631*(AEFA**2.)+.80556*(AEFA**3.)
      Y2 = -.016464 + 1.0168*AEFA
      SPANEFB = ((X2-TRWG)*Y1-(X1-TRWG)*Y2)/(X2-X1)
      SPANEFF = SPANEFB * SPANWG
 1100 CONTINUE
C
C     NOW SOLVE FOR THE VERTICAL POSITION OF THE VORTEX CORE.
C     THIS ASSUMES TRAILING EDGE SEPERATION.  THESE USE EQUATIONS
C     4.4.1-A THRUOGH 4.4.1-F IN REFERENCE 1, SECTION 4.4.1.
C
      SIGRU = (0.56 * ARWG) / CLZ
      BVRU = (0.78 + 0.10*(TRWG-0.4)+0.003*SWPWG)*SPANEFF
      LEFF = XQCBHT - XLETIP
      BV = SPANEFF - (SPANEFF - BVRU)*SQRT((2.0*LEFF)/(SPANWG*SIGRU))
```

```
      HH = ZRTHT - ZRTWG
      HVERT = HH - LEFF*((ALFZ/57.29578) - (0.41*CLZ)/(3.14159*AREFF))
     $ - (SPANEFF/2.0)*TAN(DIHED/57.29578)
C
C     NOW CALCULATE THE MODIFIED DOWNWASH ANGLE.
C
      DWANGLE = DWANGLE * (1 / (1 + ((2.0*HVERT)/BV)**2.))
C
      RETURN
      END
C
C
C    ******************************************************************
C
      SUBROUTINE GRNDEFFECT(DG,AFGDAF)
C
C     THIS SUBROUTINE DETERMINES THE CHANGES TO CLALPHA FOR TAKEOFF
C      AND LANDING CALCULATIONS.  THIS USES FIGURE 5-39 FROM REFERENCE
C      4, PG. 257.  THE HEIGHT FROM THE GROUND USED IS THE HEIGHT OF
C      AIRCRAFT WITH THE MAIN LANDING GEAR ON THE GROUND.
C
      INCLUDE 'STBCM.INC'
C
      IF (ARWG .LE. 6.) THEN
      X1 = 4.
      X2 = 6.
      Y1=1.2989-.72851*DG+.79231*DG**2.-.4131*DG**3.+.083452*DG**4.
      Y2=1.2054-.49627*DG+.54742*DG**2.-.29063*DG**3.+.059777*DG**4.
      AFGDAF=((X2-ARWG)*Y1-(X1-ARWG)*Y2)/(X2-X1)
      GO TO 190
      ENDIF
      IF (ARWG .GT. 6. .AND. ARWG .LE. 8.) THEN
      X1 = 6.
      X2 = 8.
      Y1=1.2054-.49627*DG+.54742*DG**2.-.29063*DG**3.+.059777*DG**4.
      Y2=1.1751-.46048*DG+.55655*DG**2.-.31668*DG**3.+.068063*DG**4.
      AFGDAF=((X2-ARWG)*Y1-(X1-ARWG)*Y2)/(X2-X1)
      GO TO 190
      ENDIF
      IF (ARWG .GT. 8. .AND. ARWG .LE. 10.) THEN
      X1 = 8.
      X2 = 10.
      Y1=1.1751-.46048*DG+.55655*DG**2.-.31668*DG**3.+.068063*DG**4.
      Y2=1.1498-.41037*DG+.51326*DG**2.-.30114*DG**3.+.066288*DG**4
      AFGDAF=((X2-ARWG)*Y1-(X1-ARWG)*Y2)/(X2-X1)
      GO TO 190
      ENDIF
      IF (ARWG .GT. 10.) THEN
      X1 = 10.
      X2 = 12.
      Y1=1.1498-.41037*DG+.51326*DG**2.-.30114*DG**3.+.066288*DG**4.
      Y2=1.1188-.31846*DG+.3826*DG**2.-.21725*DG**3.+.046757*DG**4.
      AFGDAF=((X2-ARWG)*Y1-(X1-ARWG)*Y2)/(X2-X1)
      ENDIF
```

```
 190  CONTINUE
C
      RETURN
      END
C
C   ***********************************************************************
C
        SUBROUTINE STABOUT
C
C  THIS SUBROUTINE PRINTS OUT THE PERTINATE DATA CALCULATED BY THE
C    MODULE.
C
      INCLUDE 'STBCM.INC'
C
      DO 53 I = 3,NPHASE+3
         IF(XCGFRWD(I) .GT. XCGFRWD(1))THEN
            XCGFRWD(1) = XCGFRWD(I)
         ENDIF
  53  CONTINUE
C
      DO 54 I = 3,NPHASE+3
         IF(XCGAFT(I) .LT. XCGAFT(2))THEN
            XCGAFT(2) = XCGAFT(I)
         ENDIF
  54  CONTINUE
C
      DO 55 I = 2,NPHASE+2
         SM(I) = ((XCGAFT(2) - XCG(I))/CBARWG)*100.
  55  CONTINUE
C
        WRITE(6,50)
        WRITE(6,105) (XAWB*CBARWG)+XLE
      IF(SHT1 .NE. 0.) THEN
        WRITE(6,150) SHT1
      ENDIF
      IF(SCN1 .NE. 0.) THEN
        WRITE(6,160) SCN1
      ENDIF
        WRITE(6,135) XCGFRWD(1)
        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) CGBAR(I)
        WRITE(6,110) DCMDCL(I)
        WRITE(6,130) SM(I)
  10  CONTINUE
C
      IF(IVECT .EQ. 0) THEN
```

```
        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(1H1,10X,21HSTABILITY OUTPUT DATA,//)
   59    FORMAT(5X,23HDATA AT END OF TAKE-OFF)
   60    FORMAT(5X,20HMISSION PHASE NUMBER,I5)
  100    FORMAT(10X,12HXCG      =     ,F15.7,4H ft.)
  105    FORMAT(10X,12HXAC      =     ,F15.7,4H ft.)
  107    FORMAT(10X,12HCGBAR    =     ,F15.7,8H % chord)
  110    FORMAT(10X,12HDCMDCL   =     ,F15.7)
  120    FORMAT(10X,12HNEUTPT   =     ,F15.7)
  130    FORMAT(10X,16HSTATIC MARGIN = ,F15.7,8H % chord)
  135    FORMAT(10X,12HXCGFRWD  =     ,F15.7,4H ft.)
  140    FORMAT(10X,12HAFT CG   =     ,F15.7,4H ft.)
  150    FORMAT(10X,12HTAIL SIZE = ,F15.7,8H sq. ft.)
  160    FORMAT(10X,14HCANARD SIZE = ,F15.7,8H sq. ft.)
  170    FORMAT(10X,15HVERTICAL THRUST,/,10X,12HFORE THRUST=,F15.7,
     $5X,12HAFT THRUST =,F15.7)
  180    FORMAT(10X,21HFORE THRUST POSITION=,F15.7,
     $5X,20HAFT THRUST POSITION=,F15.7)
  185 FORMAT(10X,22HVERT FORE THRUST POS.=,F15.7,5X,
     $22HVERT AFT THRUST POS. =,F15.7)
  190 FORMAT(10X,21HFORE THRUST ANGLE  = ,F15.7,5X,
     $22HAFT THRUST ANGLE    = ,F15.7)
  195 FORMAT(10X,24HTHRUST SPLIT (FORE/TOT)=,F15.7)
  200    FORMAT(/,10X,20HEND STABILITY OUTPUT,//)
C
        RETURN
        END
C
C    *******************************************************************
C
      SUBROUTINE CGPRINT
C
      INCLUDE 'STBCM.INC'
C
C       THIS SUBROUTINE PRINTS OUT THE COMPONENT WEIGHTS AND
C       POSITIONS WHEN ICGPRT = 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,XBOD
      WRITE (6,710) WWING,XWING
      WRITE (6,715) WCAND,XCAN
```

```
      WRITE (6,720) WHT,XHT
      WRITE (6,725) WVT,XVT
      WRITE (6,730) WLGFRT,XLGFRT
      WRITE (6,732) WLGR,XLGR
      WRITE (6,735) WNA,XNA
      WRITE (6,740) WPIV,XPIV
      WRITE (6,745) WAIRC,XLEWG
      WRITE (6,750) WAPU,XAPU
      WRITE (6,760) WELT,XELT
      WRITE (6,765) WEP,XEP
      WRITE (6,770) WINST,XINST
      WRITE (6,775) WHDP,XHDP
      WRITE (6,780) WSC,XSC
      WRITE (6,785) WPA,XPA
      WRITE (6,787) WFUR,XFUR
      DO 795 J = 1,EN
      WRITE (6,805) WZENG,XENG(J)
795   CONTINUE
      WRITE (6,810) WFS,XFS
      WRITE (6,815) WLFTF,XLIFTF
      WRITE (6,820) WPAYL,XPAYL
      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 (6,865) WMISS,XMISS
      WRITE (6,880) WFFUS,XFUSF
      WRITE (6,890) WFWG,XWGFUEL
      WRITE (6,870)
      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(5X,12H COMPONENT  ,12H   WEIGHT   ,12H  POSITION  )
 705  FORMAT(5X,12H  AIRFRAME  ,F12.6,F12.6)
 710  FORMAT(5X,12H    WING    ,F12.6,F12.6)
 715  FORMAT(5X,12H   CANARD   ,F12.6,F12.6)
 720  FORMAT(5X,12H     HT     ,F12.6,F12.6)
 725  FORMAT(5X,12H     VT     ,F12.6,F12.6)
 730  FORMAT(5X,12H  NOSE GEAR ,F12.6,F12.6)
 732  FORMAT(5X,12H  MAIN GEAR ,F12.6,F12.6)
 735  FORMAT(5X,12H   NACELLS  ,F12.6,F12.6)
 740  FORMAT(5X,12H   PIVOTS   ,F12.6,F12.6)
 745  FORMAT(5X,12H  AIR COND. ,F12.6,F12.6)
 750  FORMAT(5X,12H     APU    ,F12.6,F12.6)
 760  FORMAT(5X,12H   AVIONICS ,F12.6,F12.6)
 765  FORMAT(5X,12H  ELECTRICAL,F12.6,F12.6)
 770  FORMAT(5X,12H INSTURMENTS,F12.6,F12.6)
 775  FORMAT(5X,12H HYDRAULICS ,F12.6,F12.6)
 780  FORMAT(5X,12HCONTROL SURF,F12.6,F12.6)
 785  FORMAT(5X,12H     PA     ,F12.6,F12.6)
 787  FORMAT(5X,12H FURNISHINGS,F12.6,F12.6)
 805  FORMAT(5X,12H   ENGINES  ,F12.6,F12.6)
 810  FORMAT(5X,12H  FUEL SYS. ,F12.6,F12.6)
 815  FORMAT(5X,12H  LIFT FAN  ,F12.6,F12.6)
 820  FORMAT(5X,12H  PAYLOAD   ,F12.6,F12.6)
 830  FORMAT(5X,12H    CREW    ,F12.6,F12.6)
 835  FORMAT(5X,12H    CARGO   ,F12.6,F12.6)
 840  FORMAT(5X,12H AMMUNITION ,F12.6,F12.6)
 855  FORMAT(5X,12H     BB2    ,F12.6,F12.6)
 860  FORMAT(5X,12H    BOMBS   ,F12.6,F12.6)
 865  FORMAT(5X,12H  MISSILES  ,F12.6,F12.6)
 870  FORMAT(5X,25HEXTERNAL TANK(S) AND FUEL)
 880  FORMAT(5X,12H  FUS. FUEL ,F12.3,F12.6)
 890  FORMAT(5X,12H  WING FUEL ,F12.3,F12.6)
1050  FORMAT(5X,20HCENTERLINE FUEL TANK)
1051  FORMAT(5X,15HEXT. TANK POS.=,F12.6)
1052  FORMAT(5X,8HW. TANK=,F12.6,5X,8HW. FUEL=,F12.6)
1100  FORMAT(5X,19HTWO FUS. SIDE TANKS)
1101  FORMAT(5X,15HEXT. TANK POS.=,F12.6)
1102  FORMAT(5X,8HW. TANK=,F12.6,5X,8HW. FUEL=,F12.6)
1200  FORMAT(5X,18HTWO C/4-WING TANKS)
1201  FORMAT(5X,15HEXT. TANK POS.=,F12.6)
1202  FORMAT(5X,8HW. TANK=,F12.6,5X,8HW. FUEL=,F12.6)
1300  FORMAT(5X,17HTWO WINGTIP TANKS)
```

```
 1301 FORMAT(5X,15HEXT. TANK POS.=,F12.6)
 1302 FORMAT(5X,8HW. TANK=,F12.6,5X,8HW. FUEL=,F12.6)
 1400 FORMAT(5X,36HONE C.L. TANK AND TWO C/4-WING TANKS)
 1401 FORMAT(5X,14HC.L TANK POS.=,F12.6,5X,19HC/4-WING TANK POS.=,F12.6)
 1402 FORMAT(5X,13HW. FUS. FUEL=,F12.6,5X,13HW. WING FUEL=,F12.6)
 1500 FORMAT(5X,35HONE C.L. TANK AND TWO WINGTIP TANKS)
 1501 FORMAT(5X,15HC.L. TANK POS.=,F12.6,5X,18HWINGTIP TANK POS.=,F12.6)
 1502 FORMAT(5X,13HW. FUS. FUEL=,F12.6,5X,13HW. WING FUEL=,F12.6)
 1600 FORMAT(5X,37HTWO C/4-WING TANKS, TWO WINGTIP TANKS)
 1601 FORMAT(5X,19HC/4-WING TANK POS.=,F12.6,5X,
     $18HWINGTIP TANK POS.=,F12.6)
 1602 FORMAT(5X,13HW. WING FUEL=,F12.6,5X,16HW. WINGTIP FUEL=,F12.6)
 1700 FORMAT(//)
      RETURN
      END
C
C    ****************************************************************
C
      SUBROUTINE STBDT(ICALC,NERR,IGEO,KGPRNT,IGPLT)
C
C     THIS SUBROUTINE IS USED TO TRANSFER DATA BETWEEN THE STABILITY
C     MODULE AND THE AERODYNAMICS MODULE.  IT IS IMPORTANT TO MATCH
C     THE GLOBAL ARRAY SIZES BETWEEN THIS SUBROUTINE AND THE MODULE
C     'DATATR.DAT'.  NOTE THAT FOR EACH COMMON BLOCK OF THE MODULES
C     THERE EXISTS ONE EXTRA VARIABLE
C     THEN AS CALLED OUT IN DATATR.DAT.  THIS ALLOWS FOR A DUMMY
C     VARIABLE TO BE INCLUDED.
C
      COMMON/GLOBCM/RARAY(1400),IARAY(100)
      COMMON/ACSNT2/INOUT(1600),ILOCAT(30,4)
C    STABILITY AND CONTROL COMMON -MODULE 5.
      COMMON /STBCM/ RA5(134),IA5(2)
C    AERODYNAMICS COMMON - MODULE 3.
      COMMON /AEROCM/ RA3(170),IA3(10)
C
C    DATA TRANSFER INTERFACE BETWEEN STABILITY AND AERODYNAMICS.
C    OPERATION SEQUENCE.
C    1 - TRANSFER DATA FROM STABILITY LOCAL COMMON TO GLOBAL COMMON.
C    2 - TRANSFER DATA FROM GLOBAL TO AERODYNAMICS LOCAL COMMON.
C    3 - CALL AERODYNAMICS.
C    4 - TRANSFER DATA FROM AERODYNAMICS LOCAL COMMON TO GLOBAL.
C    5 - TRANSFER DATA FROM GLOBAL COMMON TO STABILITY LOCAL COMMON.
C
      JCALC=1
      KCALC=2
      NERR = 0
C
C    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,RARAY,IARAY)
```

```
C
C     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(NSTRTI),
      * RA3,IA3,RARAY,IARAY)
C
C     STEP 3.
       CALL AERO2(ICALC,NERR,IGEO,KGPRNT,IGPLT)
C
C     STEP 4.
       CALL DATAIO(KCALC,NMAXR,NMAXI,INOUT(NSTRTR),INOUT(NSTRTI),
      * RA3,IA3,RARAY,IARAY)
C
C     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(NSTRTI),
      * RA5,IA5,RARAY,IARAY)
C
C ********************************************************************
C *                                                                  *
C *   THE FOLLOWING IS A LIST OF THE REFERENCES                      *
C *     USED IN THIS SUBROUTINE:                                     *
C *                                                                  *
C *  1)  HOAK, D.E. et al.;  "USAF Stability and Control DATCOM";    *
C *      Wright Patterson AFB, Ohio, 45433;  Revised 1970            *
C *                                                                  *
C *  2)  Roskam, Jan; "Airplane Flight Dynamics and Automatic        *
C *      Controls, Part 1";  Published by the author, 519 Boulder    *
C *      Ave. Lawrence KA  66044;  second printing 1979              *
C *                                                                  *
C *  3)  McCormick, B.W.; "Aerodynamics, Aeronautics, and Flight     *
C *      Dynamics";  J. Wiley and Sons, 1979                         *
C *                                                                  *
C *  4)  Perkins C.D. and Hage R.E.; "Airplane Performance, Stability*
C *      and Control"; J. Wiley and Sons, 1949                       *
C *                                                                  *
C *  5)  Curry, Norman; "Aircraft Landing Gear Design, Principles    *
C *      and Practices"; AIAA Education Series; American Institute   *
C *      of Aeronautics and Astronautics;  1988                      *
C *                                                                  *
C ********************************************************************
       RETURN
       END
```

```
C     THIS CONTAINS THE COMMON BLOCK STATEMENTS NEEDED TO RUN THE STABILITY
C      AND CONTROL MODULE 'STBLCON.FOR'.  THIS FILE IS CONNECTED WITH THE
C      MODULE THROUGH THE 'INCLUDE' STATEMENT FOUND AT THE BEGINING OF EACH
C      OF THE SUBROUTINES.
C
        COMMON/OVER/ICALC,NERROR,MODNO,IGEO,KGPRNT,IGPLT,IF
C
        COMMON/STBCM/CLALFA,CLAFHT,DEDA,XQCBCN,XQCBWG,XQCBHT,XQCBVT,
     1  CBARCN,CBARWG,CBARHT,CBARVT,ROOTWG,TRWG,XLEWG,SPANWG,
     2  SWPWG,PODL,EN,BODL,XLEPOD(10),WCAND,WFTOT,FUFRAC,WWING,WFS,
     3  WLG,WHT,WVT,WETANK,WFEXT,WBODY,WAIRC,WAPU,WSC,WENG,SCN,SWG,SHT,
     4  SVT,WCREW,WELT,WEP,WHDP,WINST,WNA,WPA,WCARGO,WAMMUN,WBOMB,WMISS,
     5  ARWG,DIHED,ZRTWG,ZRTHT,ARVT,SWPCN,TRCN,AMTO,
     6  ZRTCN,VOLB,BDMAX,FRN,FRAB,ALPHA,CL,THRUST,ARHT,TNT(12),WFT(12),
     7  WFTO,WTOT,ARCN,CMWB,WGTO,WFEQ,WBB2,WAF,WE,WPL,WPS,WTSUM,WFUEL,
     8  TWTO,MACH,ALT,SWETWG,SWETHT,SWETCN,SPANHT,SPANCN,
     9  STARTM(12),DUMMY,IAO
C
        COMMON/LOCALS/ETAHT,XFCAN,XFHT,XFNA,XFVT,XFWG,XFAPU,XFENG,
     1  XFELT,XFEP,XFINST,XFHDP,XFPA,XFPAYL,XFFFUS,XFCREW,WPAYL,XFFUR,
     2  XFLGFRT,WFLGFRT,XFLGR,XFFS,XFBOMB,XFSC,XFPIV,XFLIFTF,WPIV,WFUR,
     2  XEF1,XFAMMUN,XFBB2,XFMISS,XFCARGO,XACWG,ETACN,ODEDA,XAWB,CLACN,
     3  WZET,WZEF,WEFF,WEFW,XEF2,WEFWT,WZENG,SUMMO,SUMWT,XLE,LWB,LTTWB,
     3  IDBPRT,XCG(13),DCMDCL(13),SM(13),XCGAFT(14),WLGFRT,WLGR,WFWG,
     4  WFP,WZB,WZM,WZA,IBS,IMS,IAS,NPHASE,ICGPRT,WLFTF,WFFUS,LHT,LCN,
     5  CELV,CBALHT,TZCHT,FNEUTPT(13),XFBOD,XFFWG,XFEX,XLETIP,IETANK,
     6  ZMTOR,AWTOT,ZMU,XT1,XT2,ZFMG,ZMG,ZFT1,ZT1,ZFT2,ZT2,ZFD,ZD,T1,
     7  TSPLIT,GAMAT1,GAMAT2,IHT,ICN,ELVDEF,VROT,ALPHACN,TAU,T2,
     8  SHT1,SCN1,ELVDMAX,IWG,XCGFRWD(13),QTOR,CGBAR(13),ALFTO,CLTO,
     8  CDTO,CMWBTO,ALFMAX,CDMAX,CMWBMAX,ALFO,CLO,CDO,CMWBO,XLECN,
     9  T1RAD,T2RAD,XLECG,XLET1,XLET2,CLMAX,CLAHTGRD,CLACNGRD,IVECT,
     1  SPANCN1,ZRTCN1,ARCN1,TRCN1,ALFHT,ALFCN,DCMTMUL,TWZTO,IFLAG
C
      COMMON/XCGS/XBOD,XWING,XCAN,XHT,XVT,XNA,XPIV,XAIRC,XAPU,
     1  XELT,XEP,XINST,XHDP,XSC,XPA,XENG(10),XFS,XLIFTF,XFUR,
     2  XPAYL,XCREW,XCARGO,XAMMUN,XBB2,XBOMB,XMISS,XETANK,
     3  XFUSF,XWGFUEL,XLGFRT,XLGR,LGFLAG
C
        REAL XACWG,DCMDCLF,DCMDCLT,DCMDCLW,DCMCLCN,KF,CSC,TC,MBOD,
     1  MCAN,MWG,MFS,MLG,MHT,MVT,METANK,MFEXT,MAIRC,MAPU,MSC,MENG(10),
     1  LWB,LH1,LH2,LH3,LH4,LH5,LH6,LH7,LH8,LH9,LC1,LC2,LC3,LC4,LC5,
     2  LC6,LC7,LC8,LHT,LCN,ZMTOR,ZMU,LTTWB,IWG,IHT,ICN,LEFF,MACH,
     2  MCREW,MELT,MEP,MHDP,MINST,MNA,MPA,MCARGO,MAMMUN,MBOMB,MMISS,
     3  MPAYL,MPIV,MLIFTF,K11,K12,K13,K14,K21,K22,K23,K24
```

1 0 115 GEOMETRY REAL VARIABLE CATALOG

						100059	100045	100208	100210
-61	-62	-63	-64	100060	100046				
209	100430	100428	100429	100427	100434	100432	100433	100431	100514
100512	100513	100511	-603	-604	-605	-606	100213	243	470
211	182	196	197	282	169	100426	100424	100425	100423
529	100069	73	71	72	70	370	368	369	367
-307	-308	414	412	413	411	612	609	611	608
620	619	663	664	634	100066	100607	100635	100653	-189
-610	193	194	195	661	184	185	186	662	416
-419	421	418	420	417	100422	100388	100415	100371	525
187	568	-567	665	666	667	668	669	670	671
672	673	674	675	676	677	678	679	680	681
682	683	684	-570	685					

2 0 392 TRAJECTORY REAL VARIABLE CATALOG

100013	-60	-74	-108	-109	100144	100188	192	-196	216
-241	-242	244	100338	100351	-373	-422	-472	-516	100519
532	-535	-548	100567	-570	-576	-583	-579	100245	100246
100247	100248	100249	100250	100251	100252	100253	100254	100255	100256
100229	100230	100231	100232	100233	100234	100235	100236	100237	100238
100239	100240	100473	100474	100475	100476	100477	100478	100479	100480
100481	100482	100483	100484	324	325	326	327	328	329
330	331	332	333	334	335	447	448	449	450
451	452	453	454	455	456	457	458	269	270
271	272	273	274	275	276	277	278	279	280
312	313	314	315	316	317	318	319	320	321
322	323	435	436	437	438	439	440	441	442
443	444	445	446	257	258	259	260	261	262
263	264	265	266	267	268	-540	100551	-387	-471
-85	-311	100212	183	100207	100588	100589	100590	100591	100592
100593	100594	100595	100596	100597	100598	100599	100206	100469	67
68	497	498	565	283	284	285	286	287	288
289	290	291	292	293	294	295	296	297	298
299	300	301	302	303	304	305	306	399	400
401	402	403	404	405	406	407	408	409	410
217	218	219	220	221	222	223	224	225	226
227	228	553	554	555	556	557	558	559	560
561	562	563	564	485	486	487	488	489	490
491	492	493	494	495	496	621	622	623	624
625	626	627	628	629	630	631	632	157	158
159	160	161	162	163	164	165	166	167	168
120	121	122	123	124	125	126	127	128	129
130	131	14	15	16	17	18	19	20	21
22	23	24	25	145	146	147	148	149	150
151	152	153	154	155	156	96	97	98	99
100	101	102	103	104	105	106	107	26	27
28	29	30	31	32	33	34	35	36	37
569	574	339	340	341	342	343	344	345	346
347	348	349	350	354	355	356	357	358	359
360	361	362	363	364	365	375	376	377	378
379	380	381	382	383	384	385	386	499	500
501	502	503	504	505	506	507	508	509	510
170	171	172	173	174	175	176	177	178	179
180	181	-686	-687	100688	-689	-690	-691	-692	-693
100566	694								

3	0	169	AERODYNAMIC VARIABLES (REAL)						
100013	-216	-244	-60	-46	-45	-426	-424	-425	-423
-416	-421	-418	-420	-417	-69	-73	-71	-72	-70
-66	-210	-208	-370	-414	-412	-413	-411	-514	-422
-388	-415	-371	-430	-434	-428	-432	-429	-433	-427
-431	-532	-472	-188	-529	-619	-634	-196	-470	-243
-620	-512	-513	-511	-368	-369	-367	-307	-308	-419
-61	-62	-63	-64	-603	-604	-605	-606	-197	100203
100201	100205	100204	100202	75	76	77	78	79	80
81	82	83	84	86	87	88	89	90	91
92	93	94	95	132	133	134	135	136	137
138	139	140	141	100144	100109	100108	74	242	241
389	390	391	392	393	394	395	396	397	398
110	111	112	113	114	115	116	117	118	119
695	696	697	698	699	700	701	702	703	704
705	706	707	708	-665	-666	-667	-668	-669	-612
-609	-611	-608	-670	-671	-672	-673	-674	-675	-676
-677	-678	-679	-664	-680	-681	-682	-683	-684	
4	0	65	PROPULSION VARIABLES (REAL)						
100516	-570	-192	-196	-244	-216	610	546	472	373
100065	100517	100198	100200	100337	583	100310	100199	100522	100336
-422	387	471	85	311	53	54	55	56	57
58	47	48	49	50	51	52	613	614	615
616	617	618	39	40	41	42	43	44	100189
374	515	308	307	419	524	603	604	605	606
61	62	63	64	100709					
5	0	133	STABILITY AND CONTROL REAL CATALOG						
-705	-706	-707	-664	-620	-619	-663	-70	-73	-71
-72	-370	-514	-612	-414	-426	-307	-196	-69	-670
-671	-672	-673	-674	-675	-676	-677	-678	-679	-541
-566	-211	-587	-552	-575	-572	-586	-548	-551	-539
-534	-536	-582	-546	-371	-422	-388	-415	-543	-545
-547	-571	-573	-577	-578	-542	-535	-540	-576	-60
-665	-666	-667	-59	-423	-511	-690	-669	-525	-66
-210	-208	100013	100144	-472	-46	-499	-500	-501	-502
-503	-504	-505	-506	-507	-508	-509	-510	-553	-554
-555	-556	-557	-558	-559	-560	-561	-562	-563	-564
-565	-583	-45	-708	-570	-550	-537	-533	-544	-579
-580	-585	-567	-516	100244	10038	-421	-418	-417	-412
-411	-399	-400	-401	-402	-403	-404	-405	-406	-407
-408	-409	-410							
6	0	78	WEIGHTS VARIABLES (REAL)						
-45	-46	-59	-60	-66	-69	-182	-196	-197	[illegible]
-338	-415	-422	-416	-417	-418	-420	-423	-424	-425
-426	-427	-428	-429	-430	-514	-519	-520	537	-546
-567	100570	100460	100461	100462	100463	100464	100465	100466	100467
100468	100459	549	-551	-434	-211	581	100533	100534	100536
100539	100543	100544	100545	100547	100552	100571	100572	100573	100575
100577	100578	100579	100580	100582	100585	100586	100587	100535	100540
100541	100542	100548	100576	100550	-566	-412	689		
7	0	45	CARGO REAL CATALOG						
537	607	635	653	142	143	633	602	100657	100656
100652	100650	100651	100655	100658	100659	100660	100637	100638	100639
100640	100641	100642	100643	100644	100645	100646	100647	100648	100649

```
100190 100309 100654 100636 538 100001 100002 100003 100004 100005
100006 100007 100008 100009 100010
9 0 39 ECONOMICS REAL CATALOG
-182 -196 -282 -516 -533 -534 -536 -539 -543 -544
-545 -547 0 -552 -567 -578 -570 -587 -571 -572
-573 -575 -577 -579 -580 -582 -586 -183 -68 -351
0 100012 100191 100366 100518 100352 0 -569 -574
10 0 45 NAVY REAL CATALOG
11 -66 -74 -13 -196 214 215 281 100353 -374
-413 -422 -516 520 521 523 526 527 528 530
531 -538 -540 -543 -553 -554 -555 -556 -557 -558
-559 -560 -561 -562 -563 -564 -567 -570 -579 584
600 601 244 38 -144
11 0 418 SUMMARY OUTPUT CATALOG (REAL)
-585 -69 -422 -388 -371 -415 -581 -66 -421 -418
-417 -420 -516 -525 -414 -412 -411 -413 -519 -416
-182 -426 -424 -423 -425 -282 -60 -46 -45 -59
-514 -512 -511 -513 -430 -428 -427 -429 -434 -432
-431 -433 -196 -370 -368 -367 -369 -610 -533 -189
-580 -73 -71 -70 -72 -546 -550 -612 -609 -608
-611 -515 -566 -374 -579 -497 -498 -565 -207 -68
-67 -206 -245 -246 -247 -248 -249 -250 -251 -252
-253 -254 -255 -256 -229 -230 -231 -232 -233 -234
-235 -236 -237 -238 -239 -240 -473 -474 -475 -476
-477 -478 -479 -480 -481 -482 -483 -484 -324 -325
-326 -327 -328 -329 -330 -331 -332 -333 -334 -335
-447 -448 -449 -450 -451 -452 -453 -454 -455 -456
-457 -458 -269 -270 -271 -272 -273 -274 -275 -276
-277 -278 -279 -280 -312 -313 -314 -315 -316 -317
-318 -319 -320 -321 -322 -323 -435 -436 -437 -438
-439 -440 -441 -442 -443 -444 -445 -446 -257 -258
-259 -260 -261 -262 -263 -264 -265 -266 -267 -268
-588 -589 -590 -591 -592 -593 -594 -595 -596 -597
-598 -599 -283 -284 -285 -286 -287 -288 -289 -290
-291 -292 -293 -294 -295 -296 -297 -298 -299 -300
-301 -302 -303 -304 -305 -306 -399 -400 -401 -402
-403 -404 -405 -406 -407 -408 -409 -410 -217 -218
-219 -220 -221 -222 -223 -224 -225 -226 -227 -228
-553 -554 -555 -556 -557 -558 -559 -560 -561 -562
-563 -564 -485 -486 -487 -488 -489 -490 -491 -492
-493 -494 -495 -496 -621 -622 -623 -624 -625 -626
-627 -628 -629 -630 -631 -632 -157 -158 -159 -160
-161 -162 -163 -164 -165 -166 -167 -168 -120 -121
-122 -123 -124 -125 -126 -127 -128 -129 -130 -131
-14 -15 -16 -17 -18 -19 -20 -21 -22 [illegible]
-24 -25 -145 -146 -147 -148 -149 -150 -151 -152
-153 -154 -155 -156 -96 -97 -98 -99 -100 -101
-102 -103 -104 -105 -106 -107 -26 -27 -28 -29
-30 -31 -32 -33 -34 -35 -36 -37 -339 -340
-341 -342 -343 -344 -345 -346 -347 -348 -349 -350
-354 -355 -356 -357 -358 -359 -360 -361 -362 -363
-364 -365 -375 -376 -377 -378 -379 -380 -381 -382
-383 -384 -385 -386 -499 -500 -501 -502 -503 -504
-505 -506 -507 -508 -509 -510 -170 -171 -172 -173
```

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  -174     -175     -176     -177     -178     -179     -180     -181        0        0
     0        0      -12        0        0        0     -690     -691
 14    0   65      TAKEOFF AND LANDING MODULE (REAL VARIABLES)
  -570     -422   100718   100719   100720   100721   100722   100723   100724   100725
100726   100727   100728   100729   100730   100731   100732   100733   100734   100735
100736   100737   100738   100739   100740   100741   100742      743      686      744
   745      746   100747   100748   100749   100750   100751     -144      -74     -109
  -472     -373      216      244       13      192     -196   100752   100753   100754
100755   100756   100757   100758   100759      687      690      691      692      693
100760   100761   100762     -471     -497
  1    1    4       GEOMETRY VARIABLES (INTEGER)
    16       18       19       20
  2    1    7       TRAJECTORY VARIABLES (INTEGER)
     6        8        9       10        7       14       17
  3    1    9       AERODYNAMIC VARIABLES (INTEGER)
    -6      -10       -9       -7       13      -16       18       19       20
  4    1    4       PROPULSION VARIABLES (INTEGER)
    -8   100011   100012       15
  5    1    1       STABILITY AND CONTROL VARIABLES (INTEGER)
     6
  7    1    5       CARGO INTEGER VARIABLES
100001   100002   100003   100004   100005
 10    1    2       NAVY INTEGER CATALOG
     6      -14
 11    1    1      SUMMARY OUTPUT CATALOG (INTEGER)
   -14
 14    1    3      TAKEOFF AND LANDING MODULE (INTEGER VARIABLES)
     6        8      -17
  0
```