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# FINAL REPORT

## V/STOL TILT ROTOR STUDY - VOLUME V

### A MATHEMATICAL MODEL FOR REAL TIME FLIGHT SIMULATION OF THE BELL MODEL 301 TILT ROTOR RESEARCH AIRCRAFT

MODEL 301

REPORT 301-099-001

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V/STOL TILT ROTOR STUDY-VOLUME V  
A MATHEMATICAL MODEL FOR REAL TIME FLIGHT SIMULATION  
OF  
THE BELL MODEL 301 TILT ROTOR RESEARCH AIRCRAFT

by

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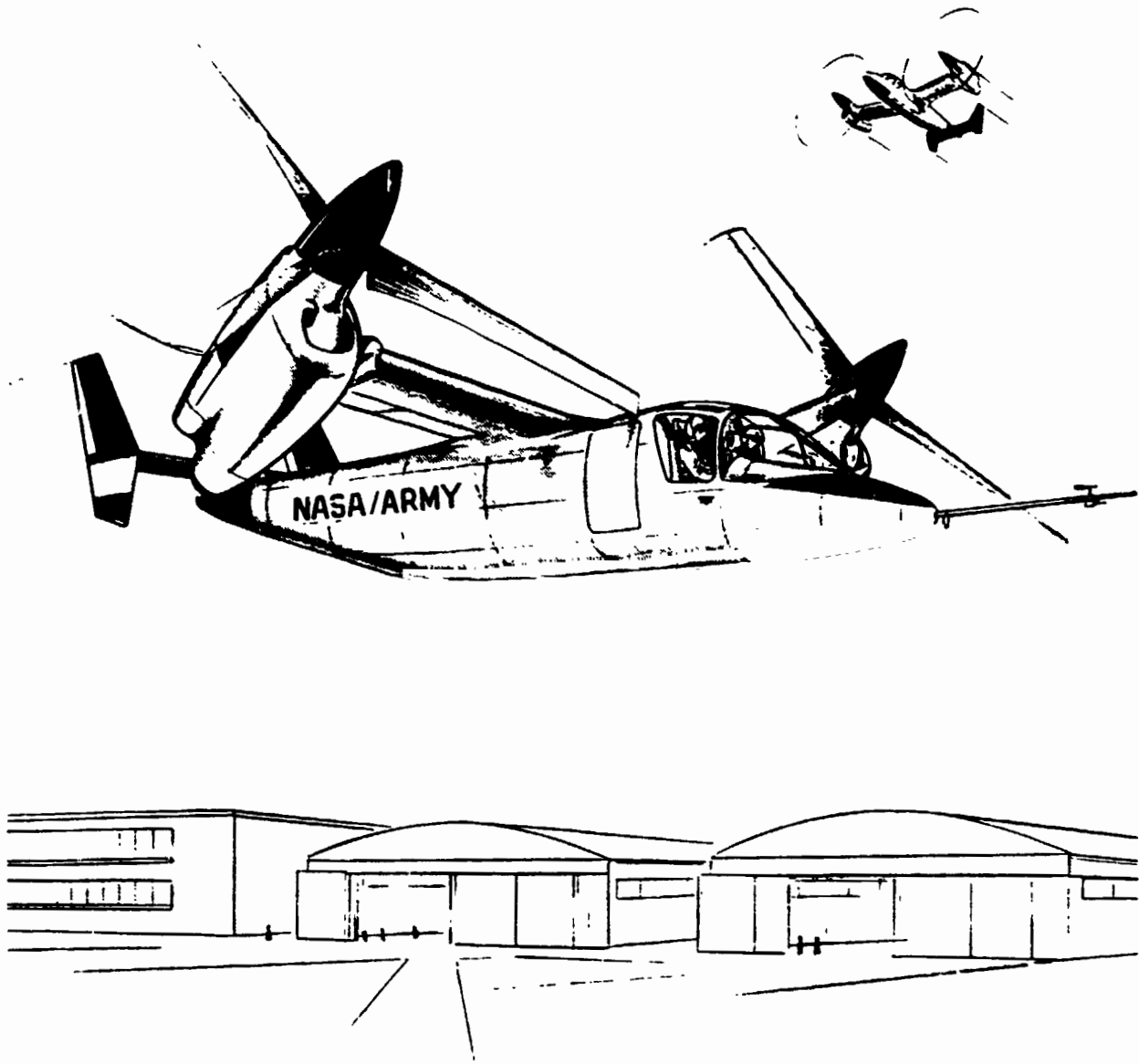
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### Foreword

This report is one of a series prepared by the Bell Helicopter Company, Fort Worth, Texas for the National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California under contract NAS2-6599. These Tilt Rotor Research Aircraft studies were jointly funded by NASA and the U. S. Army Air Mobility Research and Development Laboratory, Ames Directorate.

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- Volume I -- V/STOL Tilt Rotor Study, Conceptual Design --  
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BELL MODEL 301 TILT ROTOR  
RESEARCH AIRCRAFT



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LIST OF SYMBOLS

This section describes the symbols used for the time dependent variables that are passed between the subsystems (see subsystems block diagram, Section I).

Key to Nomenclature:

X, Y, Z	Indicate body axes forces
D, Y', L	Indicate wind axes forces
l, M, N	Indicate body axes moments
l', M', N'	Indicate wind axes moments

Subscripts

F	Fuselage
W	Wing
WP	Wing-pylon
H	Horizontal stabilizer
V	Vertical stabilizer
R	Right rotor
L	Left rotor
SP	Shaft pivot point
P	Pylon
MG	Main gear
NG	Nose gear
a	Aileron
e	Elevator
r	Rudder
PA	Pilot station
CG	Aircraft center of gravity

R/W	Due to rotor at wing
R/H	Due to rotor at horizontal stabilizer
R/V	Due to rotor at vertical stabilizers
W/H	Due to wing at horizontal stabilizer

$\left  \begin{array}{c} B \\ R/W \end{array} \right.$	Due to rotor at wing in body axes
--	-----------------------------------

$\left  \begin{array}{c} B \\ R/H \end{array} \right.$	Due to rotor at horizontal stabilizer, in body axes
--	---

$\left  \begin{array}{c} B \\ R/V \end{array} \right.$	Due to rotor at vertical stabilizers, in body axes
--	--

Symbol	Description	Units
$a_0, a_1, a_2$	Blade lift coefficients	$\frac{1}{ad}, \frac{1}{\mu}, \frac{1}{r^2}$

Symbol	Description	Units
$a_{OL}$	Steady flapping component, left rotor	rad
$a_{OR}$	Steady flapping component, right rotor	rad
$a_{1L}$	F/A flapping, left rotor + backward	rad
$\dot{a}_{1L}$	F/A flapping rate, left rotor	rad/sec
$a_{1R}$	F/A flapping, right rotor + backward	rad
$\dot{a}_{1R}$	F/A flapping rate, right rotor	rad/sec
B	Blade tip loss factor	ND
$BL_{CG}$	Butt line of aircraft C.G.	In.
$B_{1L}$	Forward cyclic input, left rotor	rad
$B_{1R}$	Forward cyclic input, right rotor	rad
$b_{1L}$	Lateral flapping, left rotor, + outboard edge down	rad
$\dot{b}_{1L}$	Lateral flapping rate, left rotor	rad/sec
$b_{1R}$	Lateral flapping, right rotor, + outboard edge down	rad
$\dot{b}_{1R}$	Lateral flapping rate, right rotor	rad/sec
$C_{\theta}$	Prandtl-Glauert factor, page A-33	
$C_{RF}$	Rotor thrust coefficient (see page A-18)	ND
$C_{TL}$	Thrust coefficient, left rotor	ND
$C_{TR}$	Thrust coefficient, right rotor	ND
$D_F$	Aero drag on fuselage	Lb.
$D_H$	Aero drag on horizontal stabilizer	lb.
$D_{iWPL}$	Aero drag on portion of left wing-pylon immersed in rotor wake	Lb.
$D_{iWPL}$	Aero drag on portion of right wing-pylon immersed in rotor wake	Lb.
$D_{MG}$	Aero drag on main landing gear	Lb.
$D_{NG}$	Aero drag on nose landing gear	Lb.
$D_{VL}$	Aero drag on left vertical fin	Lb.
$D_{VR}$	Aero drag on right vertical fin	Lb.
$D_{WP}$	Aero drag on portion of wing-pylon in freestream	Lb.
E	Distance from takeoff point in the direction grid-East (+ East)	Nautical Miles
$\dot{E}$	Eastward velocity of aircraft	Knots
ESAS, RSAS, ASAS	SCAS actuator inputs to the elevator, rudder, and aileron, respectively	In.

Symbol	Description	Units
$F_{LN}, F_T, F_{PD}$	Stick and pedal forces from pilot (+ fwd, + right. + right)	Lb.
$F_X$	Flap position indicator	-
$H_L$	Aero H-force on left rotor + backward (upward) in mast axis system	Lb.
$HP_{RL}$	HP required to drive left rotor	HP
$HP_{RR}$	HP required to drive right rotor	HP
$H_R$	Aero H force on right rotor + backward (upward) in mast axis system	Lb.
$h$	Radar altitude of aircraft	ft
$\dot{h}$	Aircraft climb rate	ft/sec
$I_{XX}$	Aircraft rolling inertia, body axes	slug ft <sup>2</sup>
$I_{XZ}$	Aircraft product of inertia, body axes	slug ft <sup>2</sup>
$I_{YY}$	Aircraft pitching inertia, body axes	slug ft <sup>2</sup>
$I_{ZZ}$	Aircraft yawing inertia, body axes	slug ft <sup>2</sup>
$K_e$	Elevator effectiveness factor, p. B-26	
$L_F$	Aero lift on fuselage	Lb.
$L_H$	Aero lift on horizontal stabilizer	Lb.
$L_{iWPL}$	Aero lift on portion of left wing pylon immersed in rotor wake	Lb.
$L_{iWPR}$	Aero lift on portion of right wing-pylon immersed in rotor wake	Lb.
$L_{LG}$	Landing gear position indicator	ND
$L_{WP}$	Aero lift on portion of wing-pylon in freestream	Lb.
$l_A, m_A, n_A$	Total rolling, pitching and yawing moment on aircraft about body X, Y, Z axes	Lb.
$l_F, m_F, n_F$	Rolling, pitching and yawing moments on fuselage about body X, Y, Z axes	Lb Ft
$l_L, m_L, n_L$	Rolling, pitching and yawing moments about body X,Y,Z axes, due to left rotor forces	Lb Ft
$l_R, m_R, n_R$	Rolling, pitching and yawing moments about X,Y,Z axes, due to right rotor forces	Lb Ft
$l_{WP}, m_{WP}, n_{WP}$	Rolling, pitching and yawing moments due to wing-pylon, about body X,Y,Z axes	Lb Ft

Symbol	Description	Units
$L'_{WP}, M'_{WP}, N'_{WP}$	Rolling, pitching and yawing moments due to wing-pylon in wind axes system	Lt ft
$L_{XH}, L_{YH}, L_{ZH}$	Distance from the CG to swashplate, Body X, Y, Z axes	ft
$M_H$	Aero pitching moment due to horizontal stabilizer in body axes	Lb ft
$M'_H$	Aero pitching moment due to horizontal stabilizer in wind axes	Lb ft
$M_N$	Mach number	ND
$M_{alL}$	F/A flapping restraint moment exerted by left rotor on airframe, mast axes system (nose up)	Lb ft
$M_{alR}$	F/A flapping restraint moment exerted by right rotor on airframe, mast axes system (nose up)	Lb ft
$l_{blL}$	Lateral flapping restraint moment exerted by left rotor on airframe, mast axes system (hub outboard)	Lb ft
$l_{blR}$	Lateral flapping restraint moment exerted by right rotor on airframe, mast axes system (hub outboard)	Lb ft
$N_N$	Distance from takeoff point in the direction grid-North (+ North)	Naut. Miles
$N_X, N_Y, N_Z$	Linear acceleration components of aircraft cg in body axes system	g
$P_{AX}, P_{AY}, P_{AZ}$	Position of aircraft cg w r t ground	Naut. Miles
$P_a$	Ambient pressure, absolute	psia
$p, q, r$	Aircraft roll, pitch and yaw rates in body axes system	rad/sec
$\dot{p}, \dot{q}, \dot{r}$	Aircraft angular accelerations in body axes system	rad/sec <sup>2</sup>
$Q_L$	Left rotor aero torque, + trying to slow rotor down, mast axes system	Lb ft
$Q_{LPT}$	Torque supplied by left power turbine	Lb ft
$Q_R$	Right rotor aero torque, + trying to slow rotor down, mast axes system	Lb ft
$Q_{RPT}$	Torque supplied by right power turbine	Lb ft
$R$	Rotor radius	ft
$RPM_{NII}$	Power Turbine RPM set at 100% (overspeed Governor)	RPM
$RPM_p$	Proprotor RPM selected by the pilot	RPM



Symbol	Description	Units
$R_{WL}$	Left rotor wake contraction ratio	ND
$SL_{CG}$	Station location of aircraft CG	In.
$T_L$	Left rotor aero. thrust, in mast axis + up (forward)	Lb.
$T_R$	Right rotor aero. thrust, in mast axis + up (forward)	Lb
$T_a$	Ambient temperature, absolute	$^{\circ}K$
$U, V, W$	Velocity of aircraft CG, w.r.t. air, in body axes system, in X, Y, Z direction	Ft/sec
$U_{EB}, V_{EB}, W_{EB}$	Velocity components of aircraft CG, w.r.t. air, along earth-based axes	Ft/sec
$U_G, V_G, W_G$	Ground velocity components of aircraft CG	Ft/sec
$U_W, V_W$	Wind velocity components w.r.t ground	Ft/sec
$U_i \begin{matrix}  B \\  R/H \end{matrix}$	Induced X-velocity at horizontal stabilizer in body axes system, due to rotors	Ft/sec
$U_i \begin{matrix}  B \\  R/V \end{matrix}$	Induced X-velocity at vertical fins, in body axes system, due to rotor	Ft/sec
$U_i \begin{matrix}  B \\  R/WL \end{matrix}$	Induced X-velocity at left wing, in body axes system, due to rotor	Ft/sec
$U_i \begin{matrix}  B \\  R/WR \end{matrix}$	Induced X-velocity at right wing, in body axes system, due to rotor	Ft/sec
$V_{CAS}$	Calibrated airspeed	MPH
$V_T$	Total linear velocity of aircraft c.g. w.r.t. air	Ft/sec
$WL_{CG}$	Waterline of aircraft CG	In.
$W_{iL}$	Uniform component of induced velocity at left rotor, + downward, mast axes system	Ft/sec
$W_{iR}$	Uniform component of induced velocity at right rotor, + downward, mast axes system	Ft/sec

Symbol	Description	Units
$W_i \begin{matrix}  B \\  R/H \end{matrix}$	Induced Z-velocity at horizontal stabilizer, in body axes system, due to rotors	Ft/sec
$W_i \begin{matrix}  B \\  R/V \end{matrix}$	Induced Z-velocity at vertical fins, in body axes system, due to rotors	Ft/sec
$W_i \begin{matrix}  B \\  R/WL \end{matrix}$	Induced Z-velocity at left wing, in body axes system, due to rotor	Ft/sec
$X_A, Y_A, Z_A$	Total forces on the aircraft in X,Y,Z directions, body axes system	Lb.
$X_{COL}$	Collective stick position, inches from full down	in.
$X_F, Y_F, Z_F$	Aero forces on the fuselage, in body axes system	Lb.
$X_{FL}$	Position indicator for flap actuator	ND
$X_H, Z_H$	Aero forces on horizontal stabilizer, in body axes system	Lb.
$X_{iWPL}, Z_{iWPL}$	Aero forces on portion of left wing-pylon in rotor wake, in body axes system	Lb.
$X_{iWPL}, Z_{iWPL}$	Aero forces on portion of right wing-pylon in rotor wake, in body axes system	Lb.
$X_L, Y_L, Z_L$	Left rotor forces in body axes system	Lb.
$X_{LG}$	Position indicator for landing gear actuator	ND
$X_{LN}$	Longitudinal stick position, inches from full aft	In.
$X_{LT}$	Lateral stick position, inches from full left	In.
$X_{MG}, Z_{MG}$	Aero forces on main landing gear, in body axes system	Lb.
$X_{NG}, Z_{NG}$	Aero forces on nose gear in body axes system	Lb.
$X_{PD}$	Position of pedal, inches from full left	In.
$X_R, Y_R, Z_R$	Right rotor forces in body axes system	Lb.

Symbol	Description	Units
$X_{VL}, Y_{VL}, Z_{VL}$	Aero forces on left fin, in body axes system	Lb.
$X_{VR}, Y_{VR}, Z_{VR}$	Aero forces on right fin, in body axes system	Lb.
$X_{THL}, X_{THR}$	Angles of the power levers on the fuel controls	Deg.
$X_{WP}, Y_{WP}, Z_{WP}$	Aero forces on wing-pylon in freestream, in body axes system	Lb.
$X_{PM}$	Position of mast tilt-actuator, percent	ND
$\alpha_F$	Fuselage angle of attack	rad
$\dot{\alpha}_F$	Rate of change of fuselage angle of attack	rad/sec
$\alpha'_{FWR}$	Angle of attack at portion of right wing immersed in rotor wake	rad
$\alpha'_{FL}$	Angle of attack at portion of left wing immersed in rotor wake	rad
$\alpha_H$	Angle of attack at horizontal stabilizer	rad
$\alpha_W$	Angle of attack on portion of wing outside rotor wake	rad
$\beta_F$	Fuselage sideship angle	rad
$\beta_M$	Mast conversion angle, + forward, measured from body (-Z) axis, $0^\circ$ is vertical, $90^\circ$ is zero incidence	deg
$\beta_V$	Angle of attack at vertical fins	deg
$\delta_3$	Pitch flap coupling	deg
$\delta_a$	Aileron mean deflection angle, + right aileron up	deg
$\delta_e$	Elevator deflection angle, + up	deg
$\delta_r$	Rudder deflection angle, + right pedal	deg
$\epsilon_{W/H}$	Wing wake deflection at horizontal stabilizer	deg
$\theta_{oL}$	Left rotor root collective pitch	deg
$\theta_{oL/G}$	Left rotor collective pitch input from proprotor collective governor	deg

Symbol	Description	Units
$\theta_{OR}$	Right rotor root collective pitch	deg
$\theta_{OR/G}$	Right rotor collective pitch input from propretor collective governor	deg
$\lambda_{oL}$	Inflow ratio at left rotor, neglecting rotor-induced velocity	ND
$\lambda_{oR}$	Inflow ratio at right rotor, neglecting rotor-induced velocity	ND
$\mu_L$	Tip speed ratio, left rotor	ND
$\mu_R$	Tip speed ratio, right rotor	ND
$\rho$	Air density	Slug/ft <sup>3</sup>
$\Psi, \theta, \phi$	Euler angles	rad
$\dot{\Psi}, \dot{\theta}, \dot{\phi}$	Rate of change of Euler angles	rad/sec
$\Psi_w, \theta_w$	Grid heading (+ clockwise from N) and Euler pitch angle of wind	rad
$\Omega_{EL}$	Left engine speed	rad/sec
$\Omega_{ER}$	Right engine speed	rad/sec
$\Omega_{INT}$	Interconnect shaft speed	rad/sec
$\Omega_L$	Left rotor speed w.r.t. aircraft	rad/sec
$\Omega_{LPT}$	Left power turbine speed, same as $\Omega_{EL}$	rad/sec
$\Omega_R$	Right rotor speed w.r.t. aircraft	rad/sec
$\Omega_{RPT}$	Right power turbine speed, same as $\Omega_{ER}$	rad/sec
$\phi_m$	Lateral mast tilt	deg
$V_n$	Distance from CG to gear	In
$\tau_e$	Elevator effectiveness, $\partial\alpha_H/\partial\delta_e$	ND
$\tau_r$	Rudder effectiveness, $\partial\alpha_V/\partial\delta_r$	ND

## I. SUMMARY

This report describes a mathematical model for real-time flight simulation of the Bell Model 301 Tilt Rotor Research Aircraft<sup>1</sup>. The mathematical model was developed under Contract NAS 2-6599 for mechanization of the NASA Ames Flight Simulator for Advanced Aircraft (FSAA)<sup>2</sup>. The objective of this program was to obtain a validated mathematical model of the tilt rotor research aircraft to support the aircraft design, pilot training, and proof-of-concept flight testing.

The development of the mathematical model involved the following specific tasks:

- (1) Derivation of equations representing the kinematic, dynamic, and aerodynamic characteristics of the Model 301's rotor, airframe, flight control system and subsystem. The mathematical model is described in Section II and the equations are presented in Appendix A. Model 301 parameters required as inputs to the equations are given in Appendix B.
- (2) Checkout and validation of the mathematical model. A non-realtime digital computer program was developed at Bell for this purpose. Comparison of performance and stability and control characteristics predicted using the subject mathematical model with experimental data and with predictions using other computer programs is presented in Section III.
- (3) Support of the Tilt Rotor Research Aircraft Project simulation effort at NASA Ames. This involved providing technical assistance during checkout and validation of the FSAA/Sigma 8 mechanization of the mathematical model, Program BELTR (Program BELTR was developed by Computer Science Corporation under a separate contract and was used for the real time simulation) and evaluation of the simulation by Bell Helicopter Company experimental test pilots. The results of the evaluation are given in Section IV and recommendations for further development of the mathematical model and the simulation are given in Section V.

The structure of the mathematical model is indicated by the block diagram shown in Figure I-1. The mathematical model differs from that for a conventional fixed-wing aircraft principally in the added requirements to represent the dynamics and aerodynamics of the rotors, the interaction of the rotor wake with the airframe, and the rotor control and drive systems (Subsystems 1, 2, 8(a) and 19 in Figure I-1). The portion of the block diagram enclosed by a dashed line is common to all simulation mathematical models; in the BELTR Program the equations given in this report were replaced by CSC program BASIC<sup>3</sup> which contains essentially identical equations and is used for all FSAA simulations. The landing gear oleo forces were also generated using the BASIC program.

The rigorousness of the mathematical model of the tilt-rotor research aircraft was constrained by two factors. One was the requirement to keep the fast loop computational time (this was a two loop simulation) to less than 50 milliseconds in order to maintain real time simulation. To achieve this it was necessary to limit the rotor representation to steady, linearized, aerodynamics with uniform inflow and to approximate the rotor following time. Rotor stall and compressibility effects are only used to define a limit for the maximum rotor thrust coefficient as a function of advance ratio. This rotor math model is satisfactory for most handling qualities studies, but may be inadequate to evaluate flight conditions or maneuvers where stall, compressibility or rotor dynamics are significant.

The second factor constraining the rigorousness of the mathematical model was the lack of sufficient experimental data on the rotor wake-airframe aerodynamic interactions such as the rotors' downwash (or upwash) at the horizontal tail. The model of the rotor wake-airframe interaction is presently based on a limited amount of data from the tests of a powered model of a tilt-rotor aircraft similar to the Model 301<sup>4</sup>. Tests of a powered model of the Model 301 to obtain detailed information on the rotor wake-airframe aerodynamic interaction have been completed and will be used to refine the model of this important characteristic of the tilt rotor VTOL.

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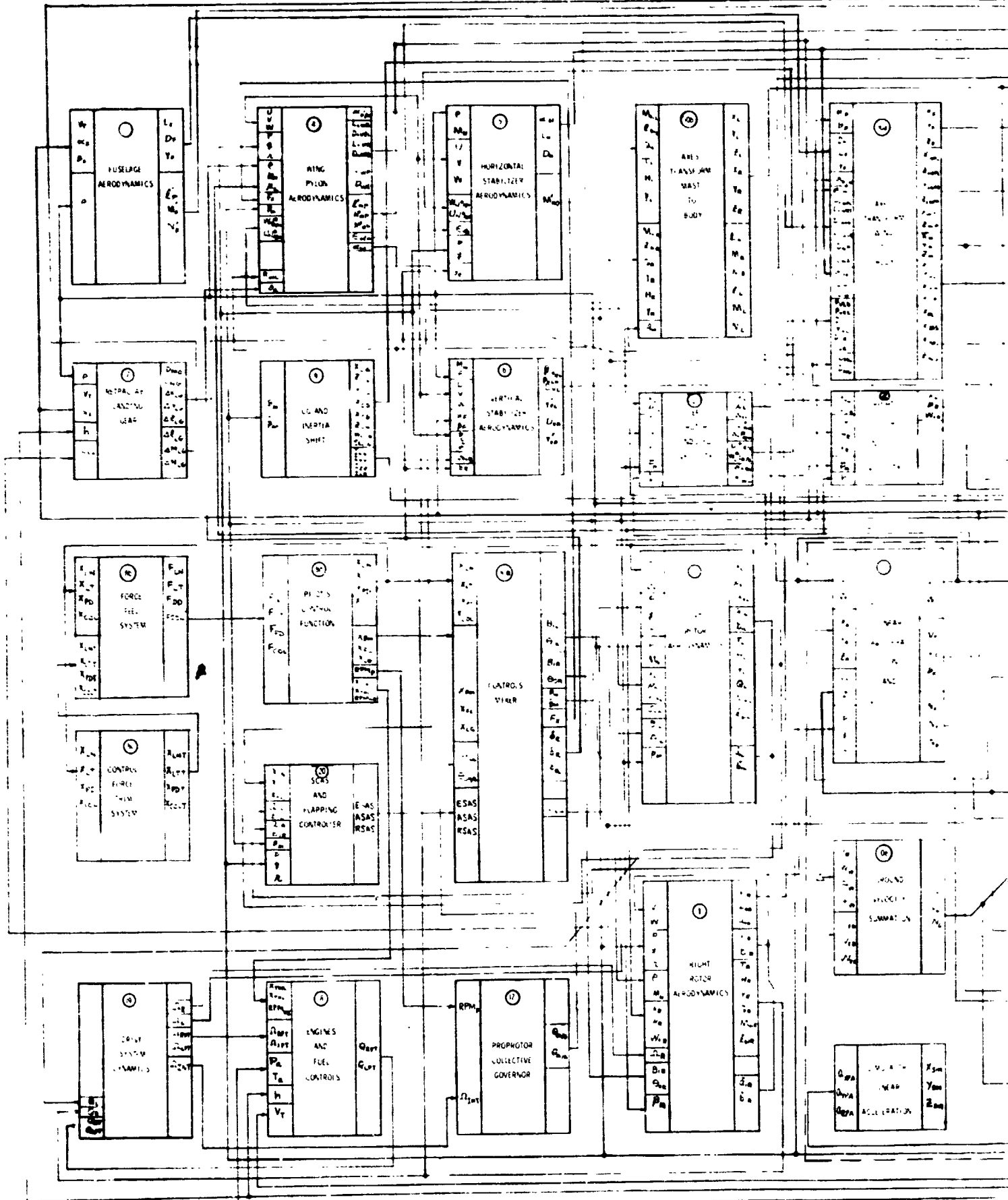
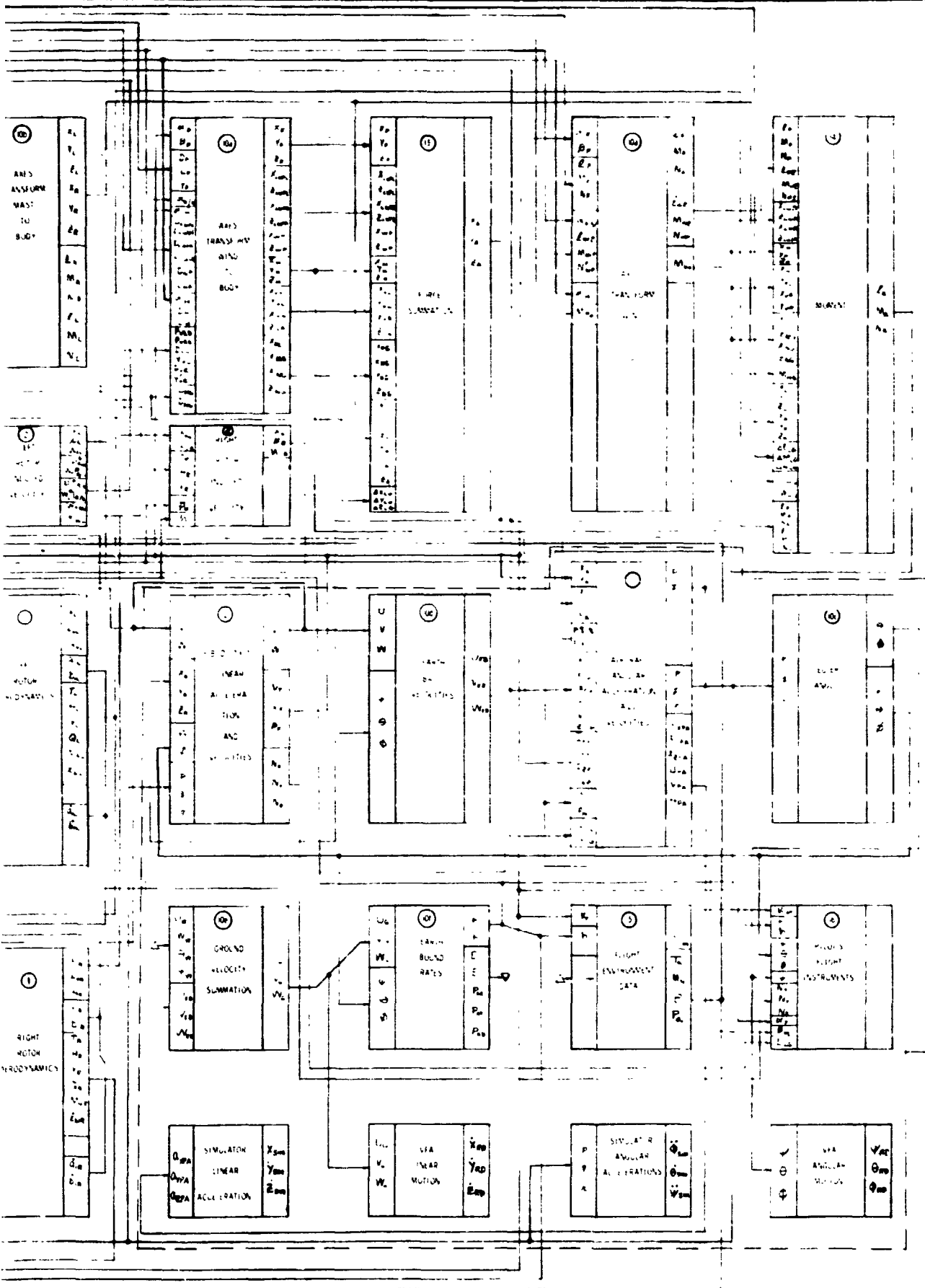


Figure I-1. Block Diagram of Mathematica

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CSC PROGRAM  
BASE 3

1. Block Diagram of Mathematical Model



## II. DESCRIPTION OF THE MATHEMATICAL MODEL

This section describes the mathematical models of the components of the tilt rotor aircraft; the rotors, the airframe, the control system, the engines and drive system, and the automatic flight control systems (Subsystems 1, 2, 3, 4, 5, 6, 7, 8, 9, 17, 18, 19 and 20 in Figure I-1). The math model used for the equations of motion (Subsystems 10 through 16) follows that of Reference 5. The coordinate systems, sign conventions and equations of motion are also described.

### A. Coordinate Systems and Sign Conventions

Earth, body, wind and mast axes systems are used in the mathematical model. The rotor flapping, forces and moments are calculated in a "wind-mast" axis system while the airframe aerodynamic forces and moments are calculated in a wind axis system. Forces and moments from the rotor and airframe are resolved into the body axis system for solution of the aircraft equations of motion. The flight path of the aircraft is described with reference to earth fixed axes with the aircraft orientation given by the Euler angles,  $\Psi$ ,  $\theta$ , and  $\phi$ , in that order of rotation.

The mast axis system and sign convention used for the rotor are shown in Figure II-1. The rotor flapping, forces and moments are calculated in the "wind-mast" axis system ( $\bar{a}_1$ ,  $\bar{b}_1$ ,  $\bar{T}$ ,  $\bar{H}$ , and  $\bar{Y}$ ) and are then transformed into the mast axis system ( $a_1$ ,  $b_1$ ,  $T_1$ ,  $H$  and  $Y$ ).

### B. Rotor Math Model (Subsystem 1)

#### 1. Rotor Forces and Moments

The mathematical model of the rotor is similar to that given in References 6 and 7 except that it is derived in a mast axis system (the theory in Reference 6 is based on an axis system perpendicular to the axis of no flapping i.e. the tip-path-plane, and that of Reference 7 is based on the axis of no feathering) and contains provisions for proprotor characteristics such as non-linear twist, flapping restraint and pitch-flap coupling.

The major assumptions that are made in the rotor math model are:

- (1) Average values for the lift curve slope and profile drag coefficient are used over the span of the blade. These are adjusted to approximate the rotor thrust and power required characteristics.
- (2) The blade angle of attack  $\alpha_r$  is approximated by  $\sin \alpha_r$ . Substitution of  $\sin \alpha_r$  for  $\alpha_r$  in the blade element equations makes it possible to develop equations for rotor forces without restricting blade pitch,  $\theta$ , and inflow angle,  $\phi$ , to small angles.

- (3) Blade stall and compressibility effects are approximated by limiting the maximum rotor thrust coefficient as a function of advance ratio and by arbitrarily modifying coefficients in the rotor power required equation (rotor profile drag is increased as a function of the cubes of the rotor inflow and advance ratios multiplied by empirically adjusted coefficients).
- (4) Blade flapping with respect to the mast is assumed small so that the small angle assumption can be made. And harmonics of flapping greater than one-per-rev are ignored.
- (5) The blade flapping due to cyclic inputs is assumed to occur instantaneously, i.e. the flapping equations assume the rotor is in an equilibrium condition. This assumption was made because of limits imposed by the computer computation time. Differential equations for blade flapping that would properly account for the rotor following time were determined to require a solution time in excess of that allowable for real time simulation. Furthermore there is a transport lag between the time a control input is made at the cab and the time aircraft response is updated at the cab, of from one to two frame times (0.050 to 0.10 seconds). By neglecting the rotor following time in the equation of motion, it is approximated by the cab-control input to computer time lag (for example, in hover the rotor following time is 0.08 seconds compared to an average computation lag time of 0.075 seconds).

## 2. Rotor Induced Velocity

The rotor induced velocity is computed by calculating the induced velocity of an isolated, out of ground effect rotor and modifying the induced velocity to account for the side-by-side rotor effect and for operation in ground effect.

The mean value of the isolated, out-of-ground effect rotor induced velocity is approximated using a modified expression from reference 8.

$$v_i = \frac{(\Omega R)C}{\sqrt{0.866\lambda^2 + \mu^2} + \frac{0.6|C_T|^{1.5} (|C_T| - 8/3\lambda|\lambda|)}{(|C| + 8\mu^2) (|C| + 8\lambda^2)}}$$

where  $C = C_T/2B^2$

(the 0.866 factor on  $\lambda^2$  has been added to improve power correlation in hover)

The side-by-side rotor effect on the rotor induced velocity is approximated using an expression derived in Reference 9.

$$\Delta v_i = x_{SS} \frac{(\Omega R) C_T}{2B^2 \mu}$$

The factor  $x_{SS}$  is called the mutual induction coefficient and is obtained from Figure 3.7 of Reference 9. The term  $\Delta v_i$  is added to the induced velocity for the isolated rotor.

In the determination of  $x_{SS}$ , the increased mass flow of the side-by-side configuration is taken into account and the rotor wakes are assumed to remain separate if the distance between the rotor centers is greater than the rotor diameter. The value of  $x_{SS}$  depends on the direction of rotation, the distance between the rotors, the advance ratio and the rotor angle of attack. The value of  $x_{SS}$  given in Reference 9 is valid for  $\mu$  greater than 0.15. In this analysis the value of  $x_{SS}$  for  $\mu$  less than 0.15 has been approximated by providing a smooth transition between a value of  $x_{SS}$  equal to zero at  $\mu = .06$  and the value at  $\mu = 0.15$ .

The reduction in induced velocity caused by ground proximity is computed using the expression

$$v_{i_{IGE}} = v_{i_{OGE}} \left[ 1 + (G-1) \left( \frac{\sqrt{U^2 + V^2}}{30} - 1 \right) \right]$$

where  $G = 0.76 + 0.24 h/D$  for  $h \leq D$

and  $G = 1.0$  if  $h > D$

The constants 0.76 and 0.24 in the expression for  $G$  were derived from model test data. The factor  $\sqrt{U^2 + V^2}/30$  washes out the effect of ground proximity with forward speed; at 30 feet per second and greater the effect is completely washed out.

The major assumption made with regard to induced velocity is that it is uniform over the rotor disc. The main effect of this assumption is that lateral flapping is underpredicted in the low speed helicopter regime ( $\mu = 0.05$  to 0.2). However, lateral flapping has only a second order effect on stability and control characteristics in the helicopter mode so this is not a serious limitation. Presently the tandem rotor effect is neglected when in sideward flight. This effect was not included in the initial development of the mathematical model since other factors significant to sideward flight simulation are only approximated.

#### C. Airframe Aerodynamics (Subsystem Nos. 3, 4, 5 and 6)

The fuselage, wing-pylon assembly, horizontal tail and vertical fins are modeled separately to facilitate accounting for the influence of the rotor

wake on the airframe aerodynamics. In general the airframe aerodynamics are extracted from wind tunnel test data; where wind tunnel data was not available characteristics were estimated using References 10, 11, and 12.

1. Fuselage (Subsystem No. 3)

Equations for the fuselage lift, drag, side force, pitching moment, yawing moment and rolling moment are referenced to the wind axis system and defined at the fuselage aerodynamics center. The coefficients in the equations for angles of attack and sideslip less than or equal to 20 degrees are based on wind tunnel data. For angles of attack greater than 20 degrees the coefficients have been approximated.

Aircraft angular rates and the rotors wakes are neglected in calculating the fuselage aerodynamic forces and moments.

2. Wing-Pylon Assembly (Subsystem No. 4)

The wing-pylon aerodynamic forces and moments are defined in the local wind axis system. Wing-body interference effects are included in the aerodynamic data.

Calculation of the wing aerodynamic forces and moments is made in two parts; (1) lift and drag generated by the rotors wakes, and (2) the forces and moments generated by the free stream flow. The math model of item (1) is discussed in Section II.3. Item (2) is discussed in the paragraph below.

The wing-pylon lift and drag generated by the freestream flow are functions of angle of attack, conversion angle, flap setting and Mach number. The pitching moment is a function of flap setting. Wing-pylon lift and drag coefficients are provided for mast angles of 0 degrees and 90 degrees, and for four flap settings as shown in Figures II-3 and II-4. Coefficients for intermediate mast angles and flap settings are obtained by interpolation. Mach number corrections are made only for the flaps-up airplane mode condition as shown in Figures II-5 and II-6. The aerodynamic coefficients for angles of attack up to stall are based on wind tunnel data; at angles of attack above stall the coefficients shown in Figures II-7 and II-8 are approximated based on the test data shown in Reference 11.

The angle of attack of the wing is modified to reflect the induction effect of the thrusting rotors. The expression for the wing angle of attack is,

$$\alpha_W = \alpha_F - 0.26 x_{R/W} * \frac{2C_{RF}}{\text{MAX}^2(\mu, 0.15)} * 57.3$$

Where  $x_{R/W}$ , the induction coefficient is a function of the distance between the rotor and the wing, and  $C_{RF}$  is the non dimensionalized rotor force coefficient.

The lateral-directional aerodynamic forces and moments are calculated using equations for stability derivatives from Reference 10. Compressibility effects and the wing loading are included in the lateral-directional characteristics. The dihedral effect of the wing-pylon is based on wind tunnel test data and is a function of angle of attack and flap setting. The aileron effectiveness is also based on wind tunnel data and is a function of angle of attack, mast angle and flaperon deflection.

3. Horizontal Tail (Subsystem No. 5)

The dynamic pressure and angle of attack at the horizontal stabilizer as shown in Figure II-9(a) takes into account wing-body blockage, mast angle, the wing-pylon wake, the rotor wake and the aircraft attitude and angular velocity. The lift and drag coefficients shown in Figures II-10 and II-11 of the stabilizer are determined from wind-tunnel test data for angles of attack up to stall. Above stall the coefficients are approximated using data from Reference 11.

The downwash at the horizontal stabilizer due to the wing-pylon shown in Figure II-12 is determined from wind tunnel data for angles of attack up to stall. Above wing stall the downwash is approximated using data for a high wing-low tail configuration given in Reference 12.

The downwash at the horizontal stabilizer due to the rotor wake is discussed in Section II.3.

4. Vertical Fins (Subsystem No. 6)

The forces and moments on the left and right fins are computed separately to account for the variation in rotor wake effects with sideslip. The dynamic pressure and angle of attack at the fins as shown in Figure II-9(b) take into account the wing-body blockage, mast angle, wing-pylon wake, rotor wake and fuselage attitude and angular rates. The lift and drag coefficients of the fins shown in Figures II-13 and II-14 are determined from wind tunnel data for angles of attack up to stall. Above stall the coefficients are approximated using data from Reference 11.

The sidewash at the fins is a function of flap setting, mast angle, fuselage angle of attack and sideslip angle.

D. Rotor Wake - Airframe Aerodynamic Interaction (Subsystem No. 2 and parts of 4, 5 and 6)

The rotor wake-airframe aerodynamic interference representation consists of three parts:

- (1) The effect of the rotor wakes on the wing lift and drag.
- (2) The effect of the rotor wakes on the horizontal stabilizer and vertical fins lift and drag.

- (3) The net rolling moment induced by the rotor wake-airframe-ground interaction when hovering near the ground.

1. Rotor Wake Effect on the Wing

As noted in Section II.C.2.2 the calculation of the wing aerodynamic forces and moments due to rotor wake effects is made separately from the forces and moments generated by the freestream flow. The calculation of the rotor wake effect involves calculating the area and angle of attack and dynamic pressure of the portion of the wing immersed in the wake; Figure II-15 illustrates the representation of this effect.

The area of the wing immersed in the rotor wake,  $S_{iW}$  (shown in Figure II-15) is computed as a function of wake radius, conversion angle, wake angle of attack and sideslip angle of the fuselage. The expression used to compute wake radius of a hovering rotor as a function of vertical distance from the rotor disc is derived in Reference 13. Experimental data also shows that the contracted wake remains stable as it reaches the wing and horizontal stabilizer surfaces. The equation for the wake radius (Equation 3 of Reference 13) has been simplified since the wing and stabilizer surfaces are located at approximately .4R below the rotor disc.

$$R_W = \{0.78 + 0.22 \text{ Exp } [-(0.3 + 2 Z \sqrt{C_{RF}} + 60 C_{RF})]\} * R$$

The rotor induced velocity at the wing varies with speed and mast tilt and is given by the following expression:

$$W_i \Big|_{R/W} = (K_0 + K_1 \mu + K_2 \mu^2 + K_3 \lambda + K_4 \lambda^2) W_{iL}$$

where the constants  $K_i$  are determined from powered rotor test data. Wing loads at high negative incidences caused by the rotor wake at low speeds are determined using lift and drag coefficient data tables that are defined up to angles of attack of  $\pm 90$  degrees.

Asymmetric flight at low speeds, which causes unequal portion of the left and right wing to be affected by the left and right rotor wakes and generates roll and yaw moments, is taken into account.

2. Rotor Wake Effect at Horizontal Stabilizer and Vertical Fins

The rotor induced velocity at the horizontal stabilizer and vertical fins is based on data from wind tunnel tests of a powered model. The velocity induced at the tail by the rotors was derived by analysis of aircraft pitching moment data with the tail on and off, and with and without the rotors. Figure II-16 shows the induced velocity at the tail (normal to the horizontal stabilizer chordline) as a function of airspeed and mast angle.

The following expression is used to approximate the velocity variation shown in Figure II-16.

$$W_i|_{R/H} = \left[ h_0 + h_1\beta_M + (h_2 + h_3\beta_M) \left( \frac{(U-168.89 + h_4\beta_M)^2}{168.89} \right) \right] * W_{iL}$$

The coefficients  $h_0$  through  $h_4$  were determined by curve fit of the test data.  $W_{iL}$  is the induced velocity at the left rotor.

The induced velocity at the empennage is presently based on the value for trimmed level flight, and is invariant with angle of attack and sideslip. Data from tests of a powered aeroelastic model indicates a significant variation in induced velocity with angle of attack and/or sideslip. This new data will be used to refine the present representation of the rotor wake effect on the empennage.

### 3. Rolling Moment Induced by Rotor Wake-Airframe Ground Interaction

When hovering in ground effect ( $h/D < 1.5$ ) an unstable roll moment is generated by aerodynamic interaction between the rotor wake, the wing and the ground. Figure II-17 shows the rolling moment as a function of  $h/D$  measured with 1/5th scale powered model of the Model 301. This effect is represented in the math model by a polynomial equation for the rolling moment, curve fitted to the test data, and applied at the aircraft center of gravity.

#### E. Flight Controls (Subsystem No. 8)

The flight controls representation consists of a controls mixing model, and a force gradient model.

The Model 301 flight control system is illustrated schematically in Figure II-18. The math model of the system represents the mixing of the pilot and automatic flight control inputs, wash out of the rotor controls as a function of mast angle and airspeed and conversion, landing gear, and flap controls. Wash out schedules are illustrated in Figure II-19 and are given in tabular form in the math model. The math model does not include friction or free play. The time constants of the control actuators are assumed zero, since in practice they are less than the computer frame time.

The pedal and cyclic stick longitudinal and lateral gradients are specified as a function of airspeed. The location of the gradient detent (zero force position) may be moved by the pilot to trim out steady stick forces.

#### F. Drive System and Engines (Subsystems 18 and 19)

The drive system is represented by the zero frequency symmetric mode, e.g. the rotors speed up or slow down in response to the unbalance between

aerodynamic torque and engine torque. The frequencies of the flexible modes of the drive system (3.67 cps and 11.8 cps for the first antisymmetric and second symmetric modes respectively) are too high to significantly influence the simulation.

The engine and power turbine (N<sub>II</sub>) governor models are composed of equations to calculate engine horsepower during transient and steady state operation. The equations are based on the operating characteristics of the combined engine-fuel control system. This approach was taken rather than one involving time constants, inertias and derivation of engine components to minimize the computational requirements.

The engine equations are derived in terms of the optimum power turbine speed and the horsepower developed at that speed. For a given throttle setting (or fuel flow rate) the engine will develop the maximum horsepower if the turbine is operating at the optimum speed. The commanded optimum power, referred to sea level, standard, static conditions is given by

$$HP_{ROC} = \begin{cases} K_8 X_{TH}^2 + K_9 X_{TH} + K_{10} & (\text{for } T_a \leq K_7) \\ \left[ 1 - K_{11} (T_a - K_7) \right] \left[ K_{12} + K_{13} (X_{TH} - K_{14}) \right] \left[ K_8 X_{TH}^2 + K_9 X_{TH} + K_{10} \right] & (\text{for } T_a > K_7) \end{cases}$$

where K<sub>8</sub> through K<sub>14</sub> are constants derived to fit the engine power versus throttle (X<sub>TH</sub>) setting characteristics given in the engine installation manual<sup>14</sup>.

The referenced optimum power HP<sub>RO</sub> at any time t, after a power lever change is given by the equation

$$HP_{RO} = (HP_{RO})_0 + \int_{t_0}^t \frac{dHP_{ROP}}{dt} dt$$

where (HP<sub>RO</sub>)<sub>0</sub> is the power before the change in the power lever position and  $\frac{dHP_{ROP}}{dt}$  is the engine power acceleration schedule and is given as

$$\frac{dHP_{ROP}}{dt} = \text{sign}(HP_{ROC} - HP_{RO}) * \text{MIN} \left\{ 1, \frac{[100] \left[ 1 - \frac{HP_{RO}}{HP_{ROC}} \right]}{\text{pctmxp}} \right\} f(HP_{RO}, h)$$

where f(HP<sub>RO</sub>, h) is the engine power acceleration schedule, derived to correlate with measured engine acceleration characteristics.



The actual horsepower HP is then computed by correcting the referred optimum horsepower,  $HP_{RO}$ , for nonstandard conditions using the following equation

$$HP = \left[ HP_{RO} \delta \sqrt{\theta} \right] \left[ K_1 \left( \frac{\Omega_R}{\Omega_{RO}} \right)^2 + K_2 \left( \frac{\Omega_R}{\Omega_{RO}} \right) + K_3 \right]$$

where  $K_1$ ,  $K_2$ , and  $K_3$  are constants used to curve fit the power to the engine characteristics given in the installation manual,  $\Omega_R$  is the actual power turbine speed and  $\Omega_{RO}$  is the referred optimum power turbine speed, and  $\delta$  and  $\theta$  are terms used to correct for nonstandard pressure and temperature, respectively.

The equations used for the power turbine governor ( $N_{II}$ ) are similar to those for the engine except that the optimum power is referred to the  $N_{II}$  speed commanded by the pilot rather than the throttle setting. It should be noted that in the Model 301 the  $N_{II}$  governing speed is set at that corresponding to the rotor limit speed so that the  $N_{II}$  governor is only used to help prevent overspeeding.

#### G. Automatic Flight Control System (Subsystem No. 17 and 20)

The Model 301's rotor rpm governor and Stability and Control Augmentation System (SCAS) are represented in the math model.

##### 1. Rotor RPM Governor (Subsystem No. 17)

The rotor RPM governor representation consists of a single channel model of the feedback network. In the Model 301 the rotor blade collective pitch is changed so as to maintain constant rpm; the blade pitch is proportional to the integral of the error in rpm (e.g. the difference between the actual and the pilot selected rpm) so that any steady error is completely washed out. The gain of the integral feedback is very low so the governor will not destabilize structural modes.

A position gain is used in parallel with the integral gain to provide damping to the rotor rotational mode under conditions of low inflow, such as in low power descents in helicopter mode. The position gain is phased out as the pylons are converted to airplane mode to prevent destabilizing structural modes.

Control of the RPM governor consists of a thumb operated, three position switch spring loaded-to-center, located on the power lever head. Pushing the switch forward, increases the reference rpm by 20 rpm for each second the switch is depressed; pulling aft decreases the reference rpm by 20 rpm per second. A pointer on the rotor tachometer indicates the selected rpm. This system was modeled in the FSAA cab. At present, this modeling does not include capability to evaluate failure modes.

## 2. SCAS (Subsystem No. 20)

The SCAS math model consists of a single channel representation of the electronic network; SCAS actuator characteristics are not modeled, however, total system authorities are used. A typical channel is shown in Figure II-20. Each channel has three functional circuits; rate feedback to augment the aircraft damping, attitude feedback for attitude retention, and pilot input feed forward to improve the apparent response to pilot control inputs. The pitch channel has, in addition, an airspeed feedback loop (this loop was implemented but not used).

The main feature of the SCAS math model is representation of the systems gains. All gains are functions of both airspeed and pylon angle. This feature was incorporated to facilitate optimizing the SCAS during the design phase.

The attitudes and airspeed hold circuits are switched off or on by the position of the pedals and cyclic stick. When the longitudinal cyclic stick is trimmed (e.g., in the force detent) the pitch channel holds are switched on; when untrimmed the pitch channel holds are switched off. When the lateral cyclic stick and the pedals are trimmed the roll and yaw attitudes holds are switched on; when either the pedal or the lateral cyclic is out of trim the roll and yaw holds are switched off.

## H. Equations of Motion and Coordinate Transformations (Subsystem 9, 10, 11, 12)

The equations of motion used to solve for the six degrees of freedom flight path are identical to the ones given in Reference 5. Change in aircraft c.g. location and inertia due to pylon tilt are computed. Equations derived to compute pilot station acceleration take into account the c.g. acceleration due to pylon tilt rate and the c.g. offset location. However, the effects of mast tilt acceleration and rate on the body pitch acceleration and rate are assumed to be small and therefore neglected. The pylon degrees of freedom are neglected as the Model 301 wing-nylon natural frequencies are well above the frequency capability of the simulation software and hardware.

Transformation of forces and moments from wind to body axes and from mast to body axes is required for a number of subsystems. These transformations are given in Subsystem 10. The transformation for the rotor equations are given in Subsystem No. 1. Rotor forces and moments are computed in the wind axis and referenced to the mast. This is noted throughout this report as the "wind-mast" axis system as described in Figure II-1. The forces and moments computed in this axis system are then transformed to the mast axis system. Transformation of rotor forces and moments to the body axis system are given in Section 10b.

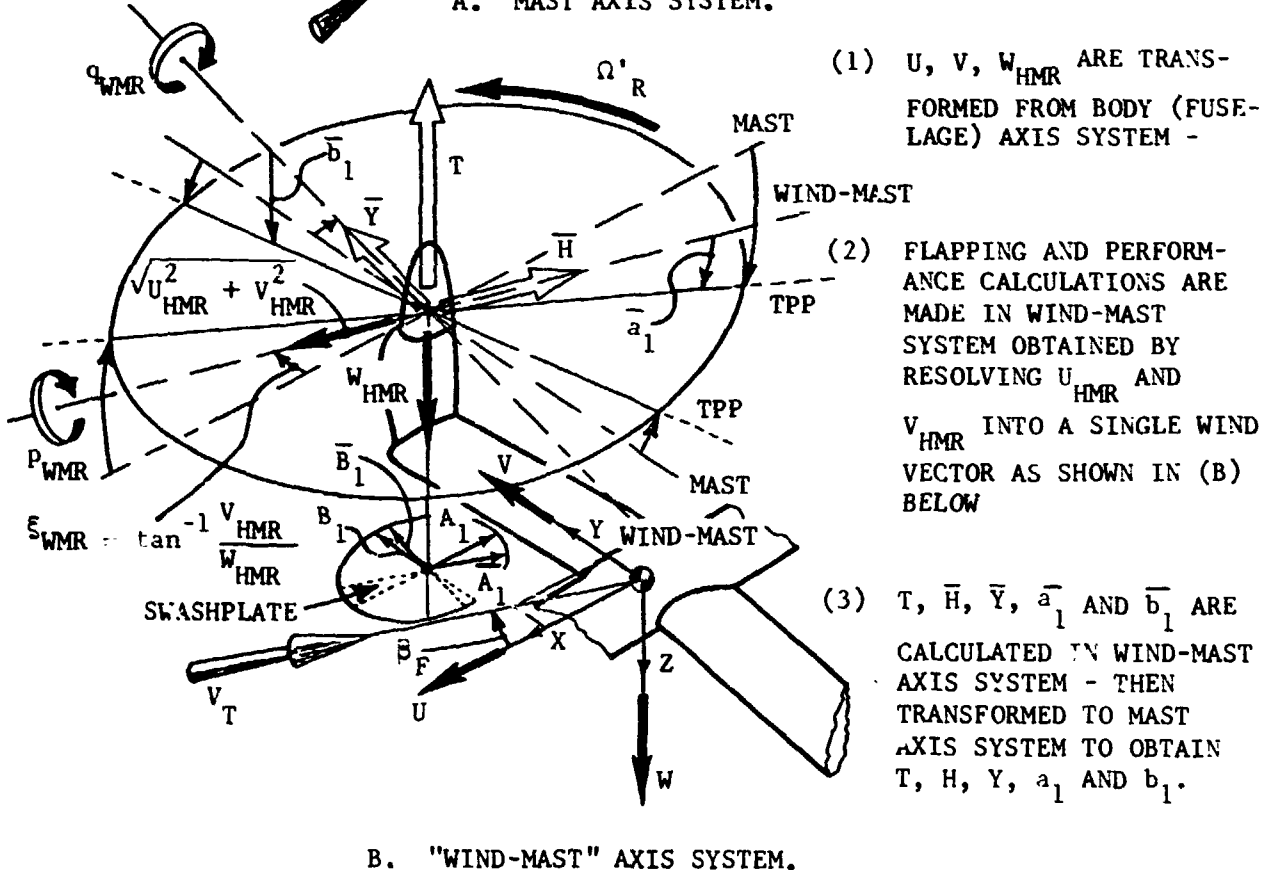
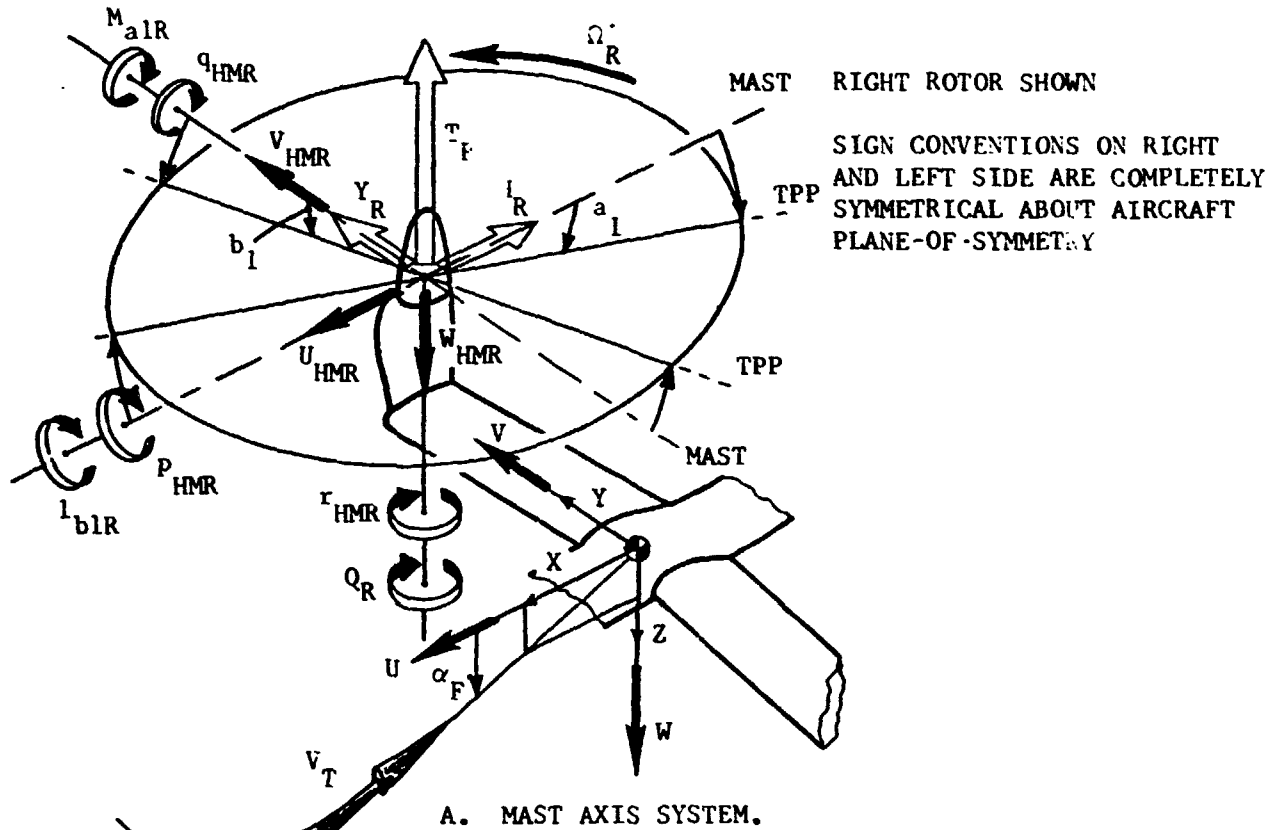


Figure II-1. Rotor Axis Systems and Sign Conventions

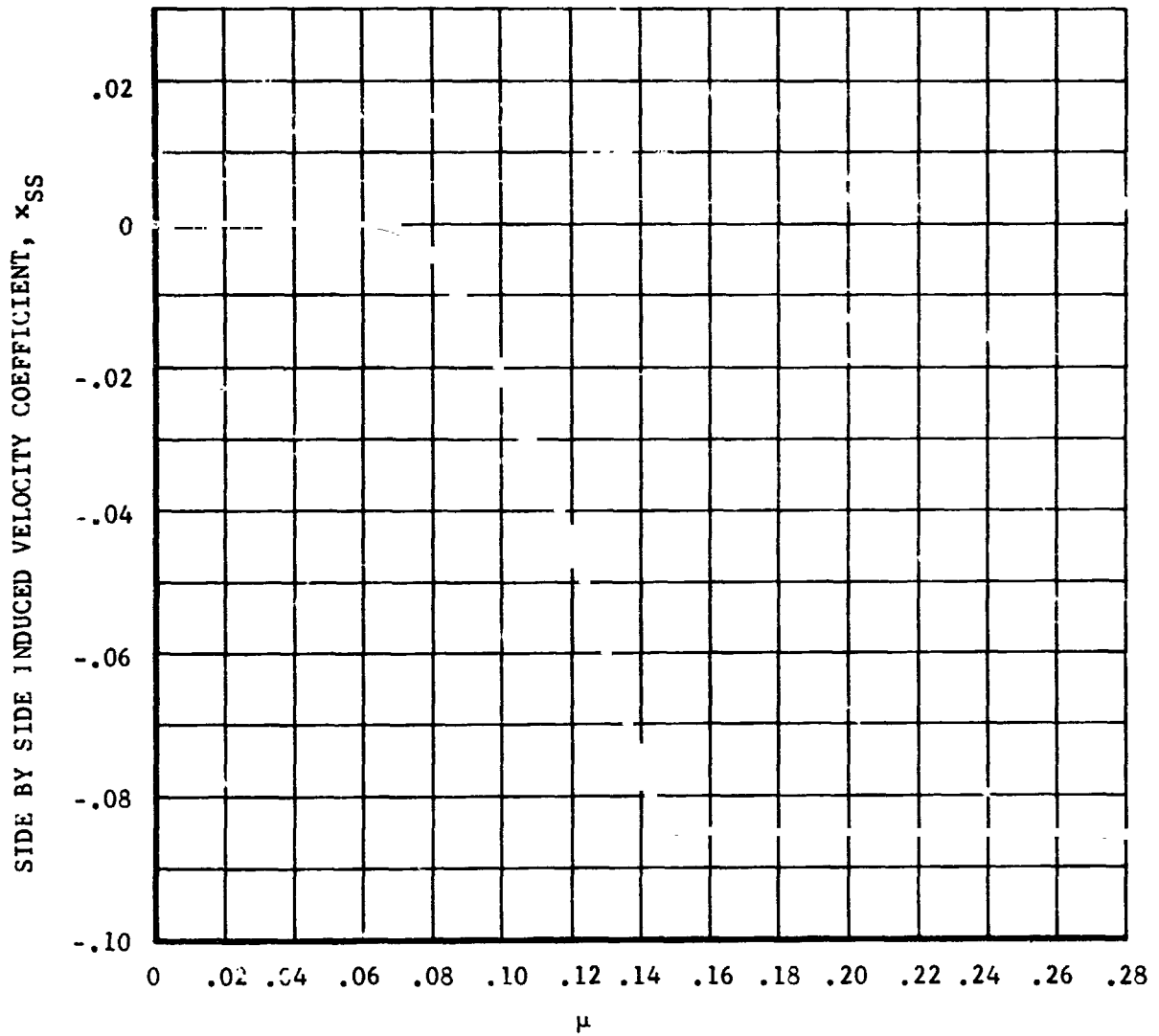


Figure II-2. Side-By-Side Rotor Effect on Induced Velocity

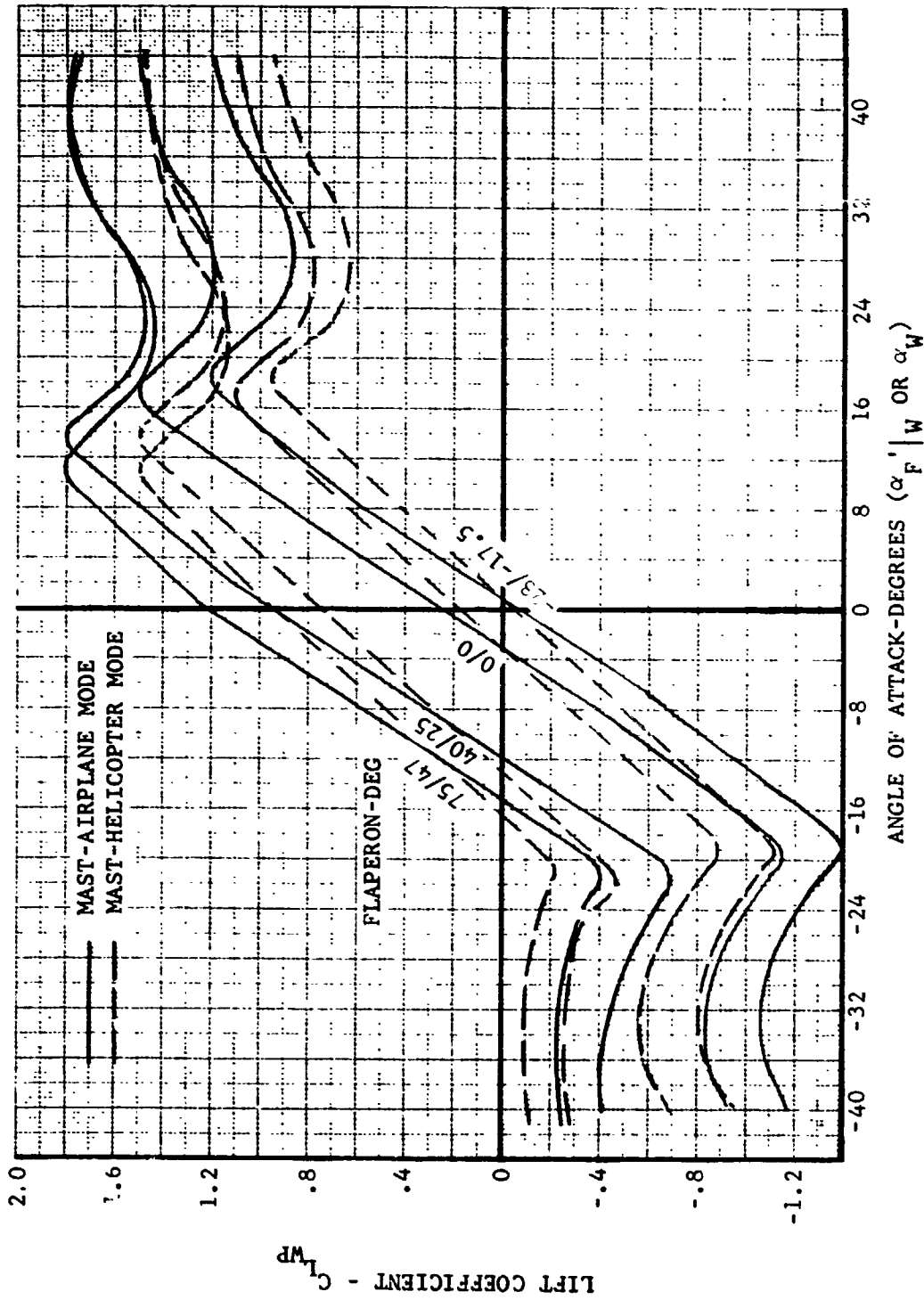


Figure II-3. Wing-Pylon Lift Coefficient Versus Angle of Attack

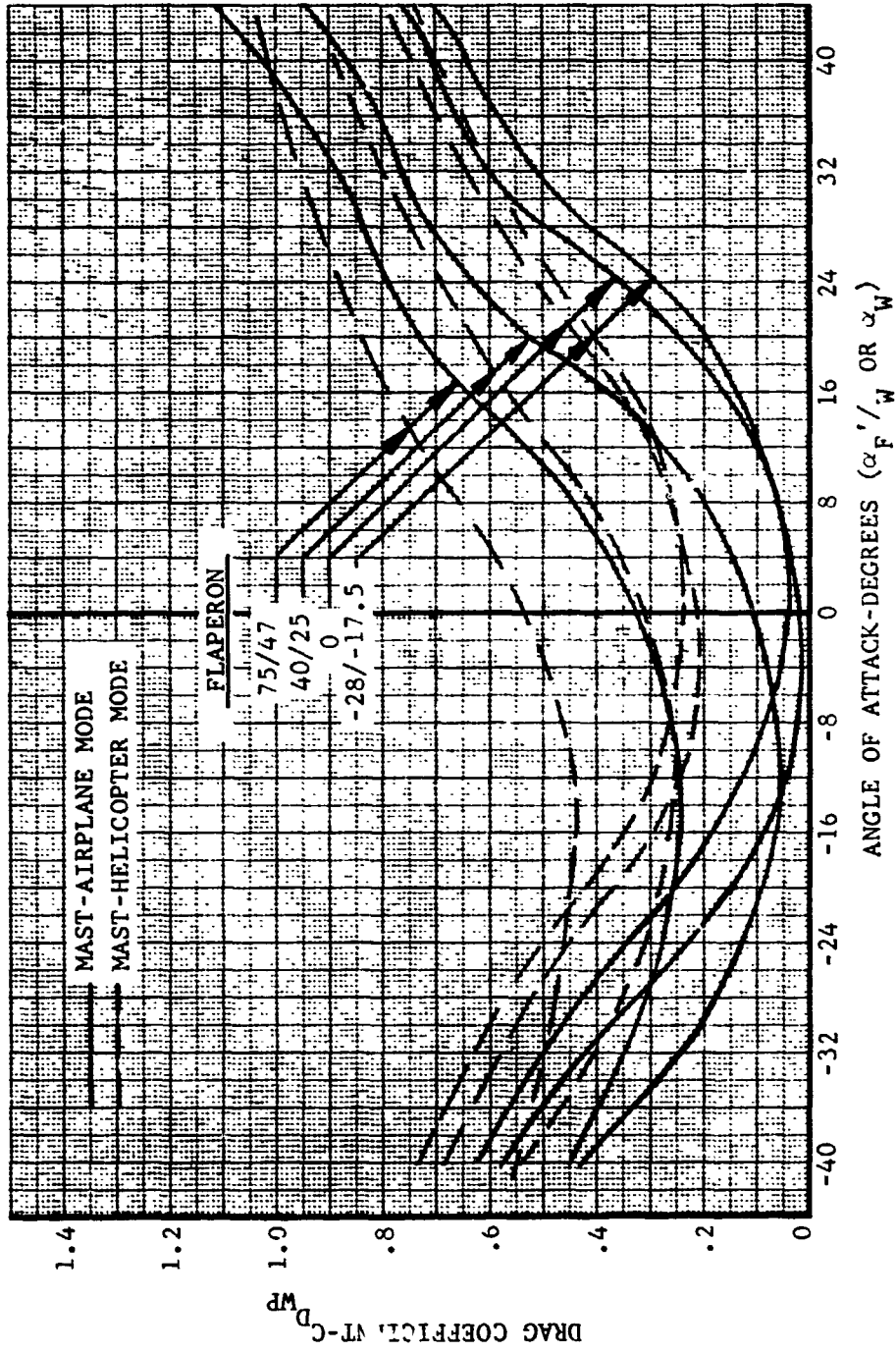


Figure II-4. Wing-Pylon Drag Coefficient Versus Angle of Attack

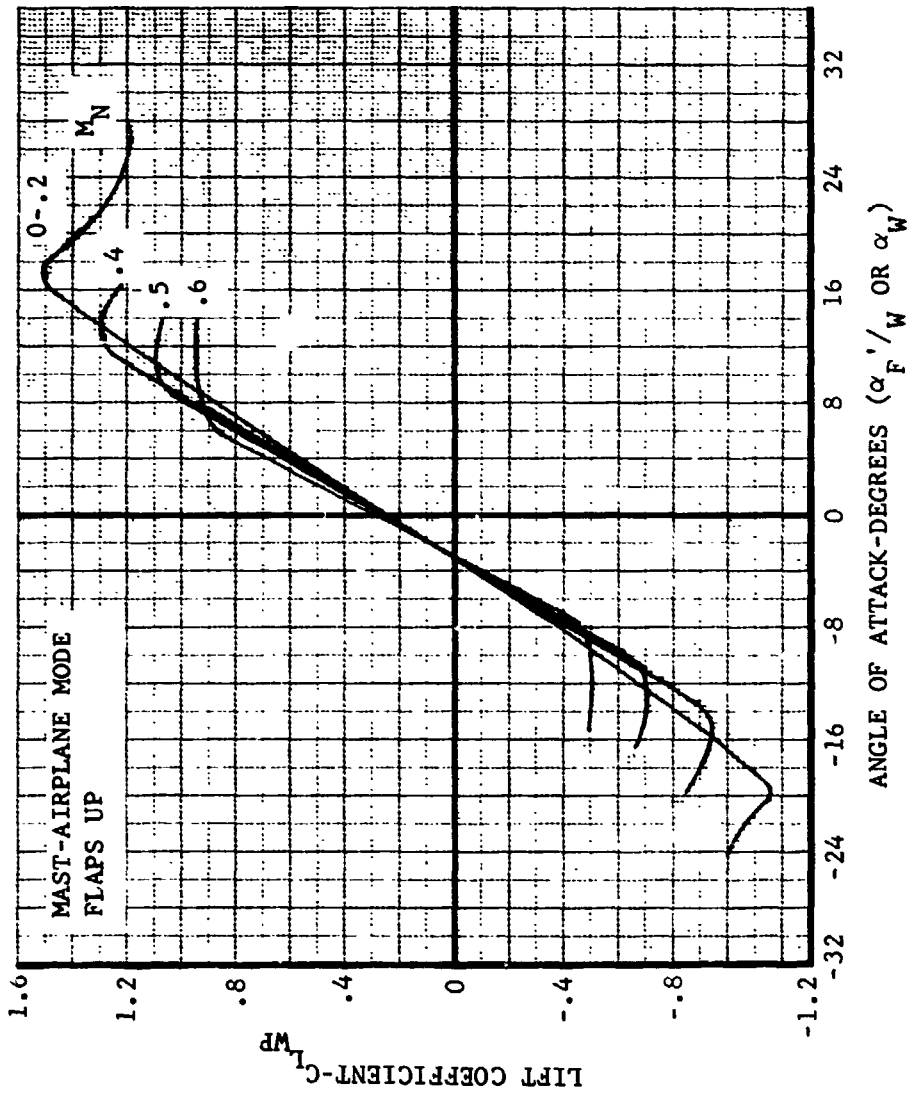


Figure II-5. Compressibility Correction to Wing-Pylon Lift Coefficient

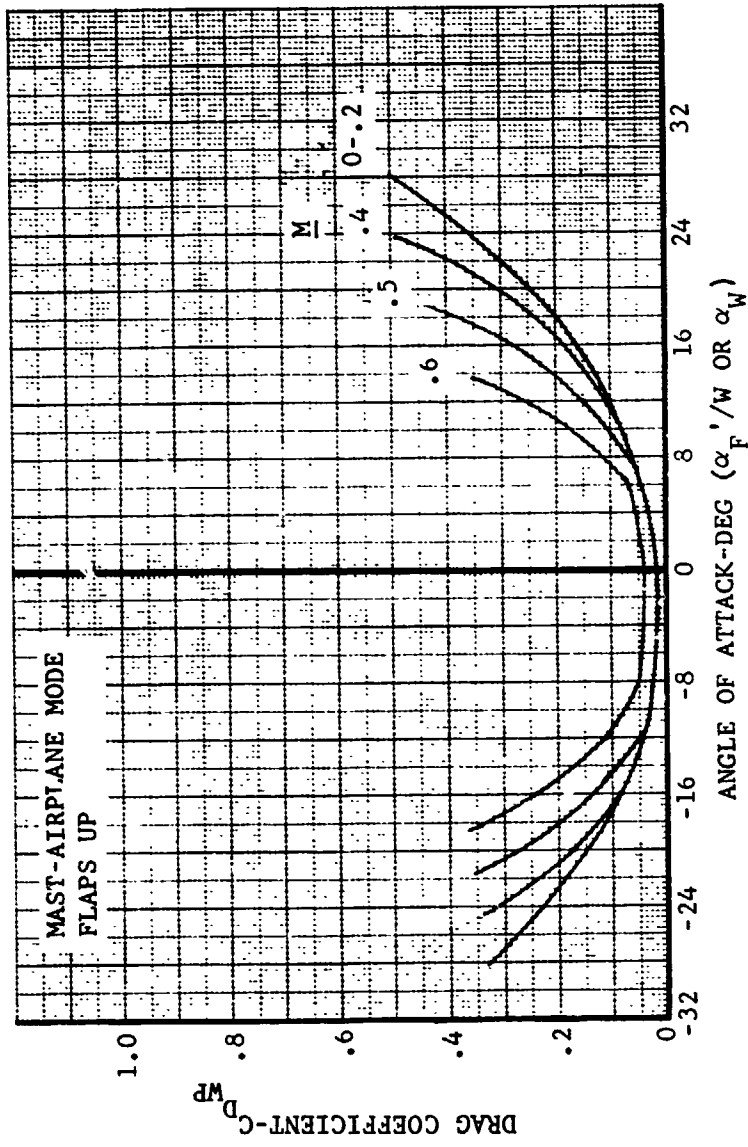


Figure II-6. Compressibility Correction to Wing-Pylon Drag Coefficient



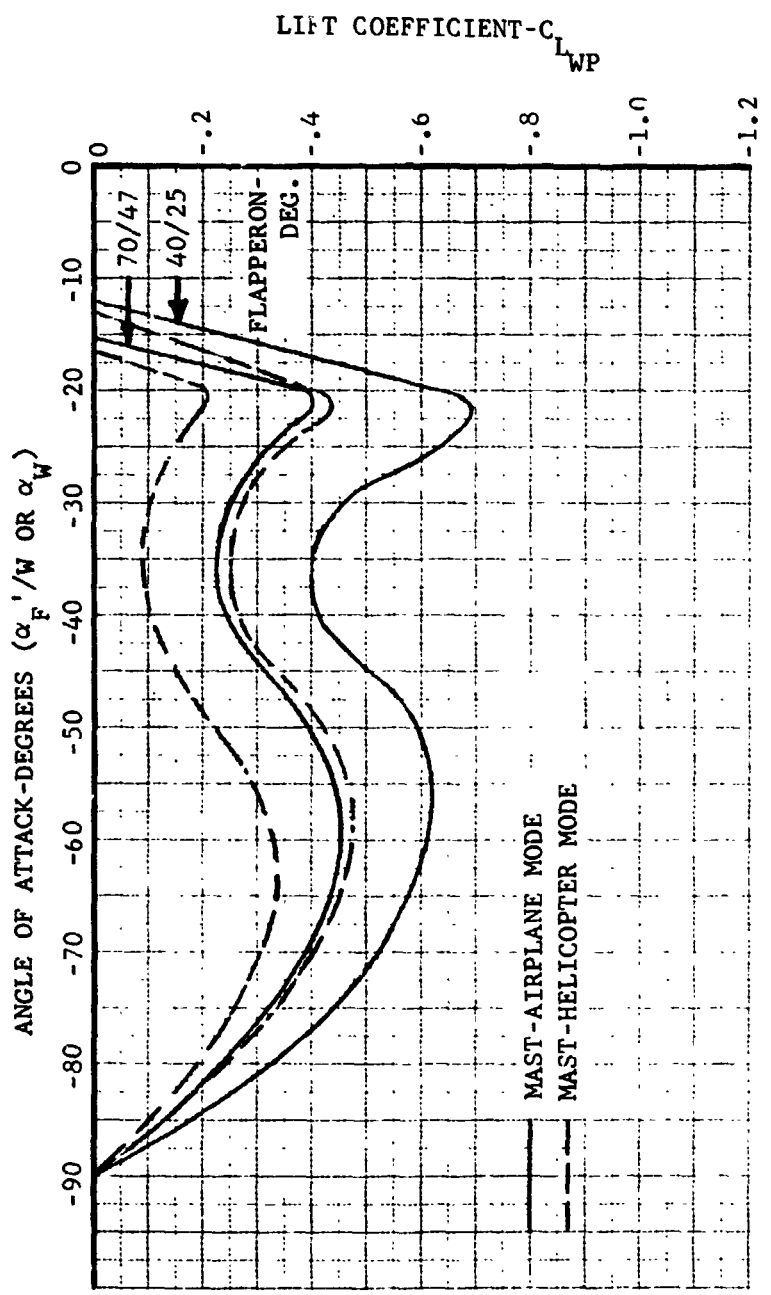


Figure II-7. Wing-Pylon Lift Coefficient at Large Negative Angles of Attack

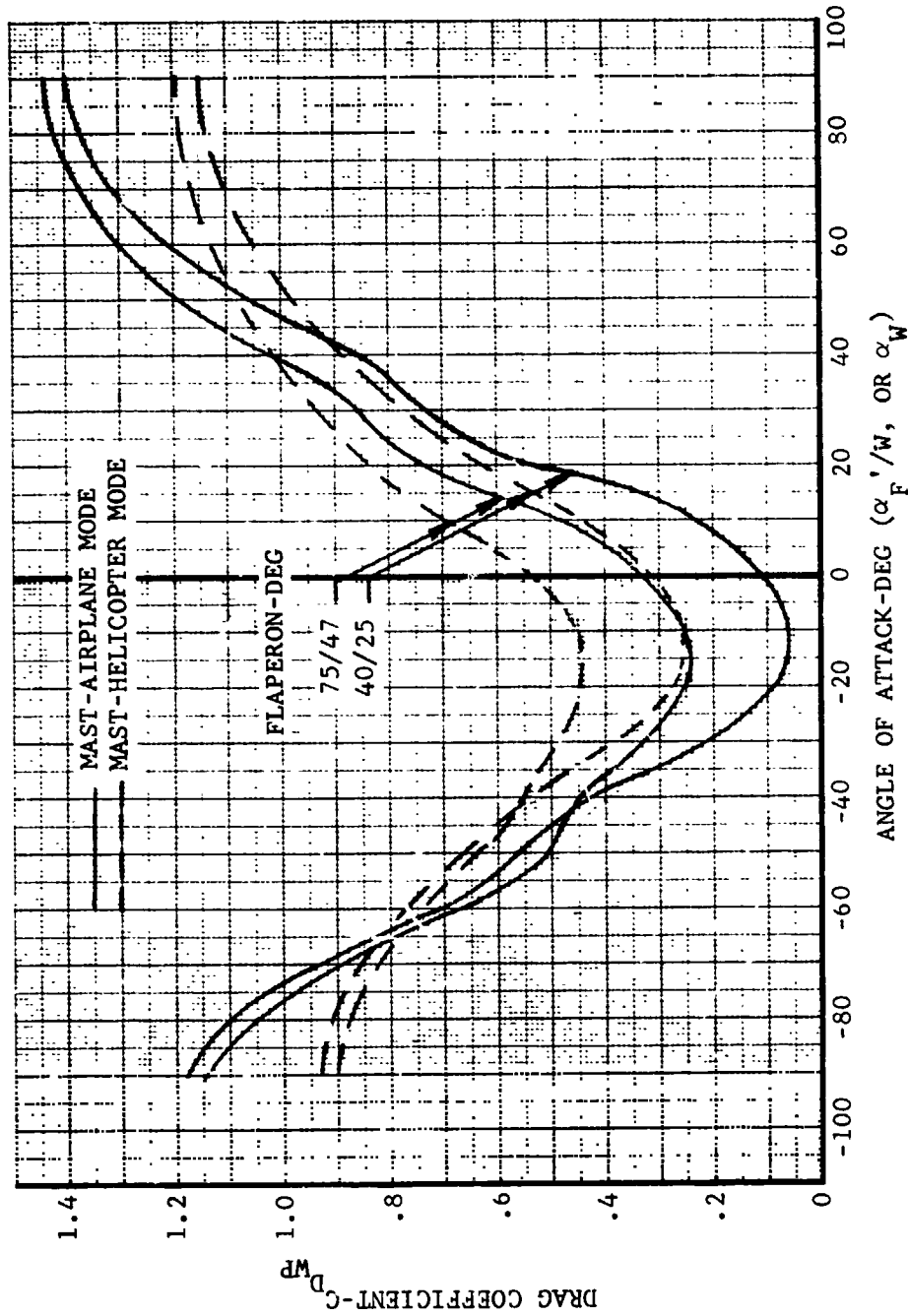


Figure II-8. Wing-Pylon Drag Coefficient at Large Negative Angles of Attack

NOTE: Angles shown are positive values in mathematical model sign convention.

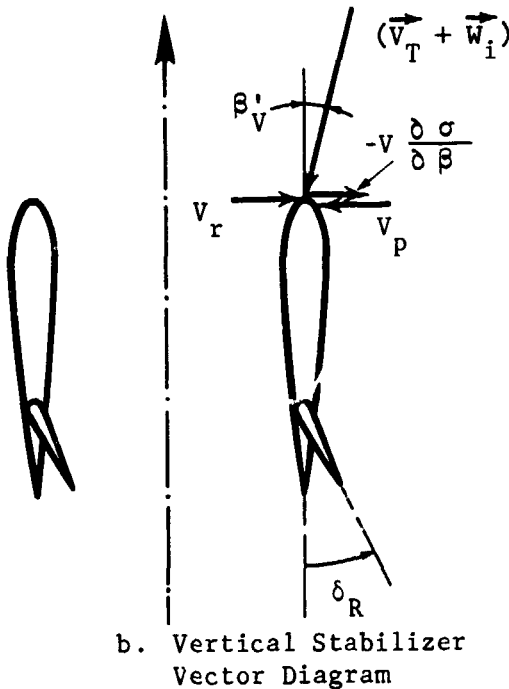
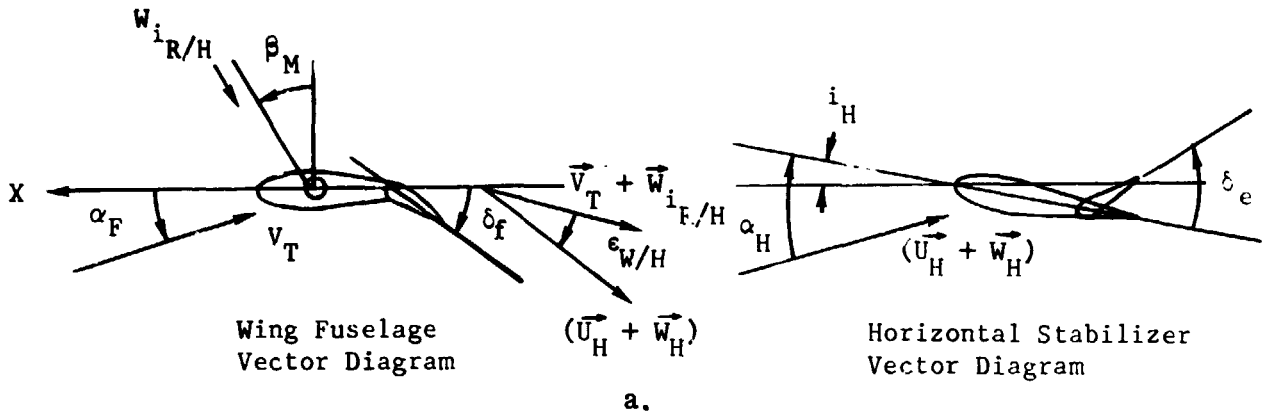


Figure II-9. Sign Conventions and Notation for Horizontal Stabilizer and Vertical Fin Aerodynamics

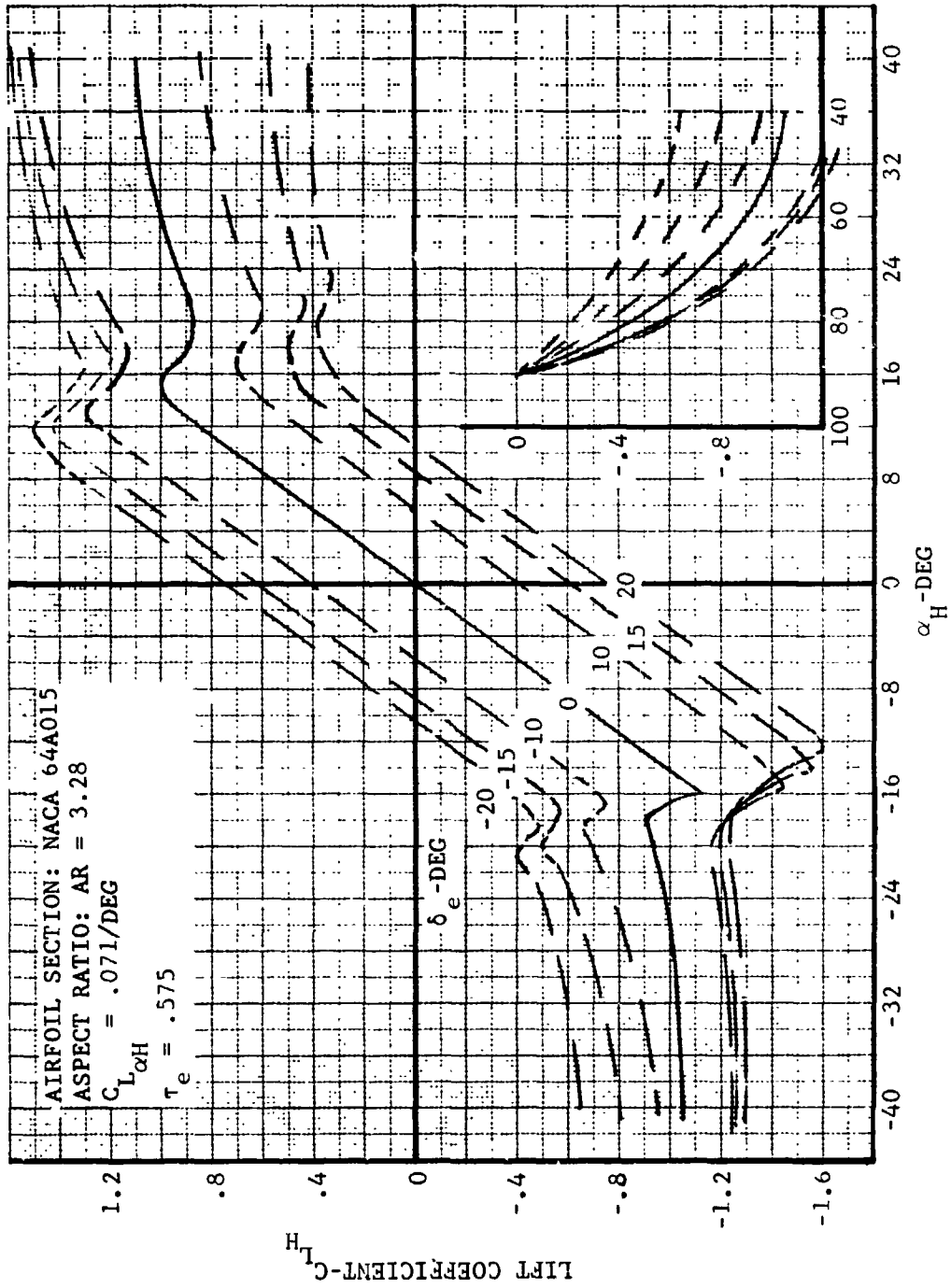


Figure II-10. Horizontal Stabilizer Lift Coefficient Versus Angle of Attack

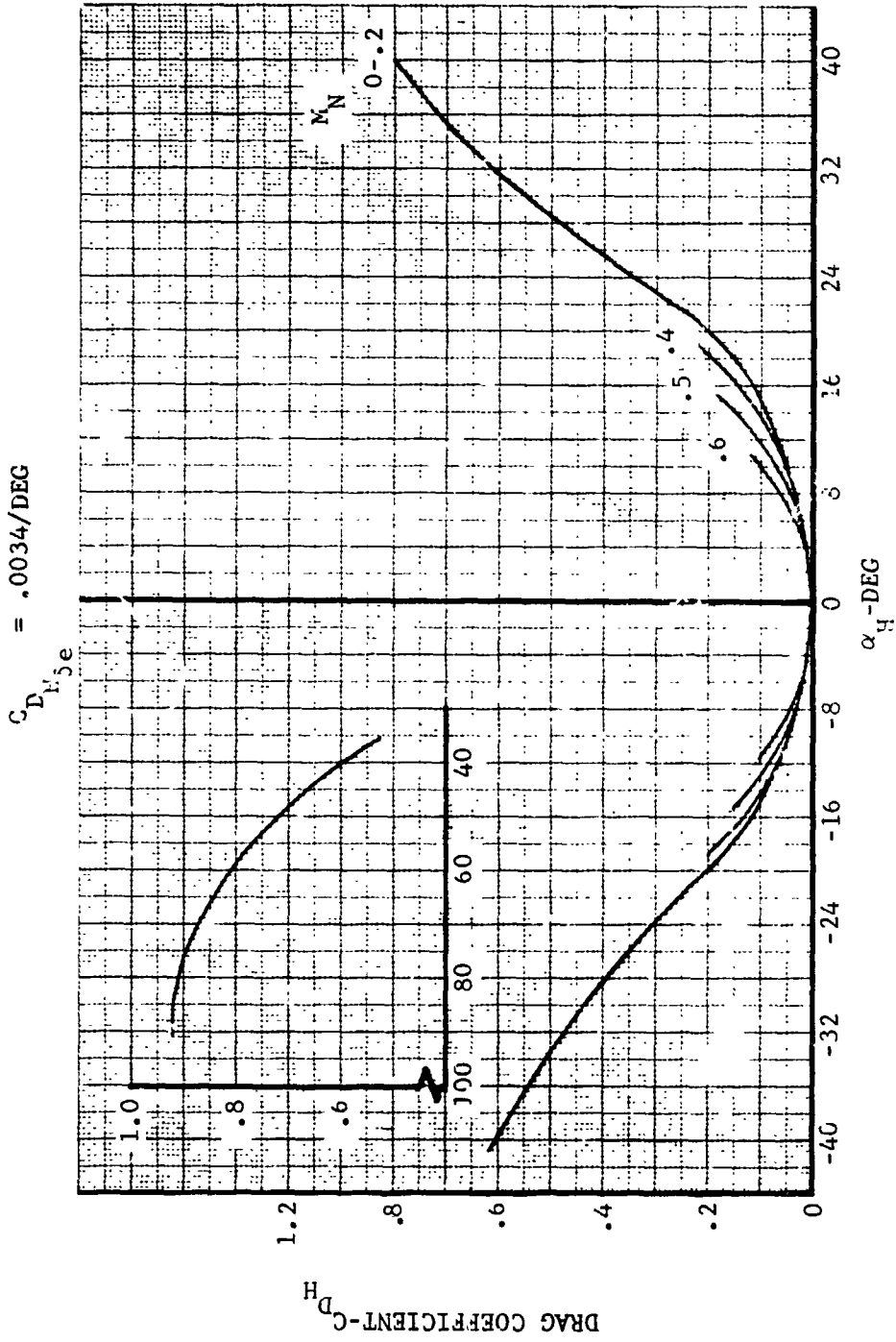


Figure II-11. Horizontal Stabilizer Drag Coefficient Versus Angle of Attack

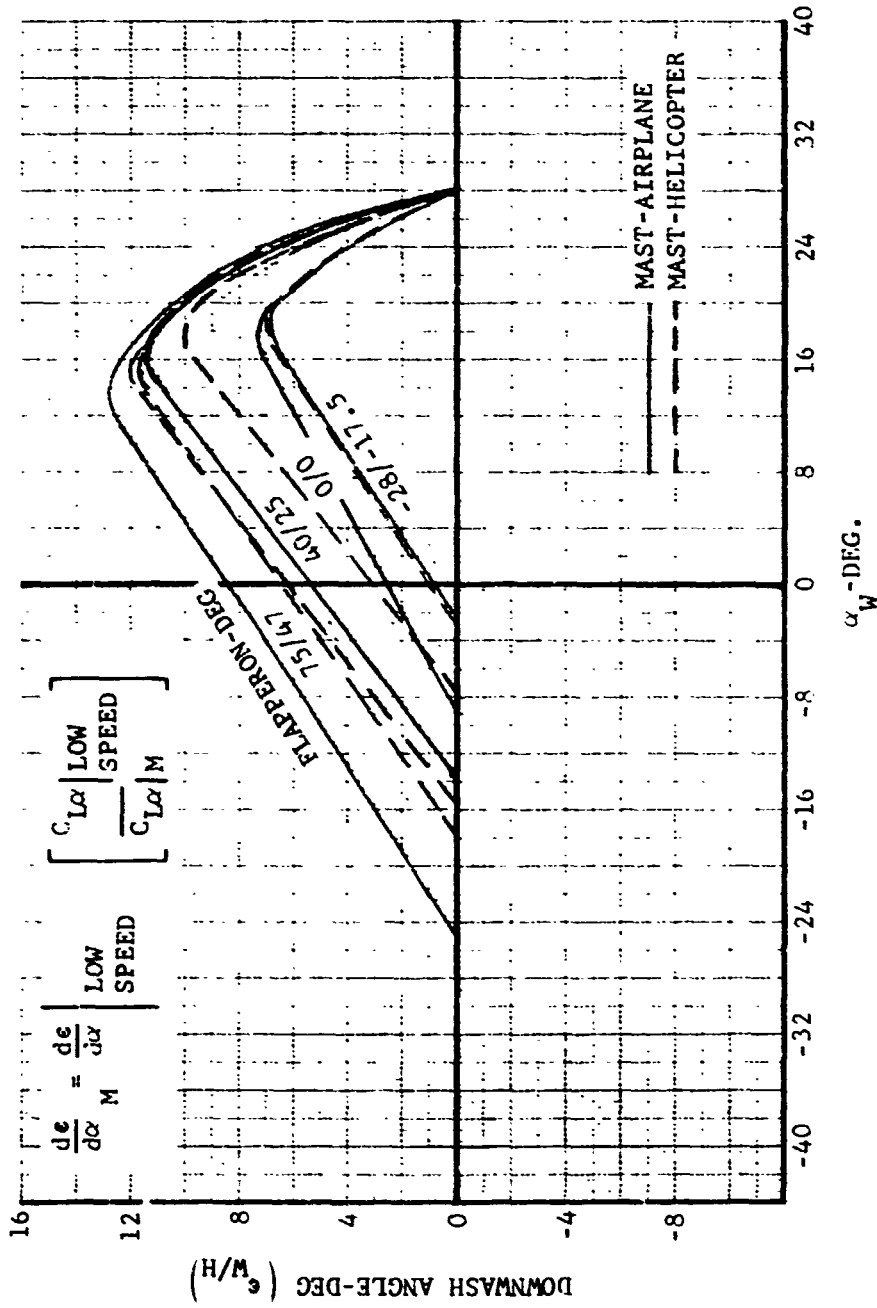


Figure II-12. Wing-Pylon Wake Deflection at Horizontal Stabilizer

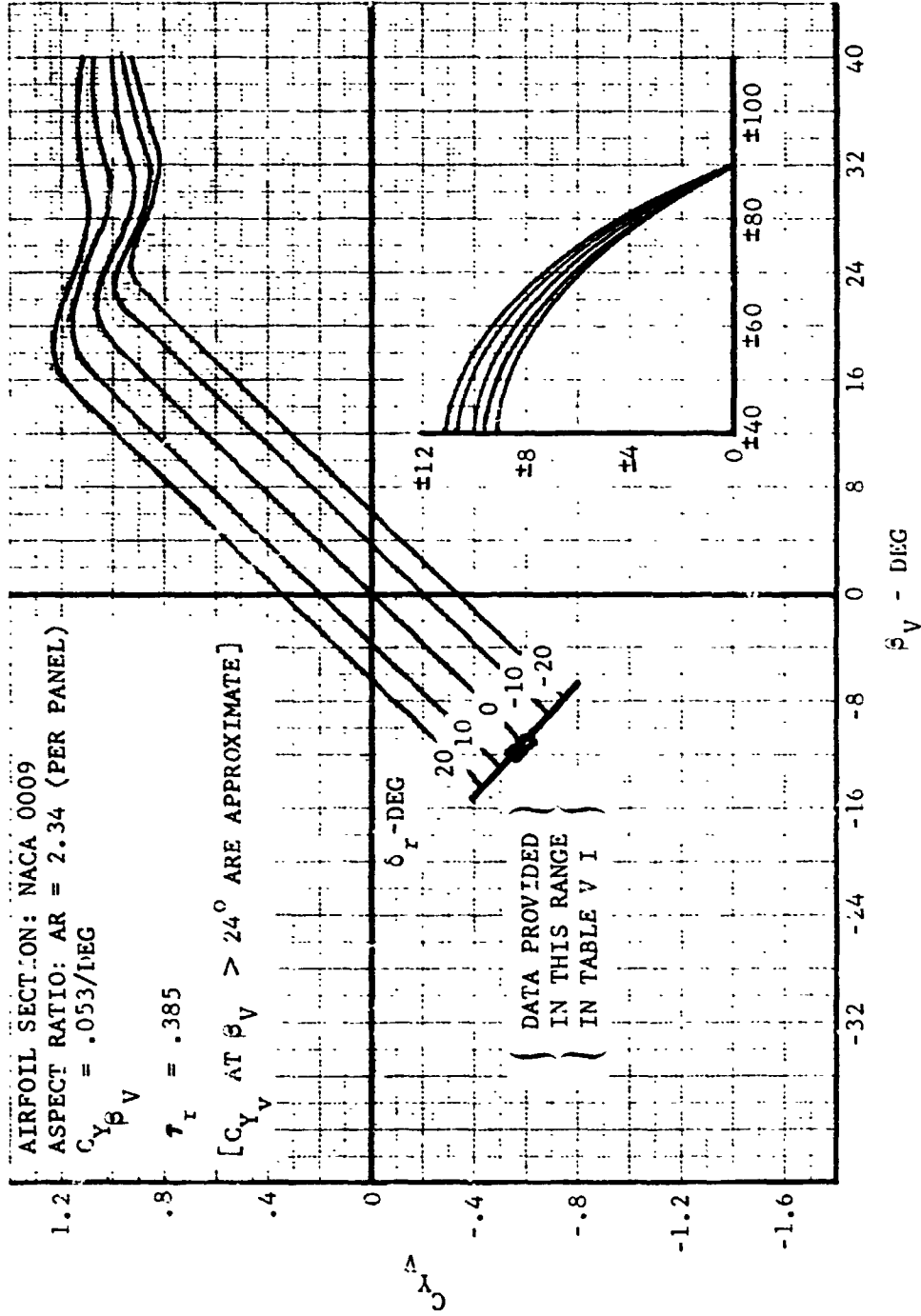


Figure II-13. Vertical Stabilizer Side Force Coefficient Versus Sideslip Angle

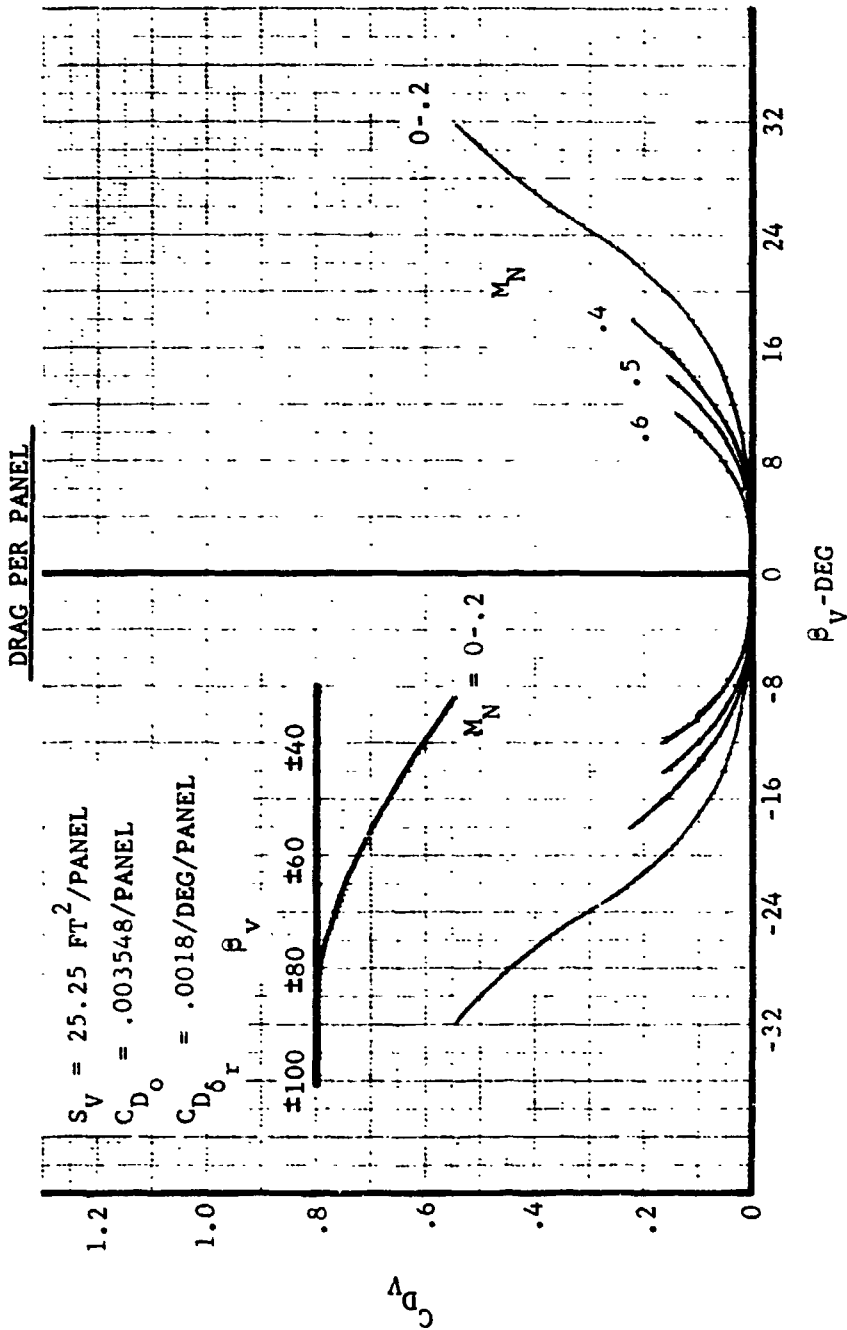


Figure II-14. Vertical Stabilizer Drag Coefficient  
Versus Sideslip Angles





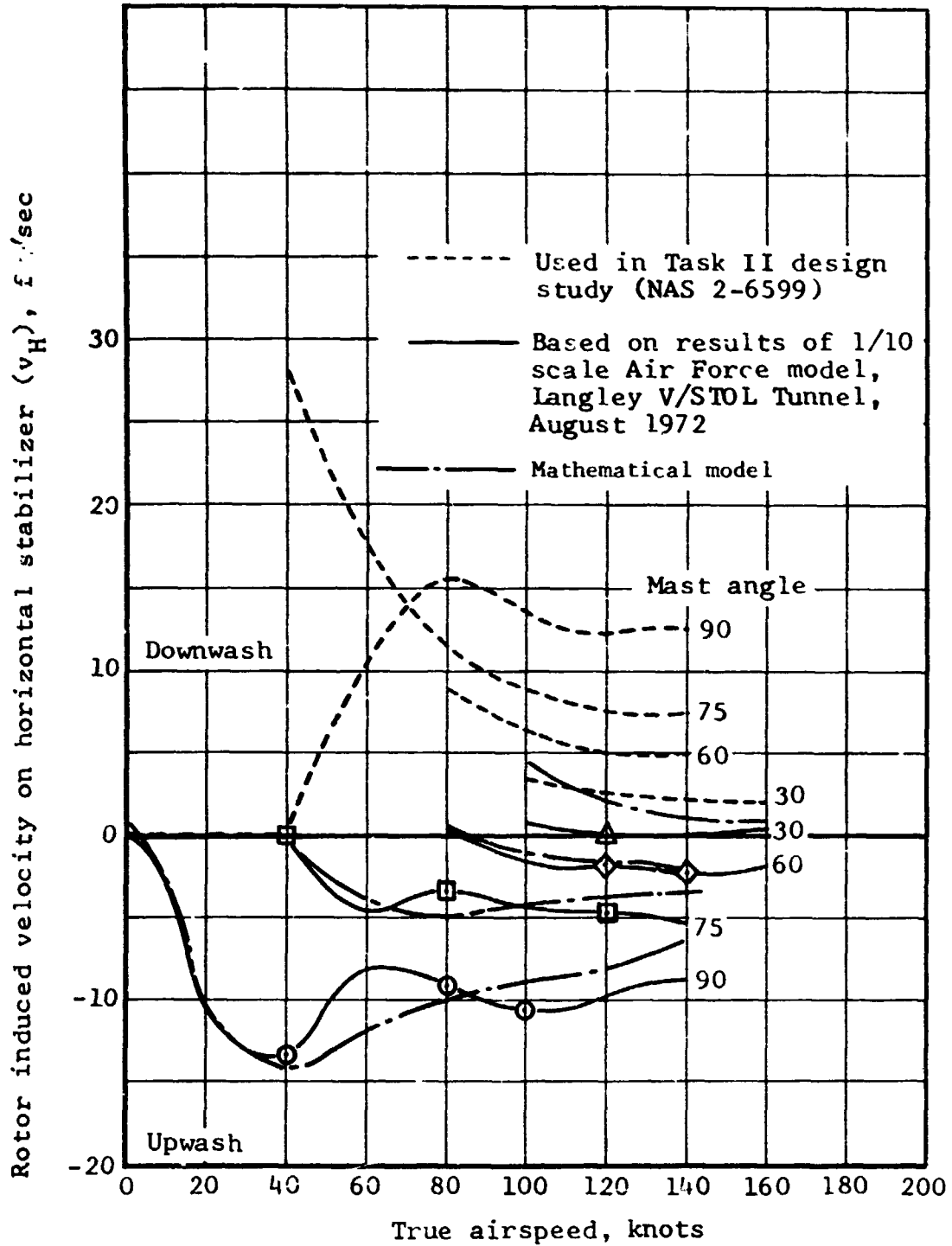


Figure II-16. Model of Rotor Wake Induced Velocity at Horizontal Stabilizer

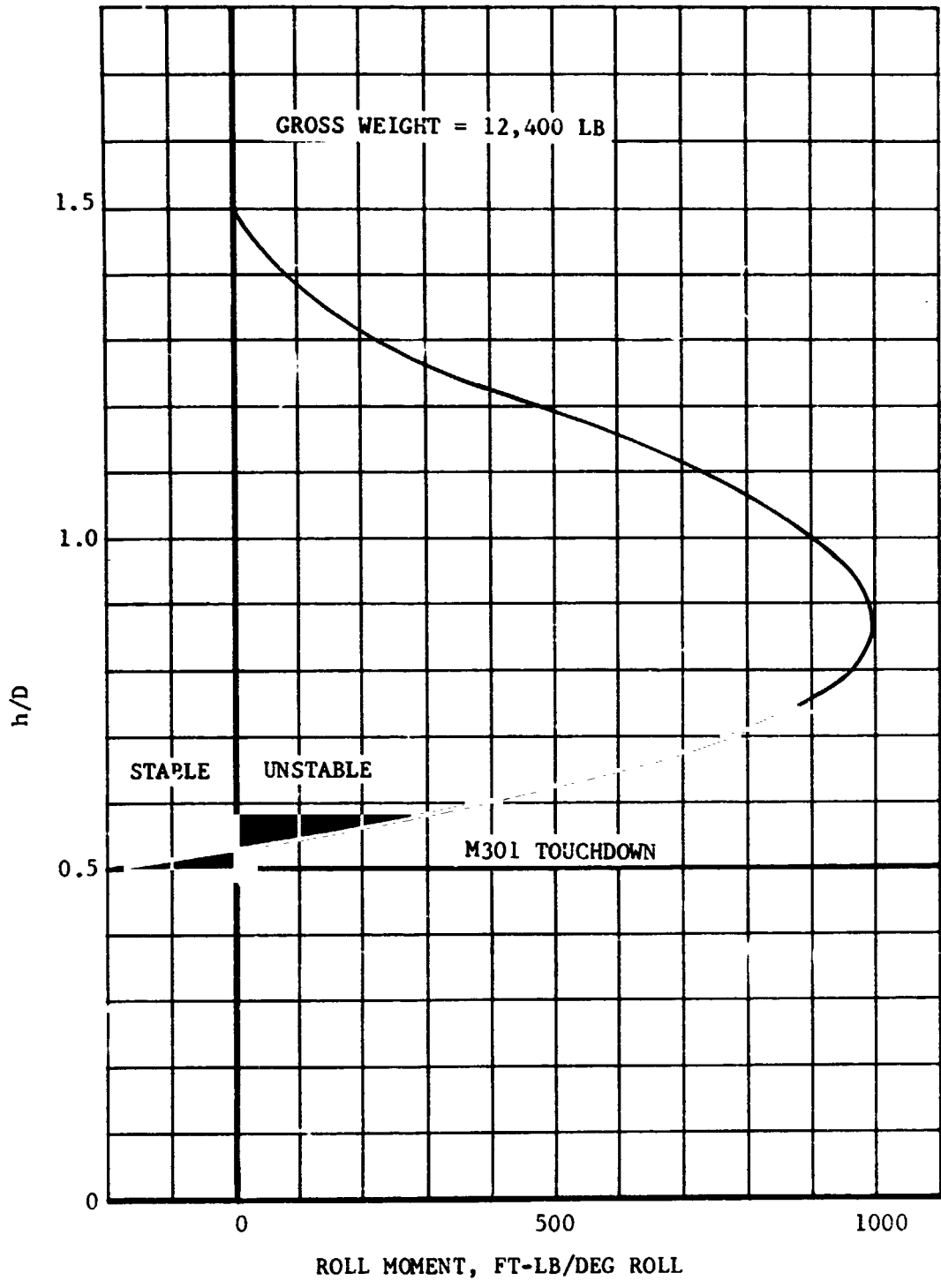


Figure II-17. Representation of In-Ground Effect Rolling Moment

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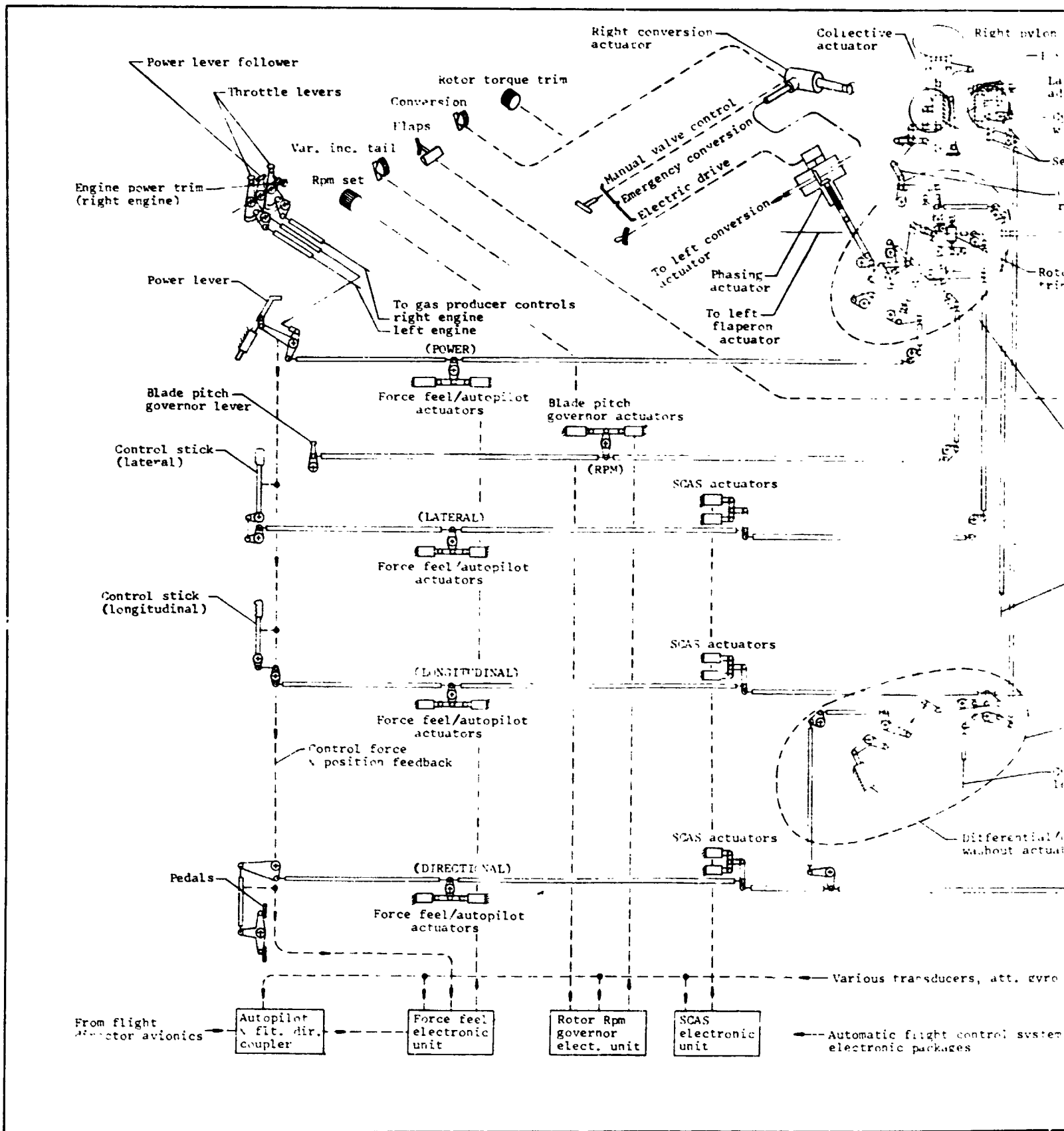
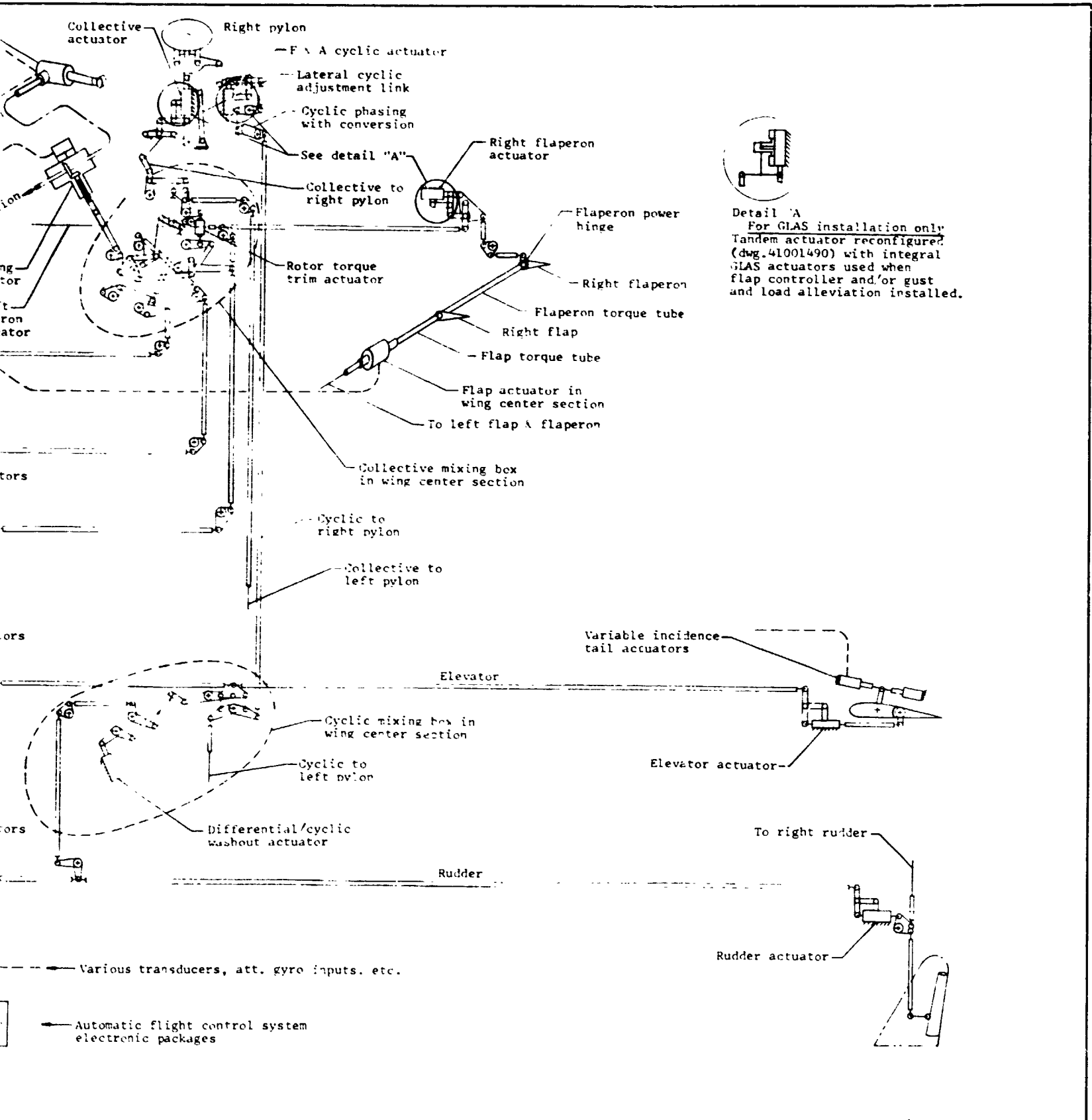


Figure II-18. Schematic of Model 301 Flight

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Schematic of Model 301 Flight Control System

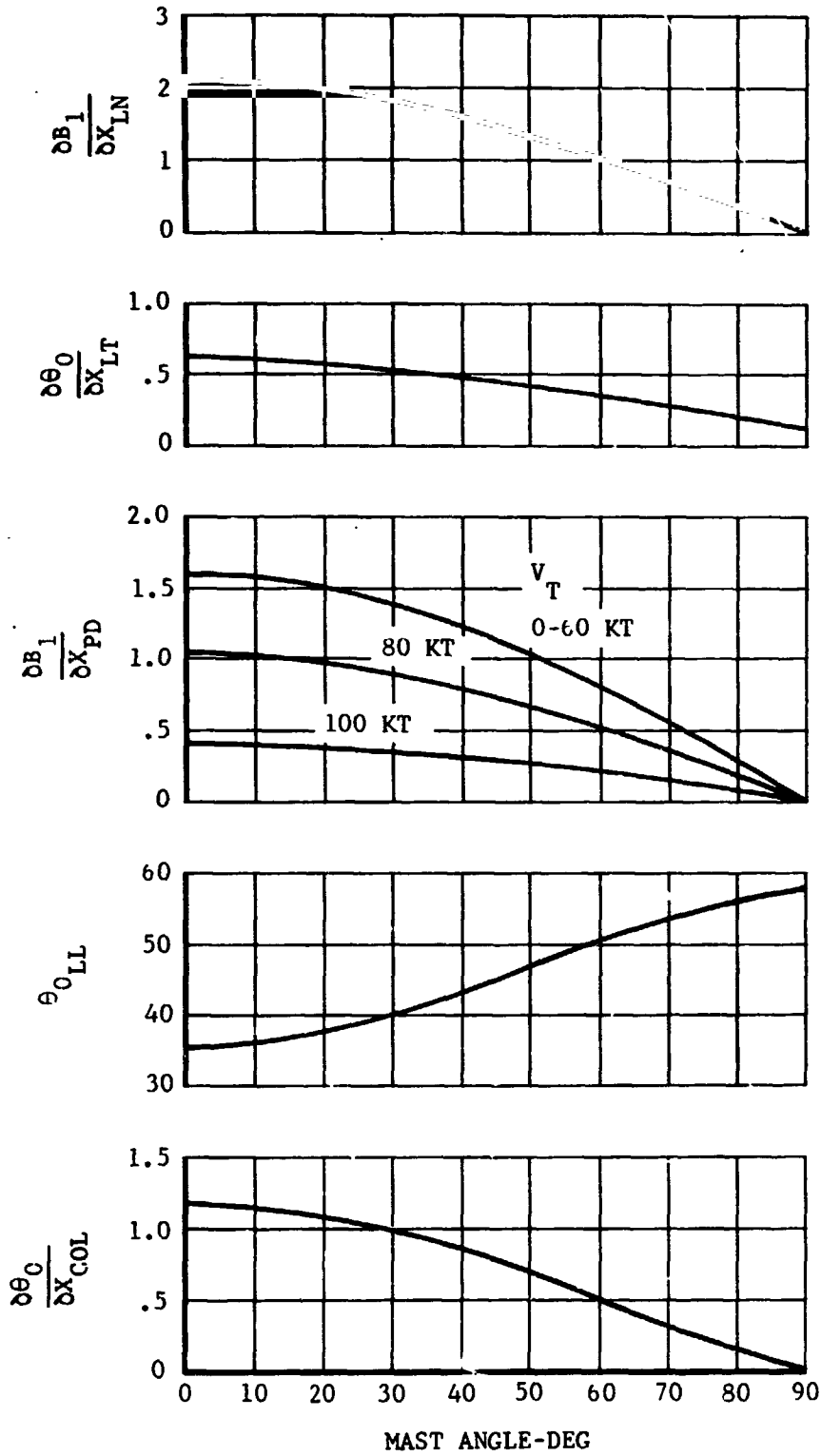


Figure 11-19. Computer Representation of Flight Control System Wash-Out Schedule With Conversion Angle

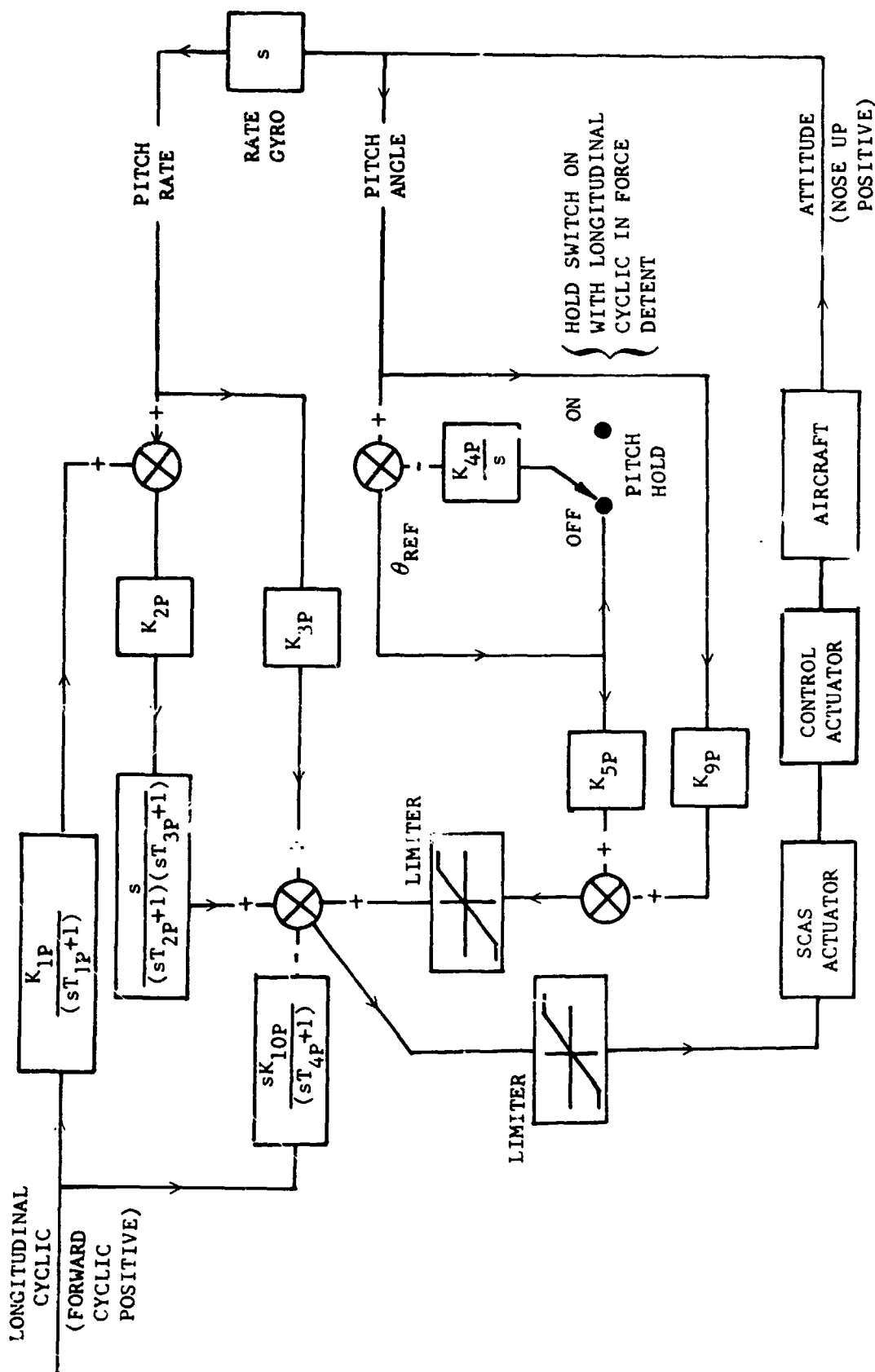


Figure 11-20. Schematic of SCAS Pitch Channel Network

### III. VALIDATION OF THE MATHEMATICAL MODEL

The accuracy of the mathematical model has been investigated with regard to rotor performance and force characteristics, airframe aerodynamics, rotor wake - airframe aerodynamic interaction, static and dynamic stability characteristics and control power and damping. Test data were used for comparison where available; when test data were not available the mathematical model was compared with BHC computer program C81, the program used to predict the Model 301's stability, control, and handling qualities presented in Reference 1.

#### A. Rotor Performance and Force Characteristics

Rotor forces and power required are compared with data from wind tunnel and whirl tests of the Model 301's 25-foot diameter rotor (References 15 and 16) in Figures III-1 through III-5. In general the correlation reflects the neglect of stall and compressibility in the mathematical model. For a given lift, power required is calculated slightly low and propulsive force is slightly high. Collective pitch and longitudinal cyclic are calculated accurately.

Figure III-6 compares the rotor damping characteristics with those calculated with program C81. Agreement between the two analyses is good.

#### B. Airframe Force and Moment Characteristics

As noted in Section II.3 the mathematical model of the airframe force and moment characteristics in the angle of attack and sideslip range below stall are based on wind tunnel data. In the mathematical model, the wind tunnel data has been broken down into the contributions of major components; the fuselage, the wing-pylon assembly, the horizontal stabilizers and the fins. Above stall the force and moment characteristics have been approximated.

In order to verify that the force and moment data add up to equal that of the complete airframe a computer "wind tunnel" test of the airframe math model (with the rotors removed) was conducted. The mathematical model force and moment characteristics are compared to the wind tunnel data in Figures III-7 through III-11. Data from two tests are shown: LTV Low Speed Wind Tunnel Test No. 408 performed at a Reynolds number of  $1.2 \times 10^6$  and NASA Langley Transonic Dynamics Test No. 195 performed at a Reynolds number of  $3.8 \times 10^6$ . Vortex generators were installed on the wing during the LTV test to simulate wing stall characteristics at the full-scale Reynolds numbers.

The reasons for significant difference between the mathematical model and the test data are as follows:

- Lift Coefficient (Figures III-7 and III-8). Correlation is good for angles of attack up to stall but above stall the mathematical



model lift coefficient is high. This error is caused by the assumed lift characteristic above stall (Reference 11) and will be corrected for the Phase II simulations.

- Pitching Moment Coefficient (Figures III-11 and III-12). The pitching moment characteristic of the full-scale aircraft is estimated to be close to the TDT test data, hence the mathematical model has been adjusted to obtain the slope of the TDT data. The discontinuity in the mathematical model pitching moment coefficient at positive and negative stall angles is caused by the assumed characteristics of the wing downwash and the fuselage pitching moment above stall. The difference in pitching moment at zero angle of attack as shown in Figure III-11 results from an assumption made in the mathematical model - that the wing-pylon zero pitching moment is independent of pylon tilt angle. This will be modified for Phase II simulations.
- Rolling Moment Coefficient (Figures III-15 and III-16). The mathematical model defines wing-pylon dihedral effects accurately up to  $\pm 10$  degrees sideslip angle. Beyond this range, the dihedral stability of wing-pylon is maintained constant. (The test data do not show zero rolling moment at zero yaw because of model asymmetry.)

Figure III-19 compares the mathematical model aileron control power variation with angle of attack with test data. The decrease in roll control power at high angles of attack is accurately represented.

#### C. Rotor Wake - Airframe Aerodynamic Interaction

The rotor wake induced velocity at the horizontal tail and the unstable rolling moment in-ground effect are represented using a polynomial fit of test data. The curve fit and test data are compared in Figure II-16.

The effect of the rotor downwash on wing lift is calculated using the analytical model discussed in Section II.D. This effect is compared to test data (from a model of a similar aircraft) in Figure III-20. One to one agreement is not expected because of differences in the configuration of the Model 301 and that of the model. What is significant is the trend in down load with airspeed which is quite good.

#### D. Stability Derivatives

Stability derivatives calculated with the mathematical model are compared with those calculated with program C81 in Table III-1. Significant differences between the mathematical model and C81 derivatives are evident but are not fully understood. One of the reasons for the difference in the longitudinal derivatives may be explained by the fact that C81 airframe inputs are linear below the stall angle of attack, whereas the mathematical model accounts for the variation in the coefficients with angle of attack. The lateral-directional derivatives are in better agreement since both the programs use stability derivatives for airframe

inputs; the difference in derivatives is primarily caused by differences in the rotor mathematical models.

However, the differences in computed derivatives of the two programs are small enough that the resulting dynamic stability characteristics are similar, as discussed later in Section IV.F.

**E. Static Stability**

Figure III-21 compares the level flight trim longitudinal stick position, blade root collective pitch, and fuselage pitch attitude calculated with the mathematical model with those calculated with program C81. These are in good agreement with the exception of the longitudinal stick position at airspeeds below 50 knots. This difference is caused by the difference in the wing download representation in the two programs. In C81 the center of pressure of the download does not vary with airspeed; in the simulator mathematical model the download center of pressure moves aft with increasing airspeed.

**F. Dynamic Stability**

Flight mode characteristics calculated with the mathematical model are compared with those calculated by program C81 in Table III-2. Note that characteristics for the mathematical model were determined both from response time histories and by solving the characteristic equations using the derivatives given in Table III-1. The differences in the flight mode characteristics (particularly for the phugoid mode) may be caused by the rotor rpm degree of freedom being active when the response time histories were obtained. A higher short period damping predicted by program C81 is due to a higher estimate of the wing downwash lag derivative  $C_{m\dot{\alpha}}$ . Differences in short period frequencies are primarily due to inaccuracies in the rotor mathematical model computation of inplane forces resulting from a neglect of stall effects.

The flight mode characteristics computed by the two programs are compared with the MIL-F-83300 level 1 requirements in Figures III-22 and III-23. Characteristics computed from these two programs are such that they meet the level 1 requirements and would be expected to produce similar pilot evaluation.

**G. Control Sensitivity and Damping**

Figures III-24 and III-25 compare the control sensitivity calculated with the mathematical model with that calculated with program C81. Agreement is good except for the control power at  $\alpha_{\text{mast}} = 30^\circ$ . Investigation of this difference revealed that the C81 data is in error at this mast angle; the mathematical model reflects the correct sensitivity.

Comparison of aircraft damping characteristics as shown in Figure III-26 is good throughout the speed and pylon tilt range.

TABLE III-1 COMPARISON OF STABILITY DERIVATIVES WITH THOSE COMPUTED BY PROGRAM C81

G. W. = 13,000 LB., AFT CG, SEA LEVEL STANDARD

FLIGHT CONDITION DERIVATIVE	HELICOPTER MODE $\alpha_M=90^\circ$ FLAPS 40/25 565 RPM 120 KNOTS		CONVERSION MODE $\alpha_M=60^\circ$ FLAPS 40/25 565 RPM 120 KNOTS		CONVERSION MODE $\alpha_M=30^\circ$ FLAPS 40/25 565 RPM 140 KNOTS		AIRPLANE MODF. $0/0^\circ$ FLAPS 0/0 458 RPM 260 KNOTS	
	(1)	(2)	(1)	(2)	(1)	(2)	(1)	(2)
$-C_{XU}$	.865	.95	1.64	1.59	3.19	3.05	1.22	.125
$-C_{ZU}$	1.34	.83	.54	.36	.82	.61	1.46	.61
$-C_{MU}$	.29	-.49	-.64	-.67	-1.26	-.99	-.887	-.058
$-C_{X\alpha}$	.605	.675	-2.72	-1.87	-2.06	-1.80	-.201	-.246
$-C_{Z\alpha}$	9.105	8.45	8.30	8.01	6.59	6.35	6.47	6.47
$-C_{M\alpha}$	3.5	1.74	2.67	2.61	1.85	2.36	1.32	1.17
$-C_{Xq}$	-.013	-1.21	5.33	1.17	5.16	1.42	-.551	-.41
$-C_{Mq}$	39.9	37.4	33.7	39.2	29.2	35.8	29.8	33.8
$-C_{Y\beta}$	1.77	1.95	1.8	1.61	1.87	1.62	1.83	1.87
$-C_{L\beta}$	.365	.305	.285	.106	.289	.202	.318	.312
$-C_{N\beta}$	-.371	-.474	-.38	-.46	-.303	-.34	-.242	-.305
$-C_{Yp}$	.45	.344	-.13	-.72	-.517	-.366	.168	.0064
$-C_{Lp}$	3.13	3.2	2.7	3.0	2.06	1.96	1.34	1.28

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TABLE III-1 COMPARISON OF STABILITY DERIVATIVES WITH HOSE COMPUTED BY PROGRAM C81 (CONTINUED)

G. W. = 13,000 LB., AFT CG, SEA LEVEL STANDARD

FLIGHT CONDITION DERIVATIVE	HELICOPTER MODE $\alpha_M=90^\circ$ F 40/25		CONVERSION MODE $\alpha_M=60^\circ$ FLAPS 40/25		CONVERSION MODE $\alpha_M=30^\circ$ FLAPS 40/25		AIRPLANE MODE $\alpha_M=0^\circ$ , FLAPS 0°/0° 458 RPM 260 KNOTS	
	565 RPM (1)	120 KNOTS (2)	565 RPM (1)	120 KNOTS (2)	565 RPM (1)	140 KNOTS (2)	458 RPM (1)	260 KNOTS (2)
$-C_{N_P}$	-0.635	-0.276	0.45	0.74	0.734	0.94	0.19	0.26
$-C_{L_P}$	-0.936	-0.44	0.002	0.384	0.234	0.664	0.414	0.388
$-C_{N_z}$	1.13	1.03	1.015	1.206	1.54	1.86	1.22	1.24
$C_{L_o}$	1.48	1.48	1.48	1.48	1.085	1.085	0.314	0.314

(1) Mathematical Model - Constant RPM and Collective Pitch

(2) C81 Program - Constant RPM, for  $\alpha_M < 20^\circ$  Collective Pitch Adjusted to Maintain Constant Torque

TABLE III-2 COMPARISON OF FLIGHT MODE CHARACTERISTICS WITH THOSE PREDICTED BY PROGRAM C81

G. W. = 13,000 LB., AFT CG, SEA LEVEL STANDARD

FLIGHT MODE	HELICOPTER MODE $\alpha_M=90^\circ, F_x=40^\circ/25^\circ$ 565 RPM 120 KNOTS	CONVERSION MODE $\alpha_M=60^\circ, F_x=40^\circ/25^\circ$ 565 RPM 120 KNOTS	CONVERSION MODE $\alpha_M=30^\circ, F_x=40^\circ/25^\circ$ 565 RPM 140 KNOTS	AIRPLANE MODE $\alpha_M=0^\circ, F_x=0^\circ/0^\circ$ 458 RPM 260 KNOTS
SHORT PERIOD	(1) P=2.0 $t_{1/2}=1.0$	P=3.2 $t_{1/2}<1/2$	P=2.9 $t_{1/2}=0.6$	P=1.6 $t_{1/2}=0.35$
	(2) P=1.9 $t_{1/2}=.48$	P=2.2 $t_{1/2}=0.53$	P=2.4 $t_{1/2}=0.55$	P=1.5 $t_{1/2}=0.31$
	(3) P=4.6 $t_{1/2}=0.35$	P=2.2 $t_{1/2}=0.37$	P=2.0 $t_{1/2}=0.35$	P=1.3 $t_{1/2}=0.26$
PHUGOID	(1) P > 30 $t_{1/2}=7.5$	P=29.5 $t_{1/2}=25.0$	P=28.5 $t_{1/2}=28.0$	P=27.7 $t_{1/2}=30.6$
	(2) P=51.6 $t_{1/2}=20.4$	P=23.2 $t_{1/2}=9.5$	P=20.0 $t_{1/2}=5.1$	P=45.0 $t_{1/2}=56.0$
	(3) P=32.0 $t_{1/2}=14.0$	P=41.0 $t_{1/2}=8.0$	P=46.0 $t_{1/2}=4.0$	P=56.0 $t_{1/2}=48.0$
DUTCH ROLL	(1) P=4.8 $t_{1/2}=3.0$	P=4.0 $t_{1/2}=3.6$	P=4.0 $t_{1/2}=3.5$	P=2.4 $t_{1/2}=1.6$
	(2) P=4.6 $t_{1/2}=2.24$	P=4.1 $t_{1/2}=3.4$	P=1.95 $t_{1/2}=.25$	P=2.6 $t_{1/2}=1.41$
	(3) P=4.0 $t_{1/2}=2.9$	P=3.9 $t_{1/2}=2.2$	P=4.0 $t_{1/2}=1.8$	P=2.2 $t_{1/2}=1.0$
SPIRAL	(1) $t_{1/2}=16$	$t_{1/2}=16.5$	$t_{1/2}=7.4$	$t_{1/2}=9.8$
	(3) $t_{1/2}=17.5$	$t_{1/2}=16.4$	$t_{1/2}=6.3$	$t_{1/2}=8.3$
ROLL CONVERGENCE	(1) $T_R=0.6$	$T_R=1.5$	$T_R=0.8$	$T_R=0.52$
	(2) $T_R=0.63$	$T_R=0.61$	$T_R=0.63$	$T_R=0.57$
	(3) $T_R=0.44$	$T_R=0.46$	$T_R=0.52$	$T_R=0.58$

NOTE: Period, P, time to one half amplitude,  $t_{1/2}$ , are in seconds and time constant  $\tau$  is per second.

- (1) Mathematical Model - Characteristics extracted from time histories of airplane response to control inputs.
- (2) - Characteristics obtained by solving characteristic equation using stability derivatives determined from the Mathematical Model (Table III-1)
- (3) C81 Program - Characteristics obtained by solving characteristic equation using stability derivatives (data from Section 6, BHC Report 301-199-002).

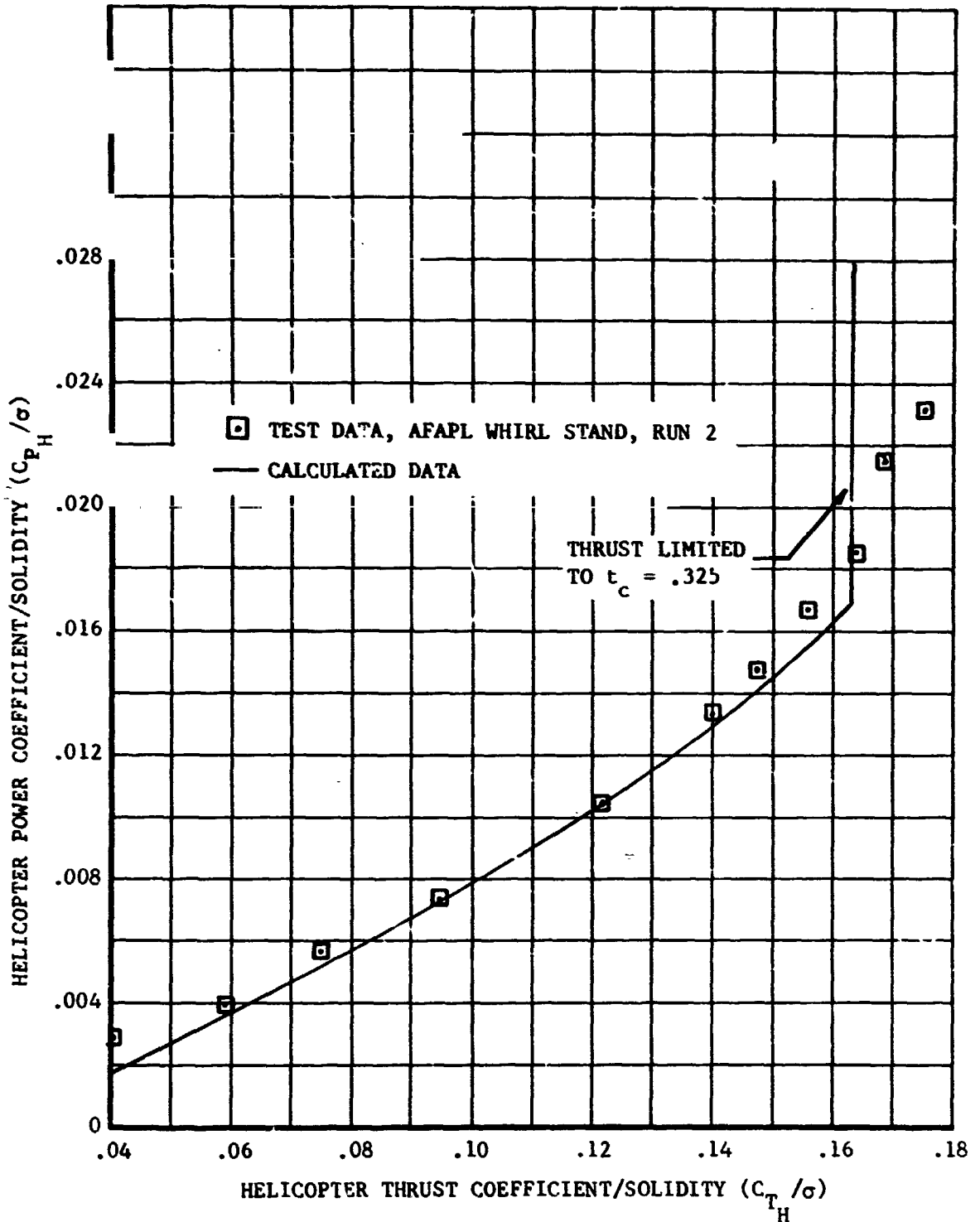


Figure III-1. Rotor Performance Correlation - Hover

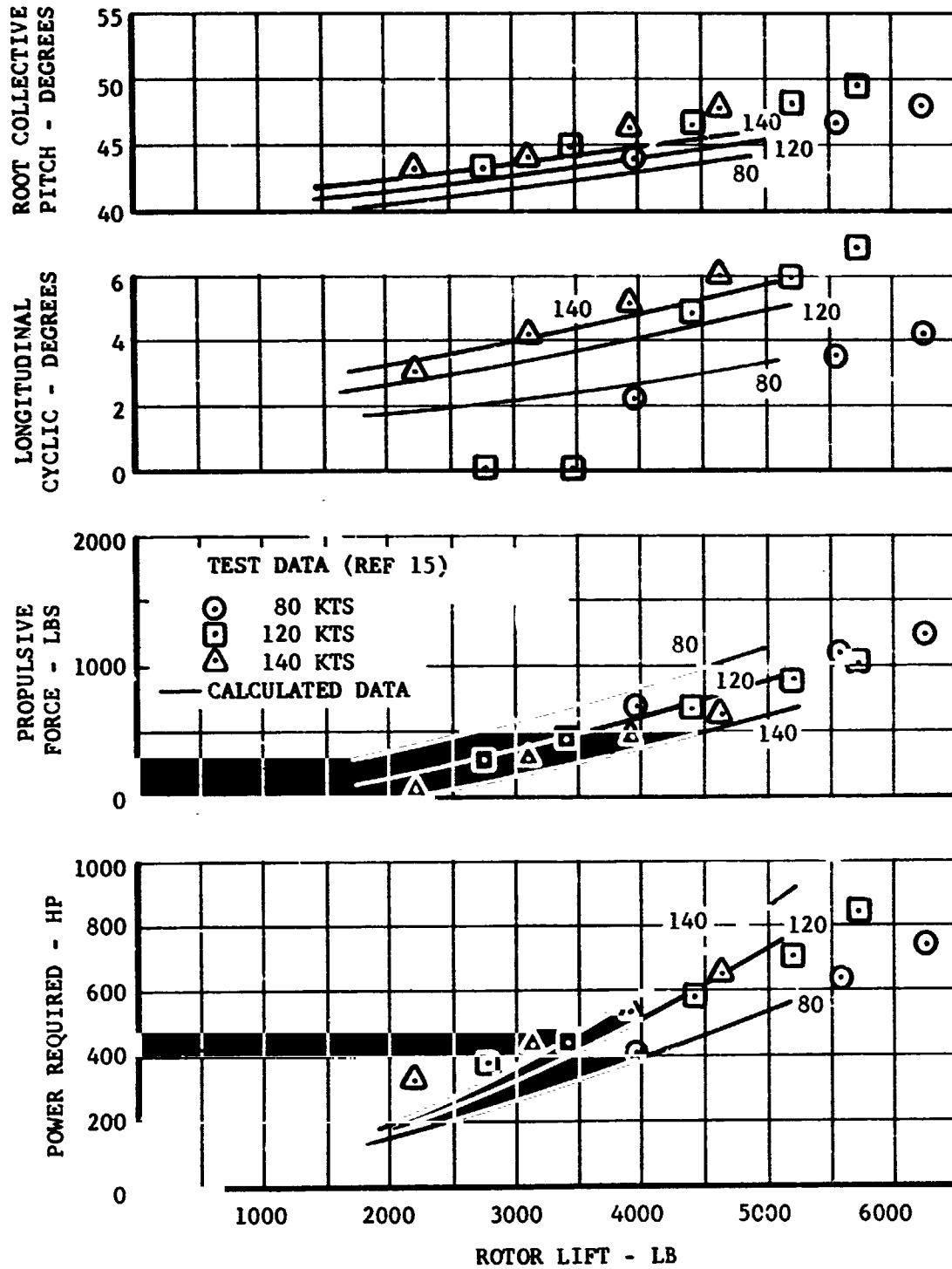


Figure III-2. Rotor Performance Correlation -  $\alpha_m = 75^\circ$

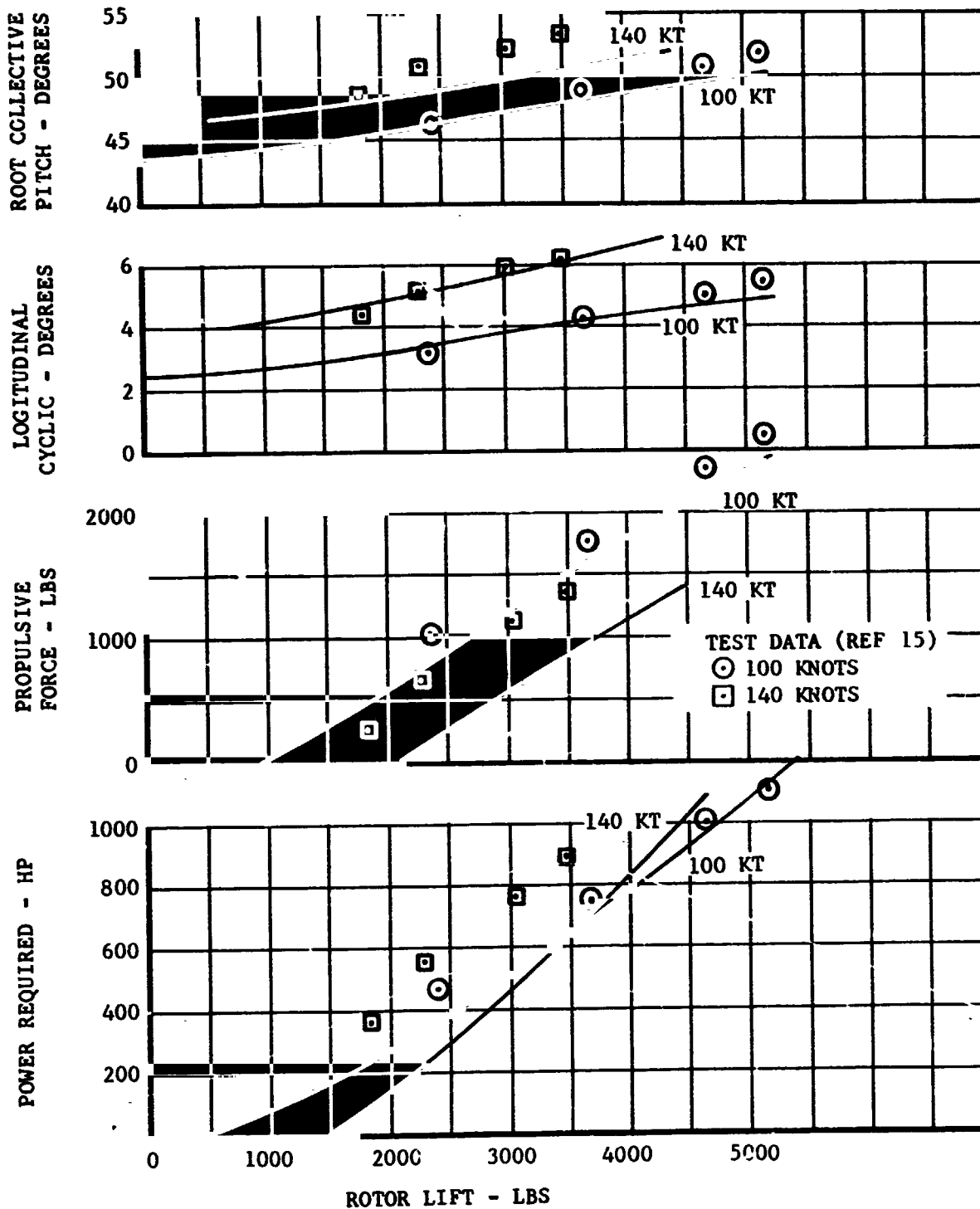


Figure III-3. Rotor Performance Correlation -  $\alpha_m = 60^\circ$



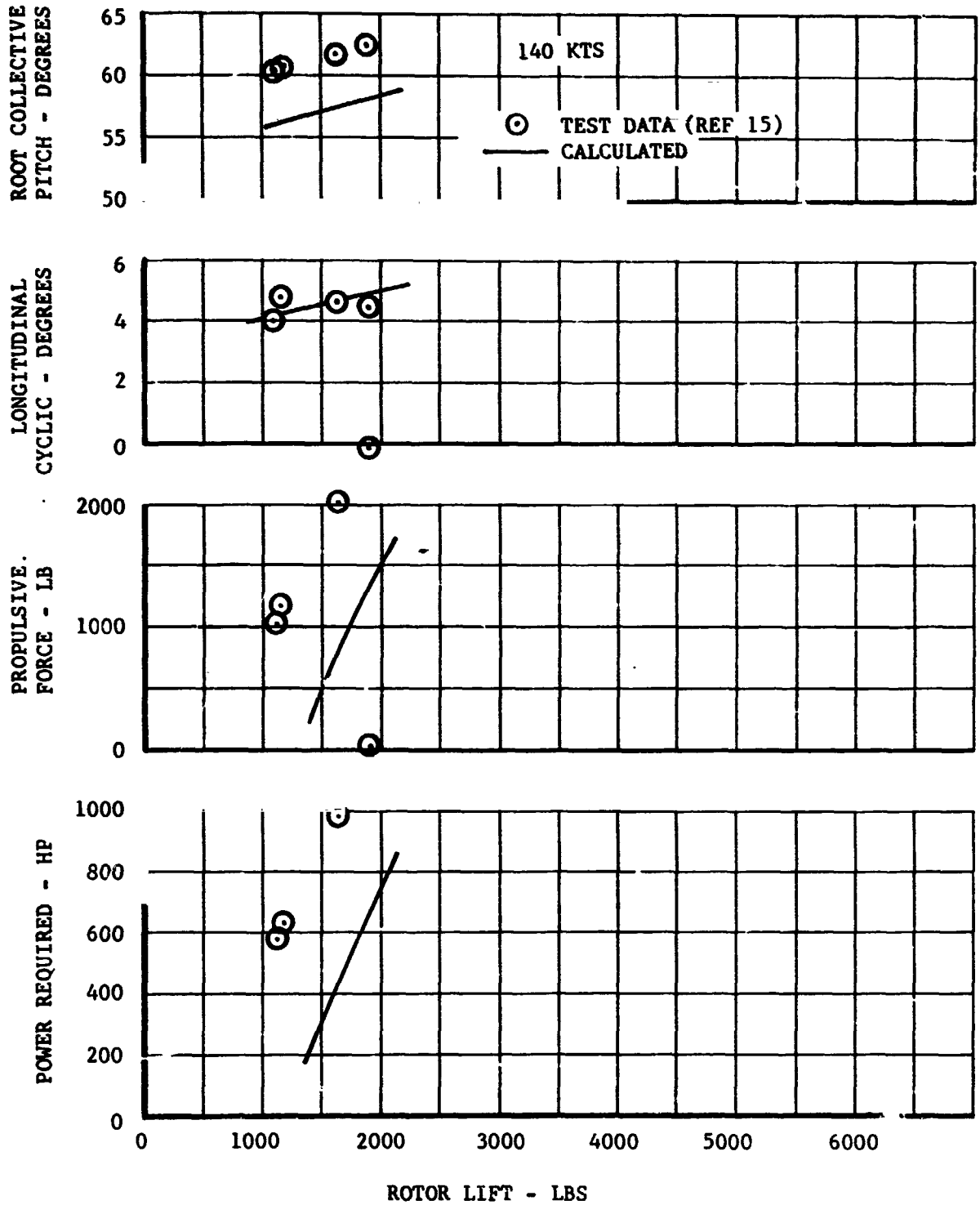


Figure III-4. Rotor Performance Correlation -  $\alpha_m = 30^\circ$

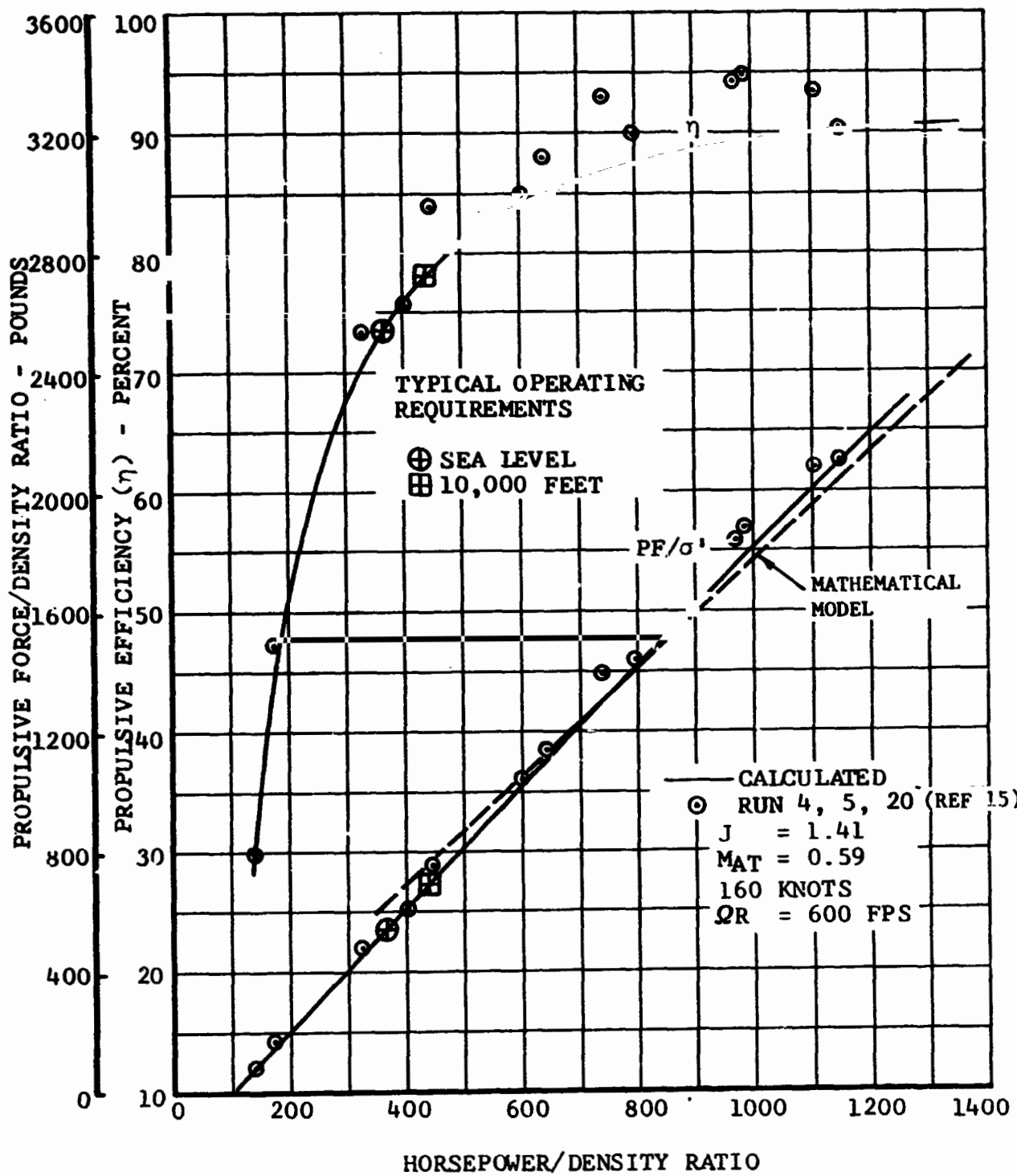


Figure III-5. Rotor Performance Correlation -  $\alpha_m = 0^\circ$

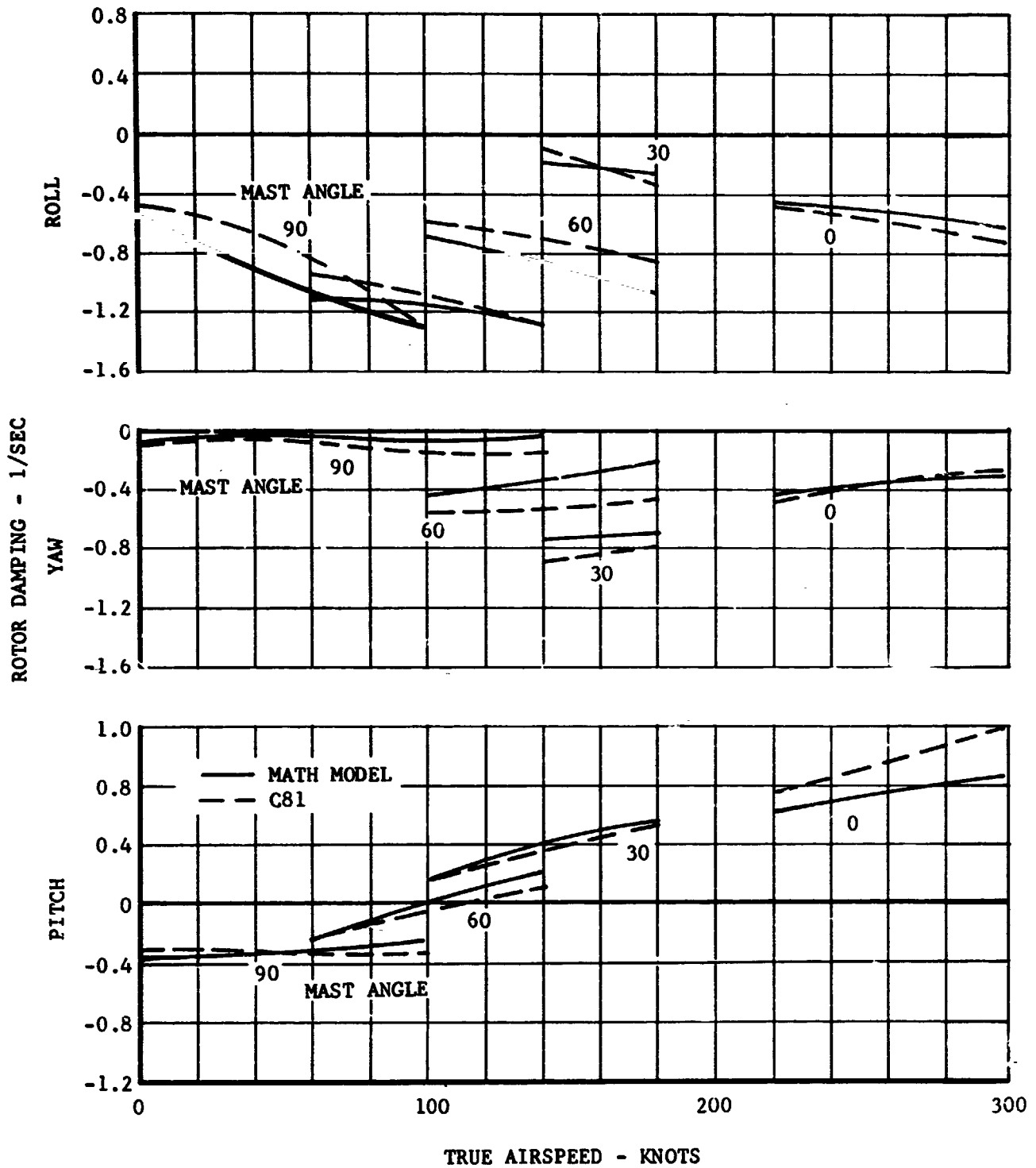


Figure III-6. Rotor Rate Derivative Correlation

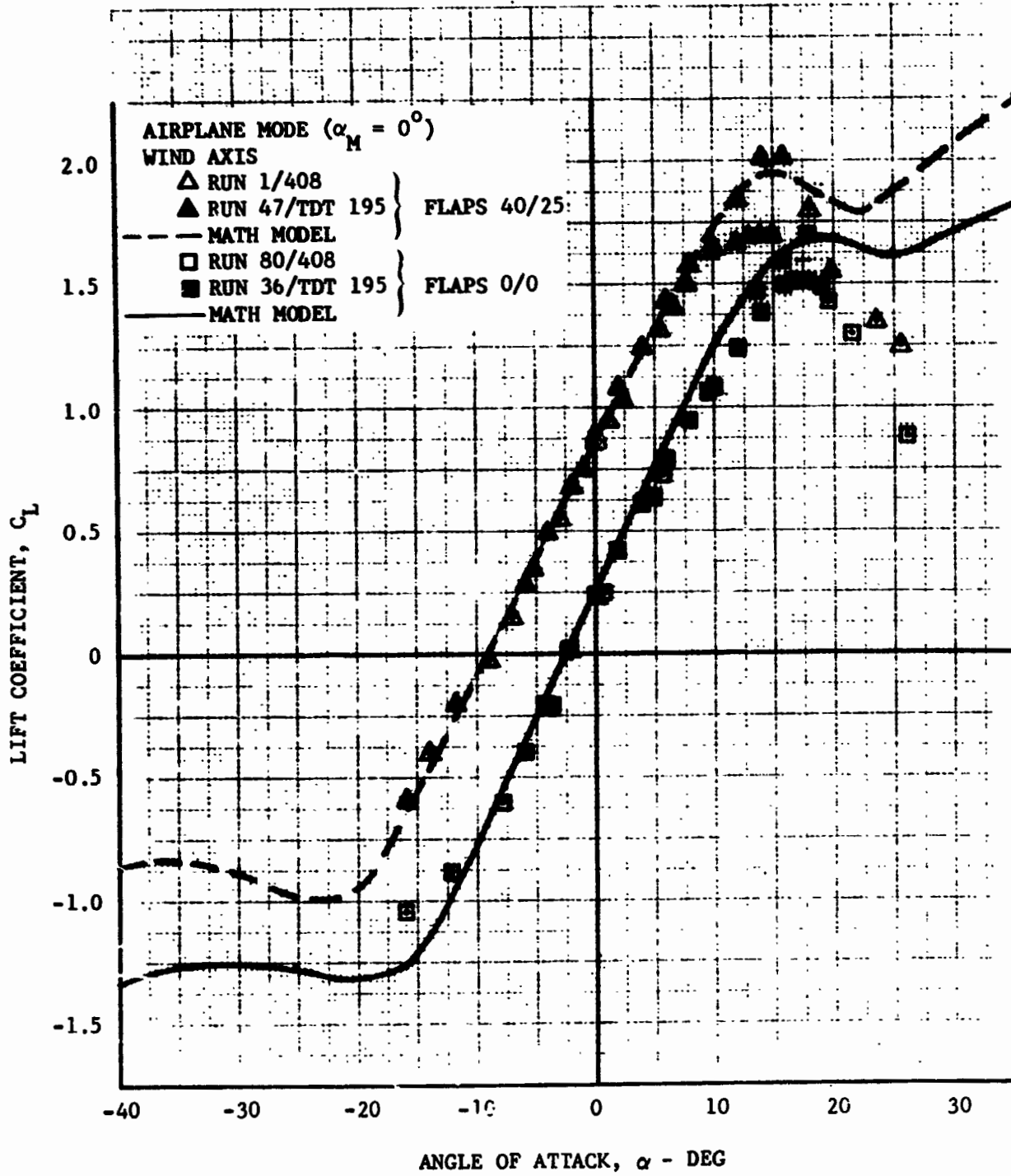


Figure III-7. Airplane Mode Lift Coefficient Correlation

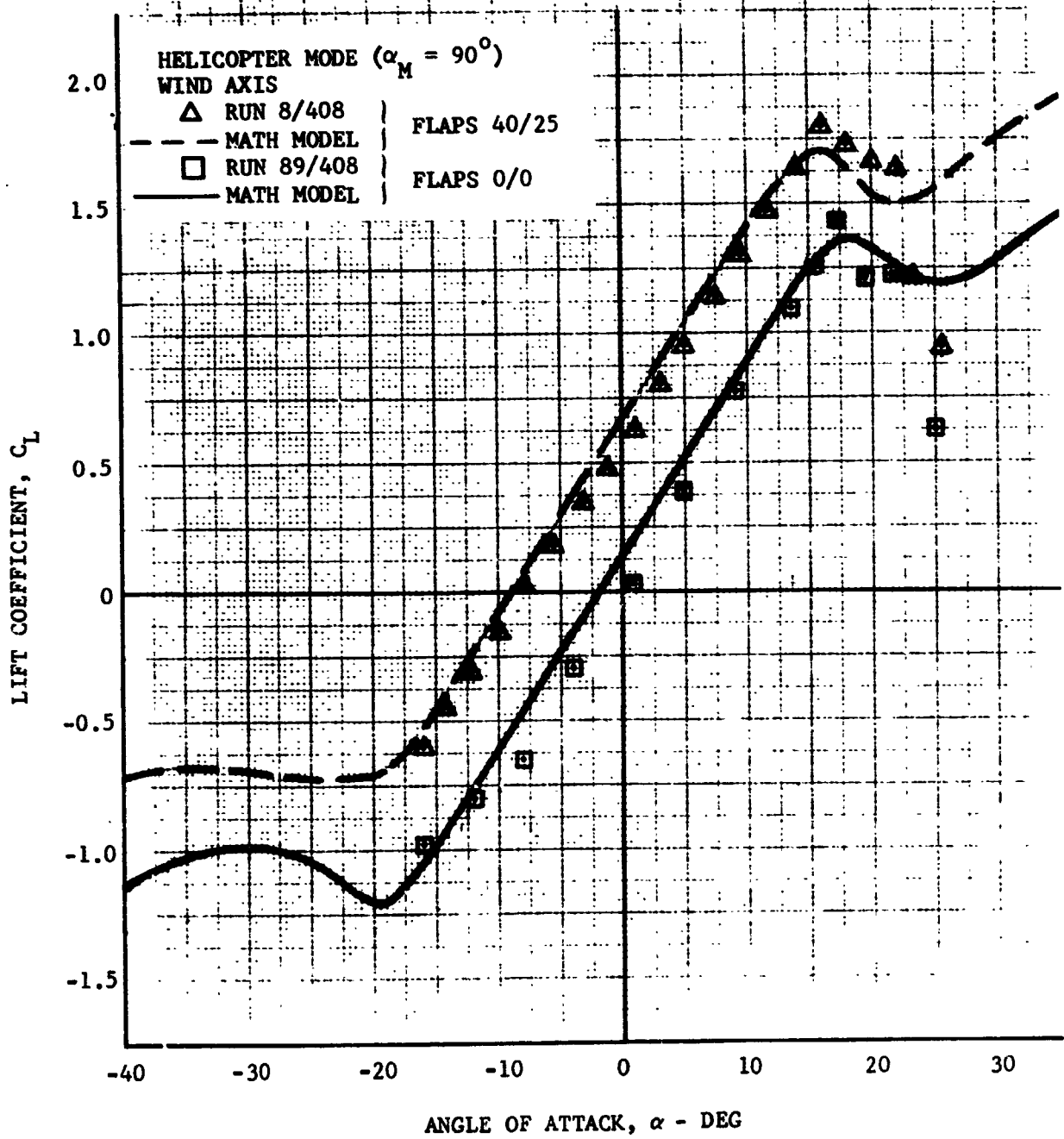


Figure III-8. Helicopter Mode Lift Coefficient Correlation

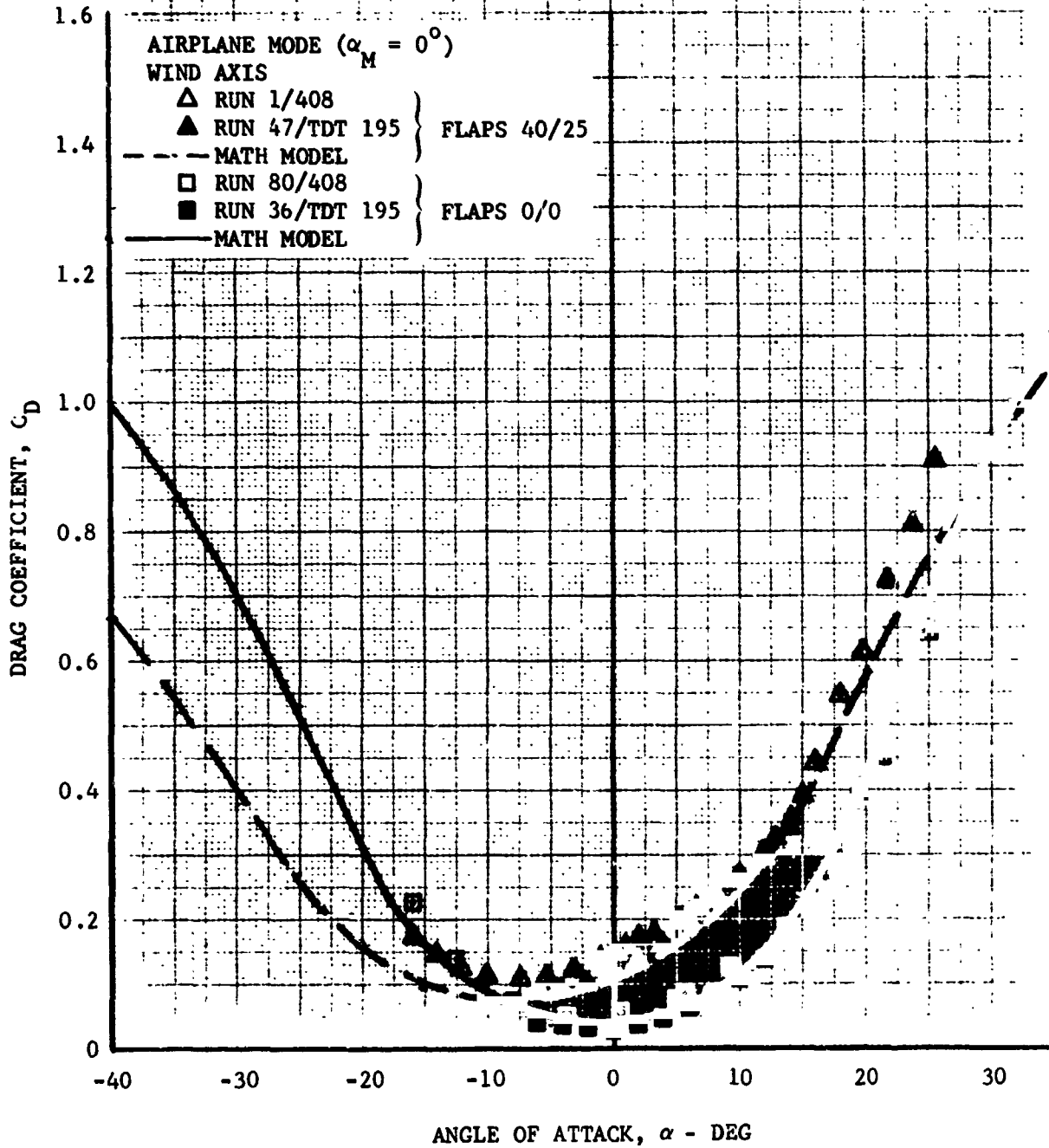


Figure III-9. Airplane Mode Drag Coefficient Correlation

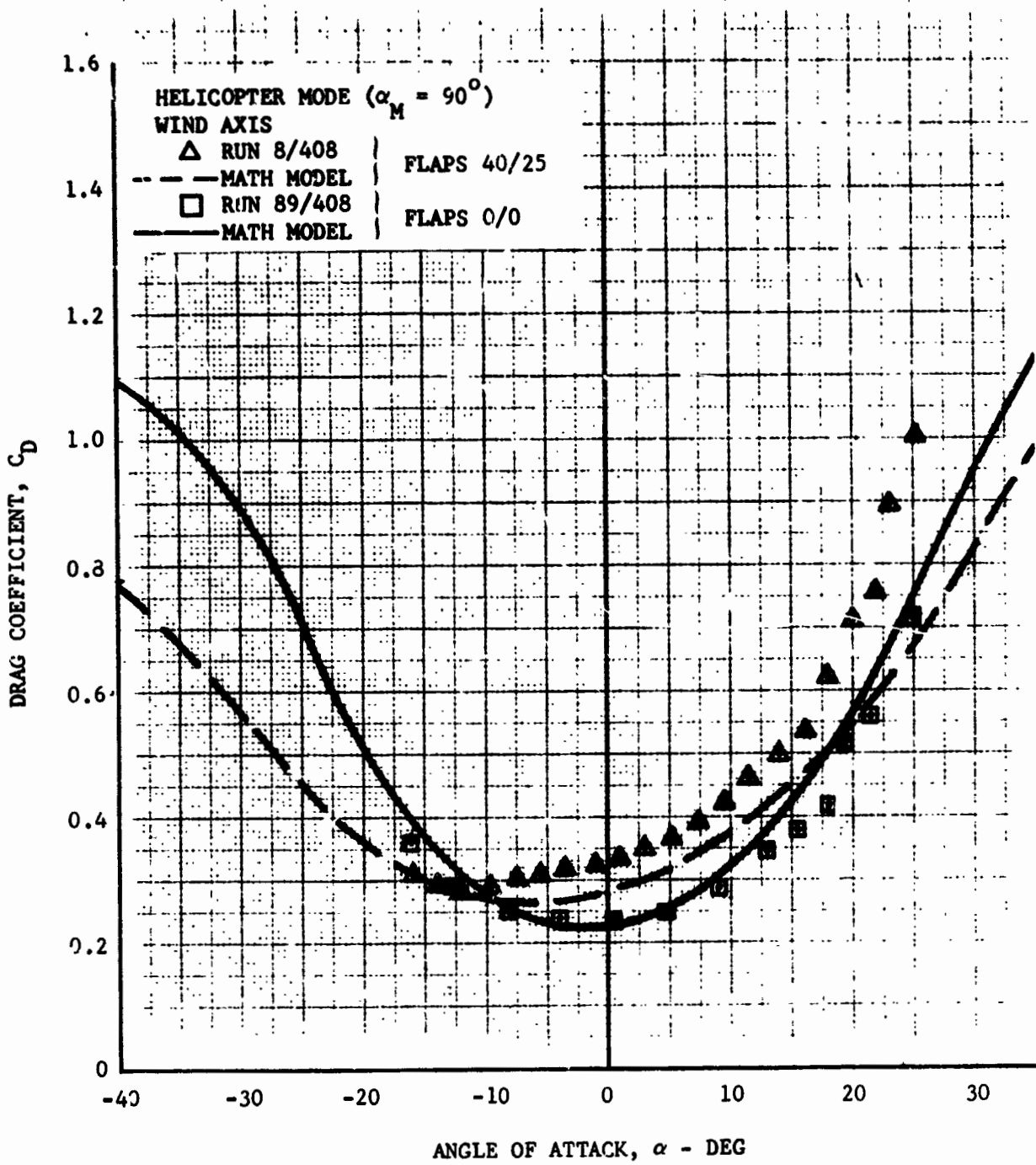


Figure III-10. Helicopter Mode Drag Coefficient Correlation

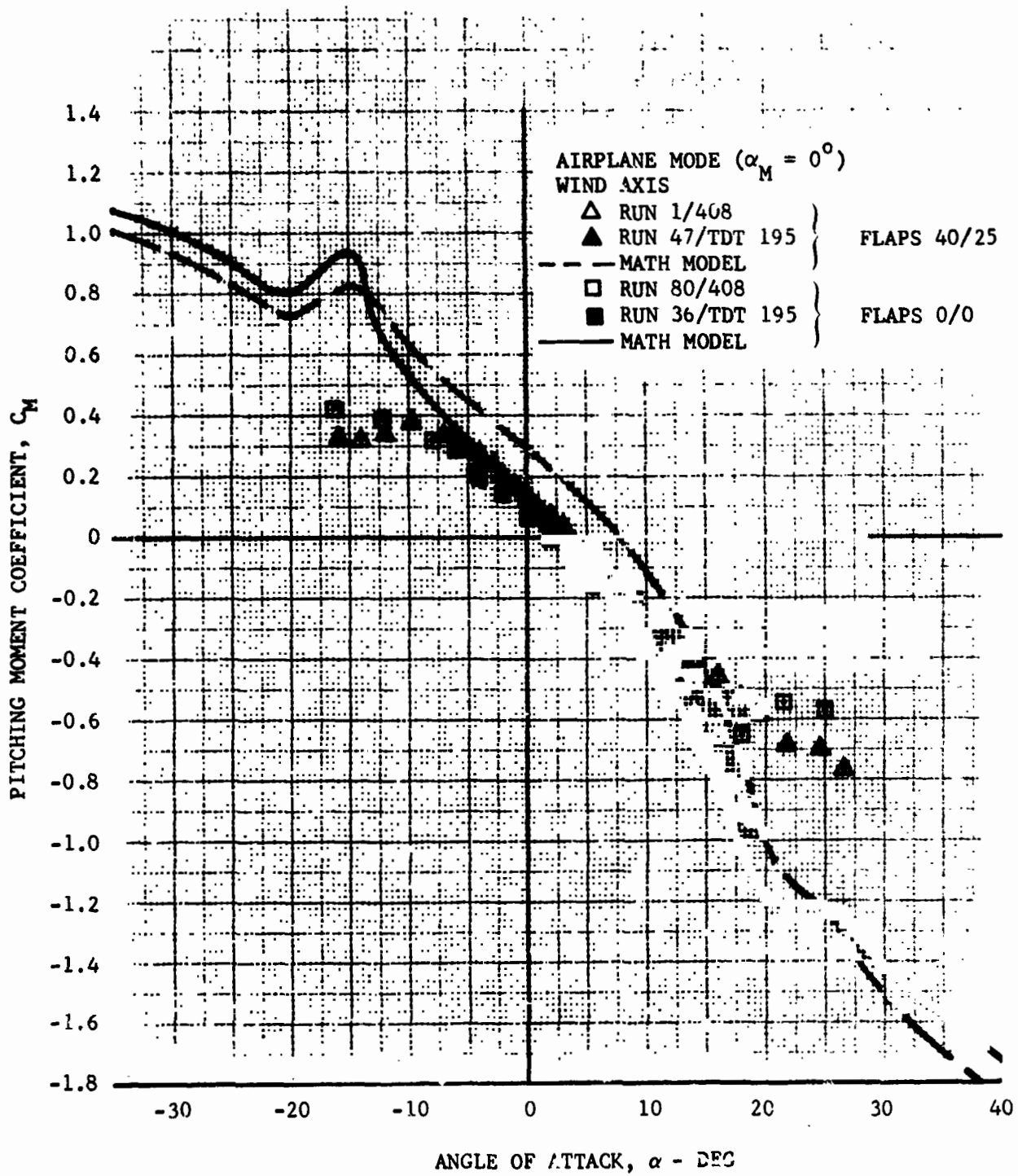


Figure III-11. Airplane Mode Pitching Moment Correlation



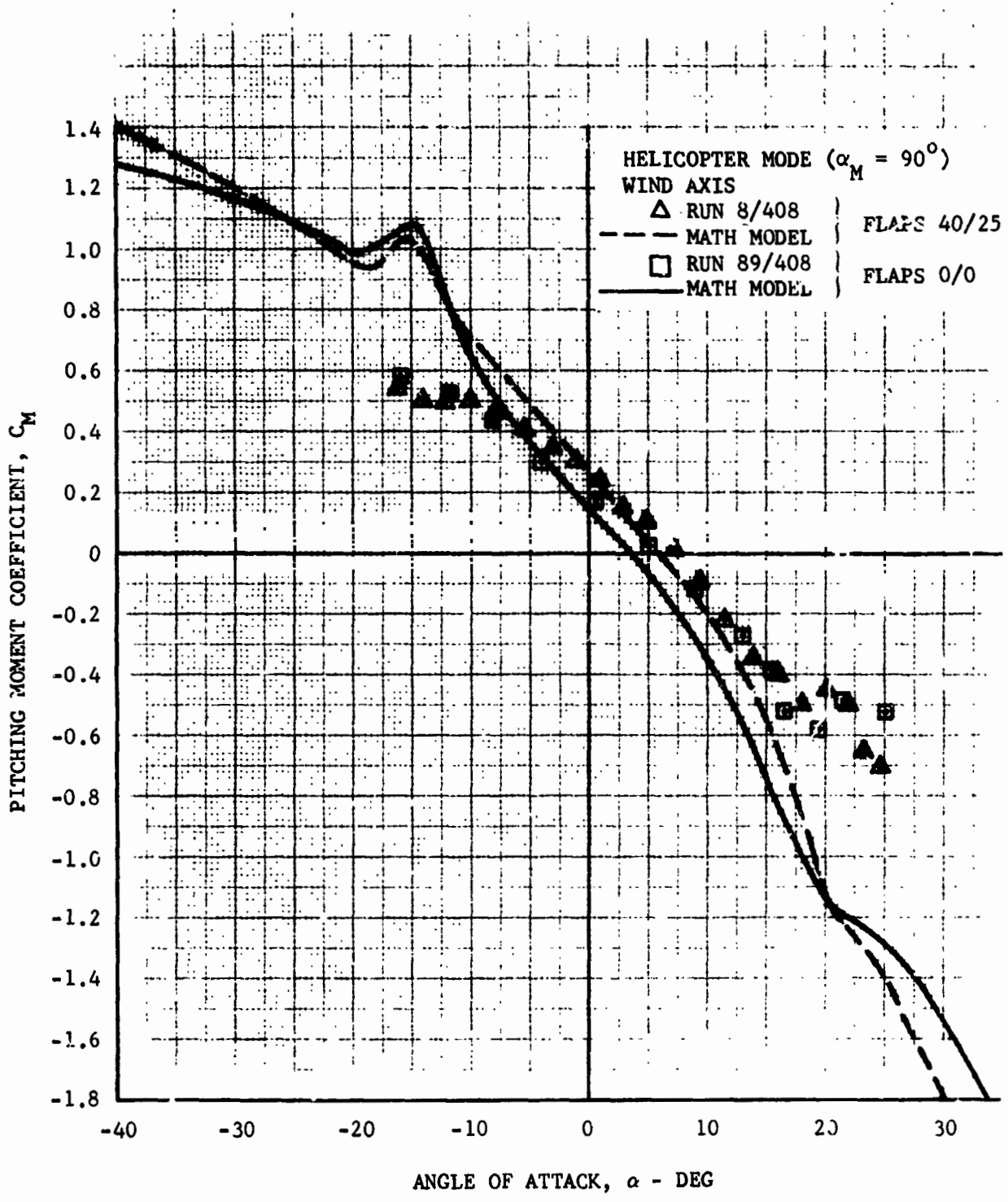


Figure III-12. Helicopter Mode Pitching Moment Correlation

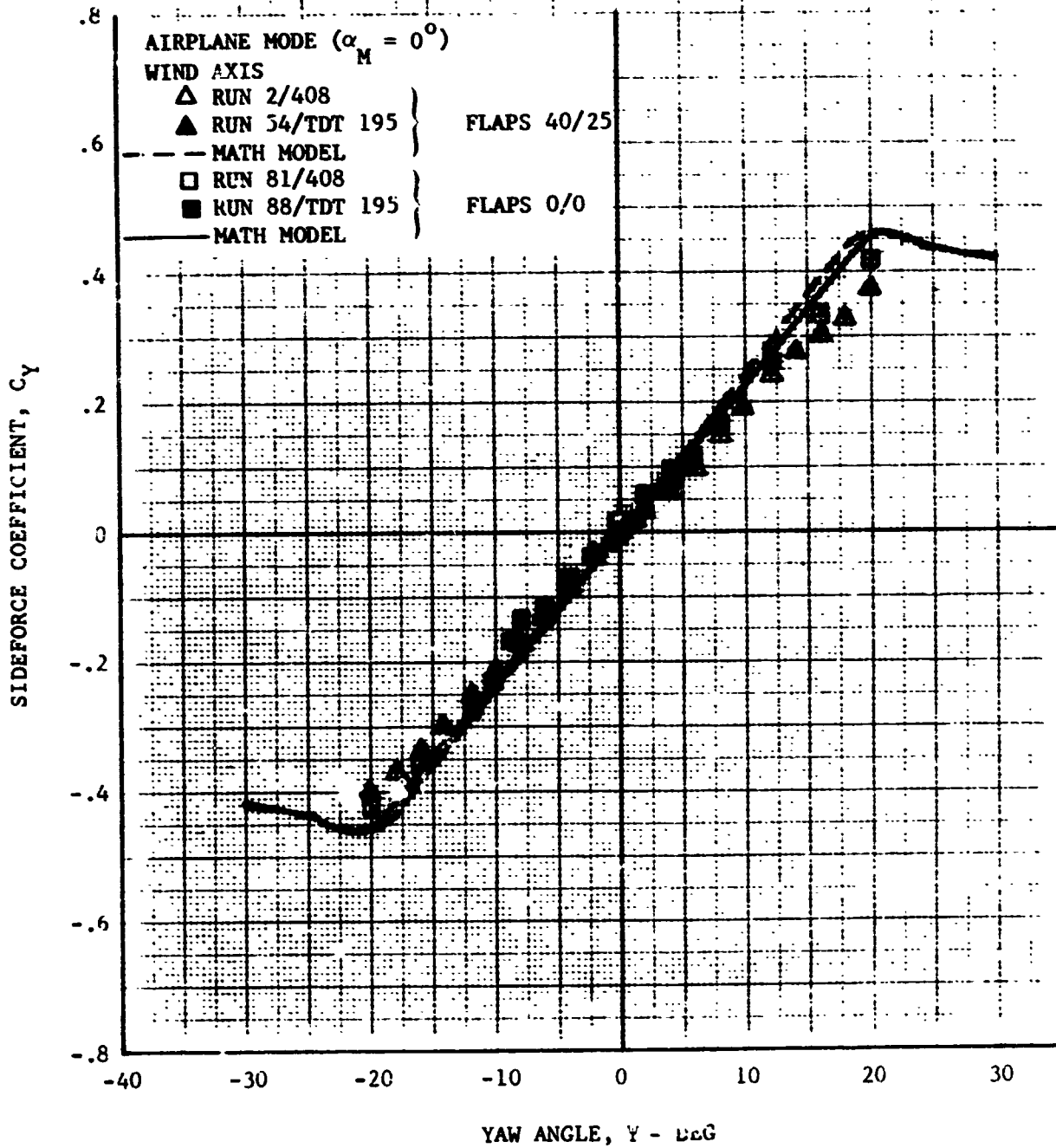


Figure III-13. Airplane Mode Sideforce Coefficient Correlation

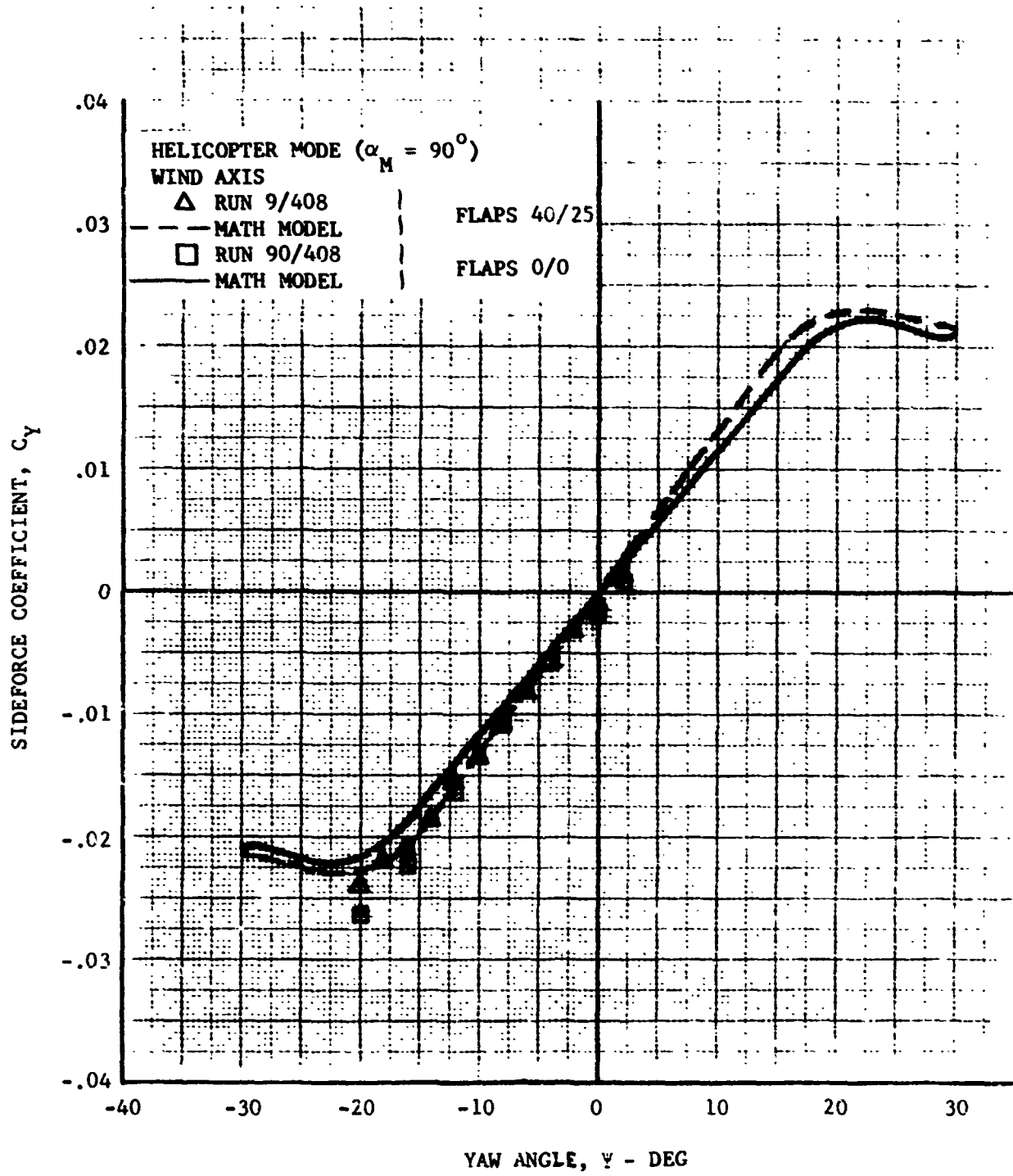


Figure III-14. Helicopter Mode Sideforce Coefficient Correlation

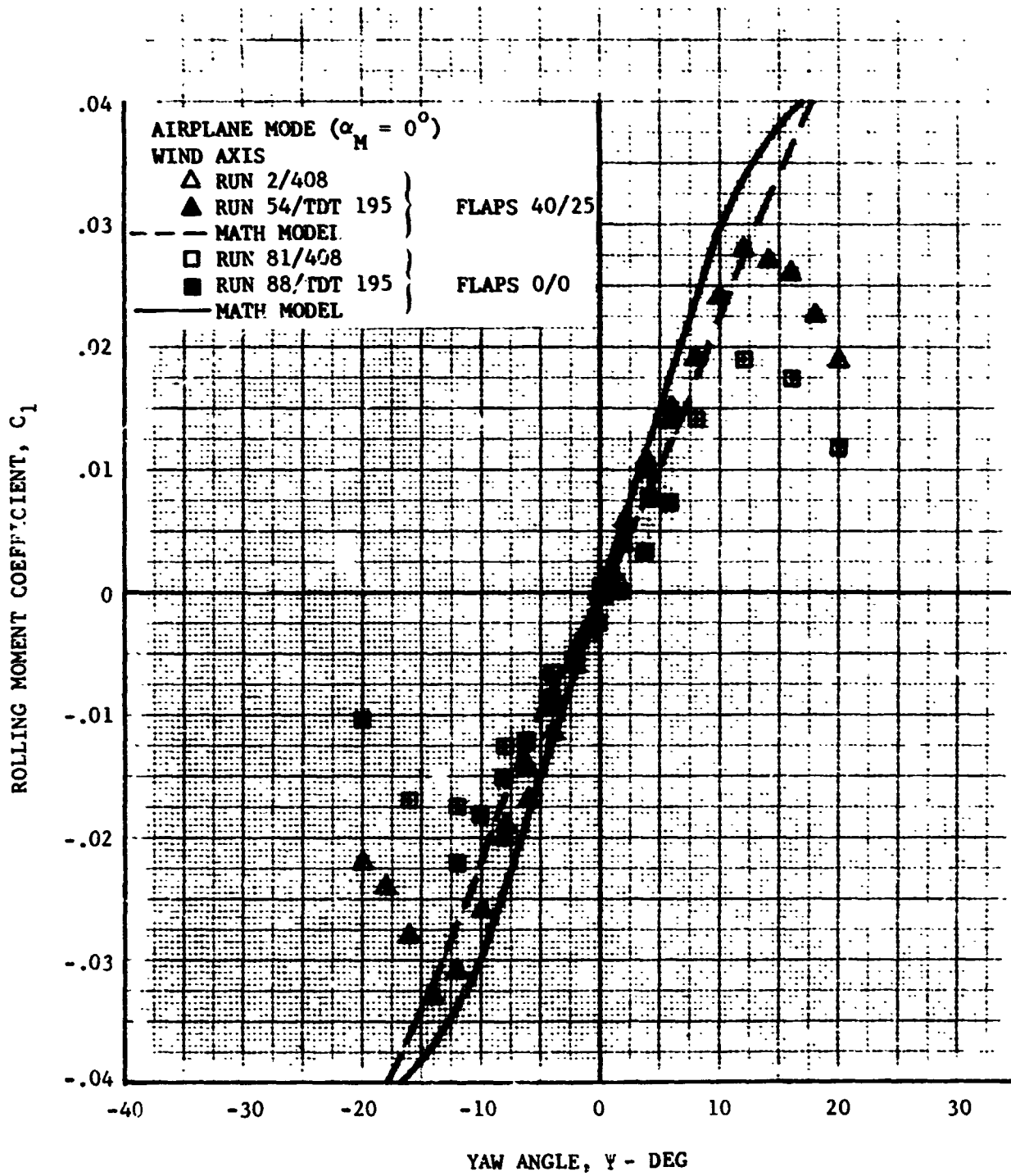


Figure III-15. Airplane Mode Rolling Moment Coefficient Correlation

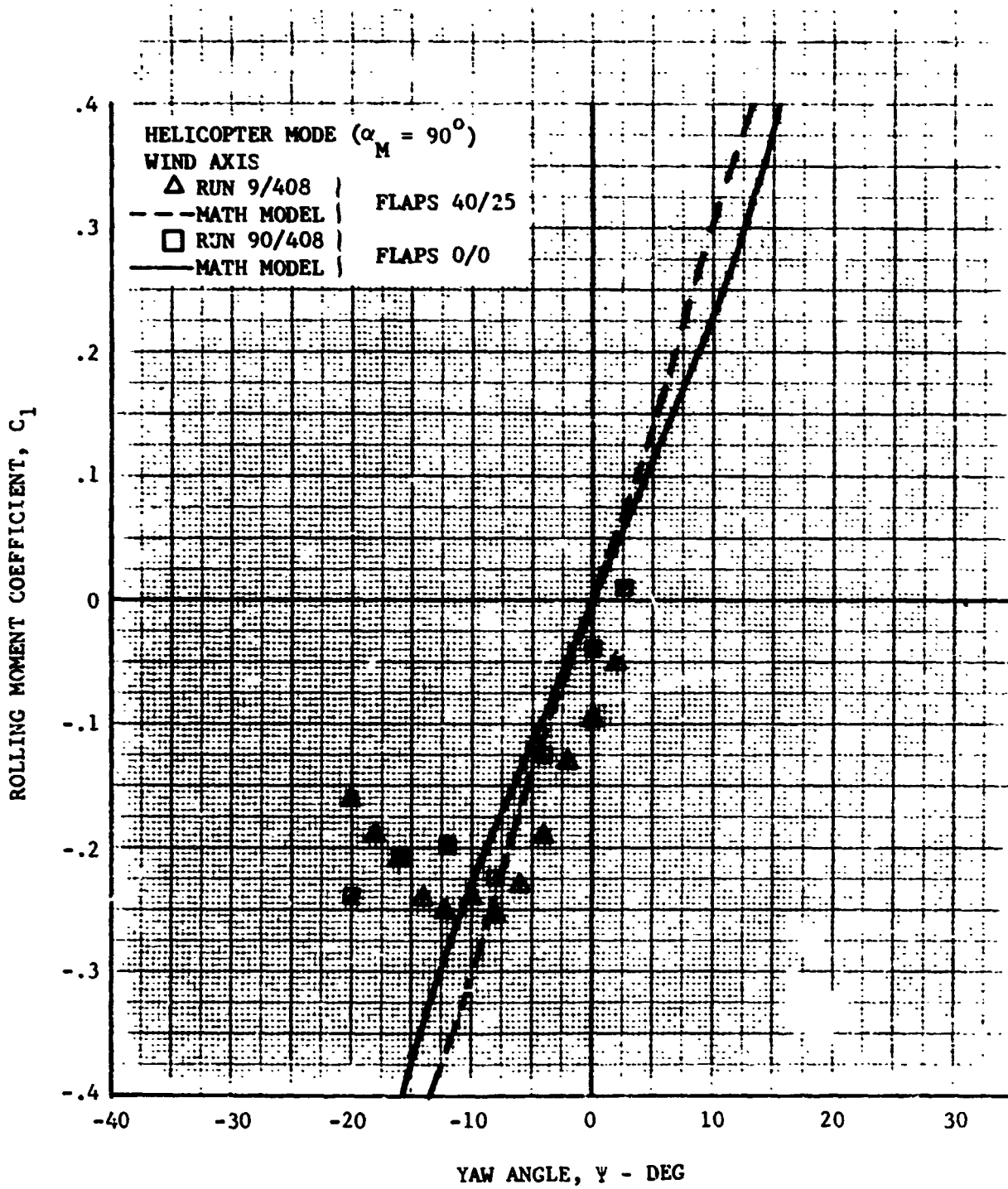


Figure III-16. Helicopter Mode Rolling Moment Coefficient Correlation

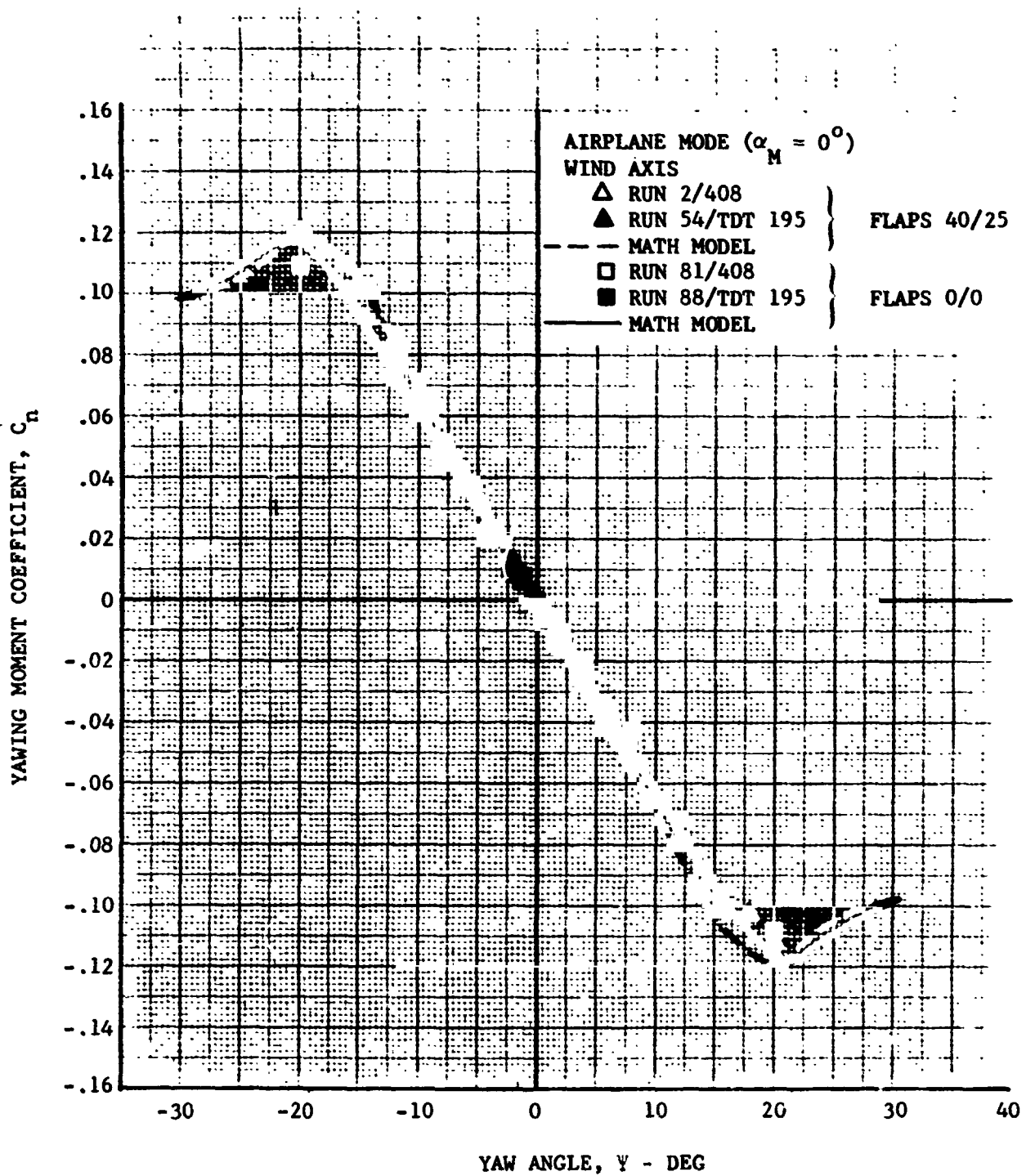


Figure III-17. Airplane Mode Yawing Moment Coefficient Correlation

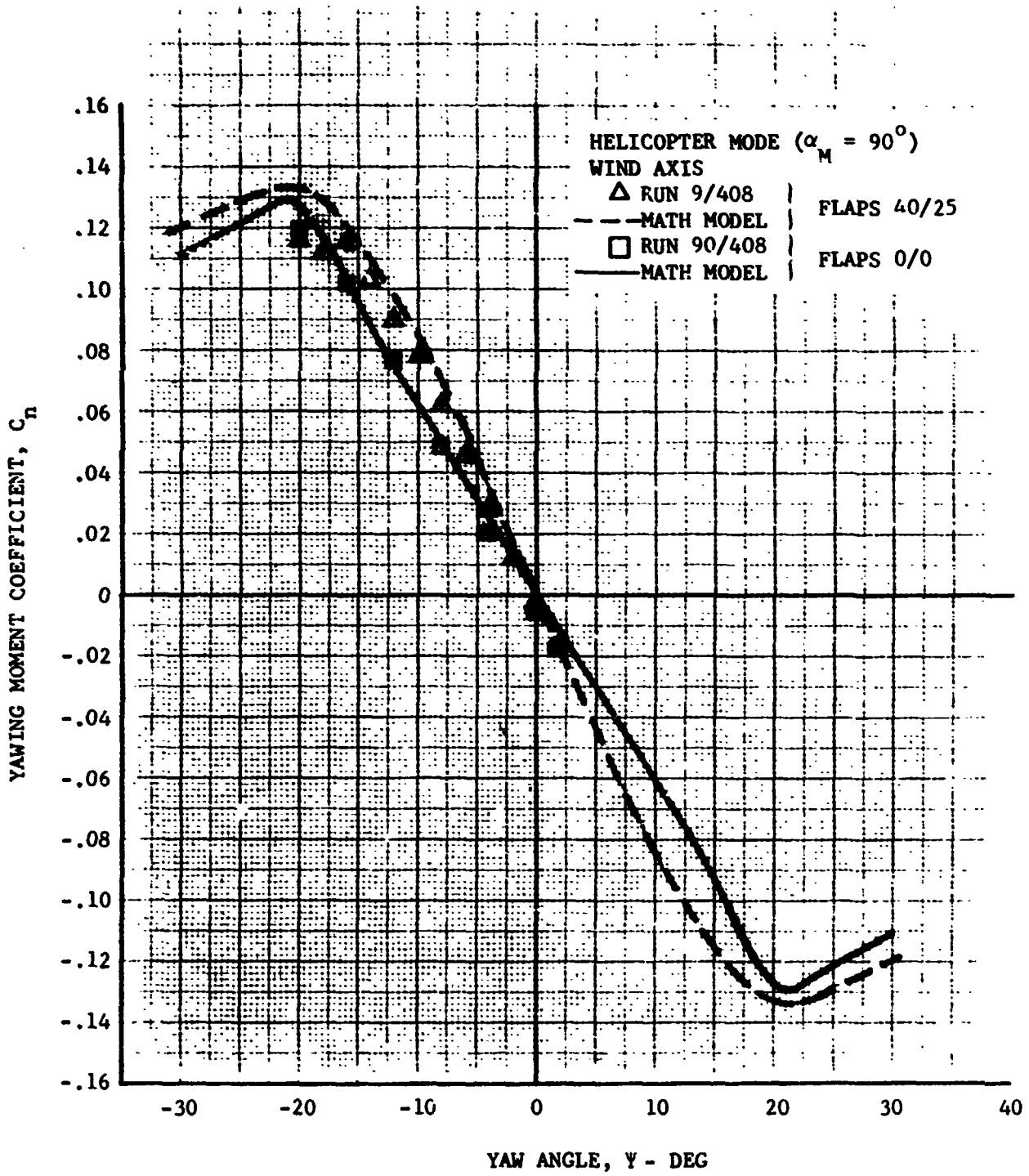


Figure III-18. Helicopter Mode Yawing Moment Coefficient Correlation

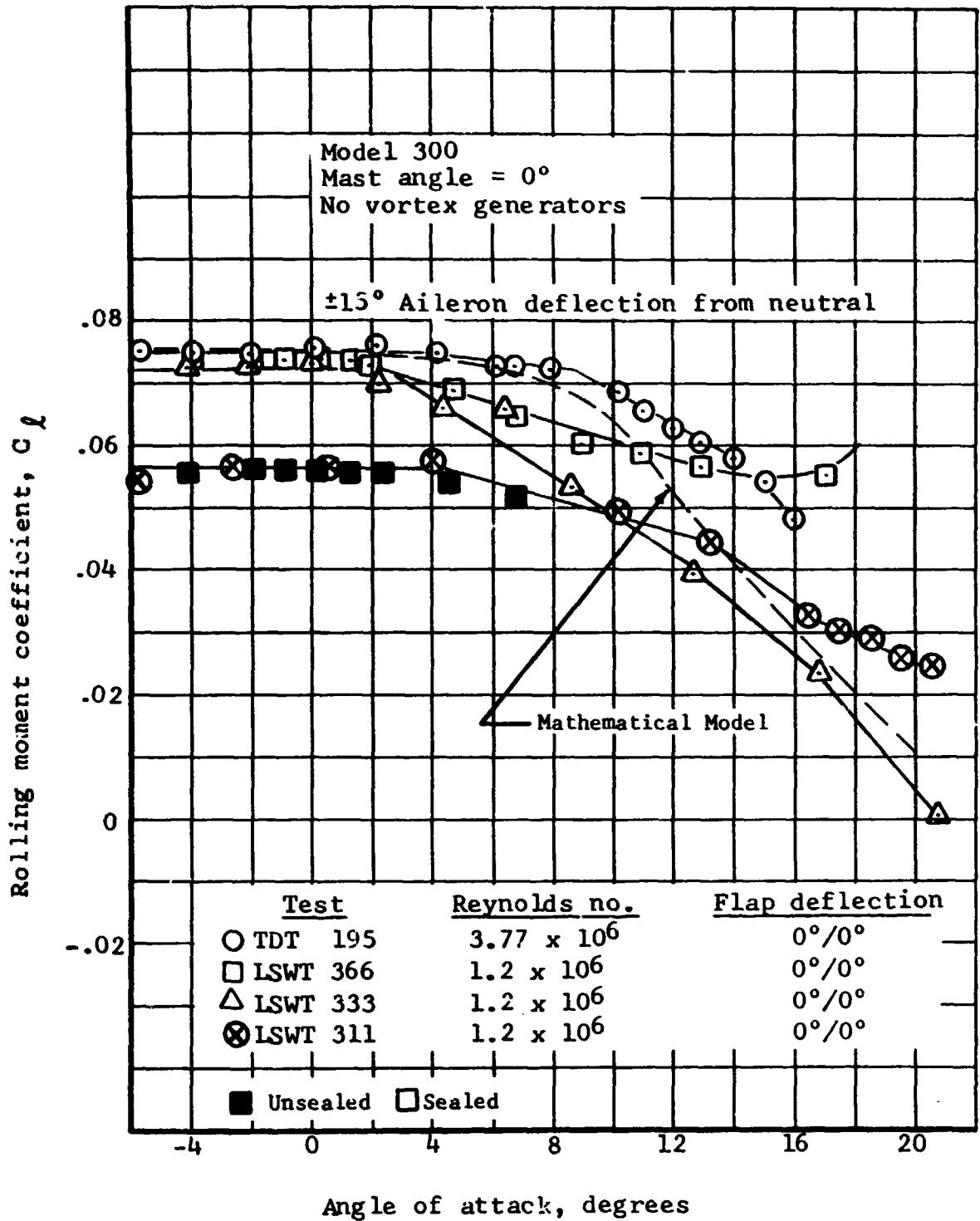


Figure III-19. Aileron Control Power Correlation



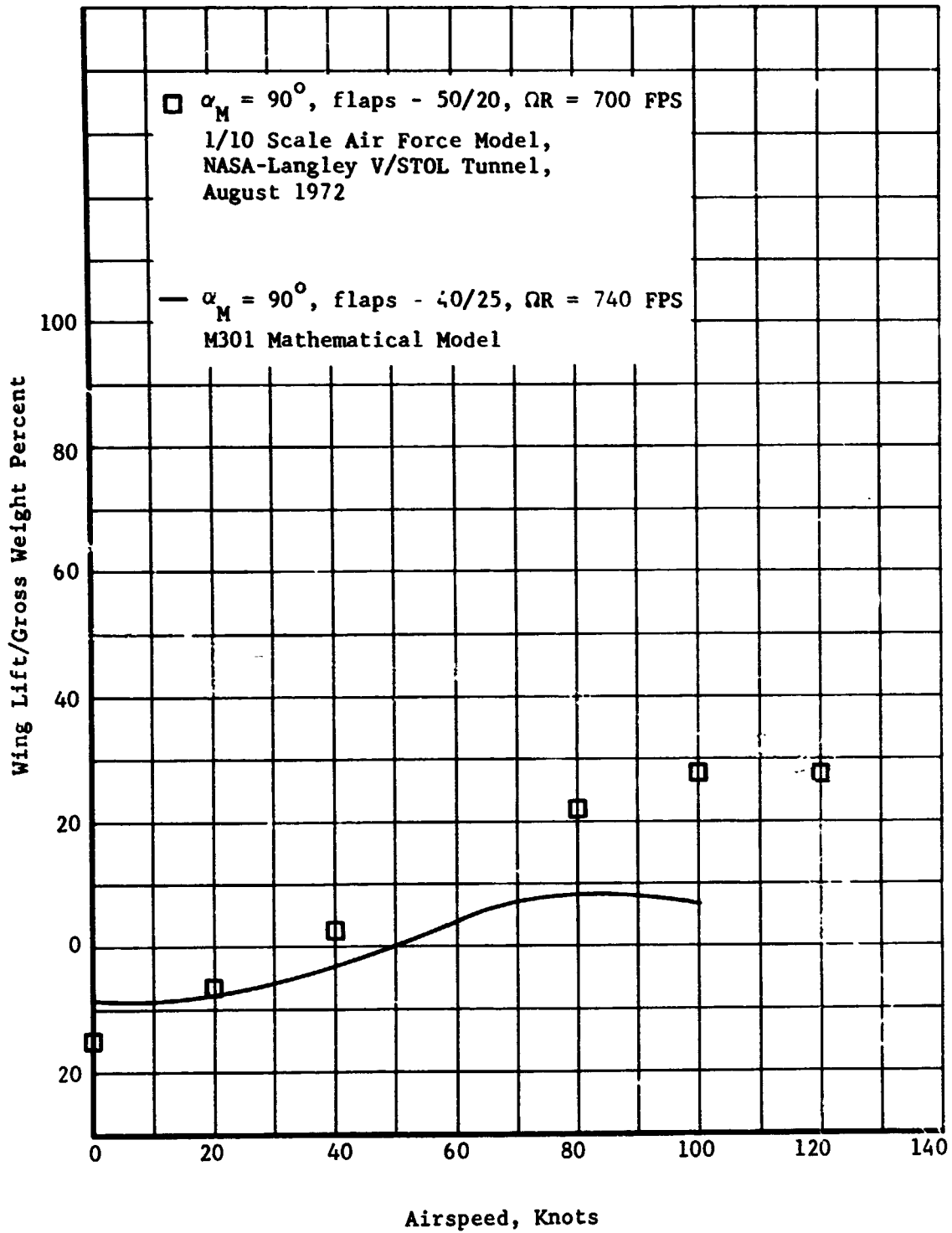


Figure III-20. Rotor Wake Effect on Wing Lift

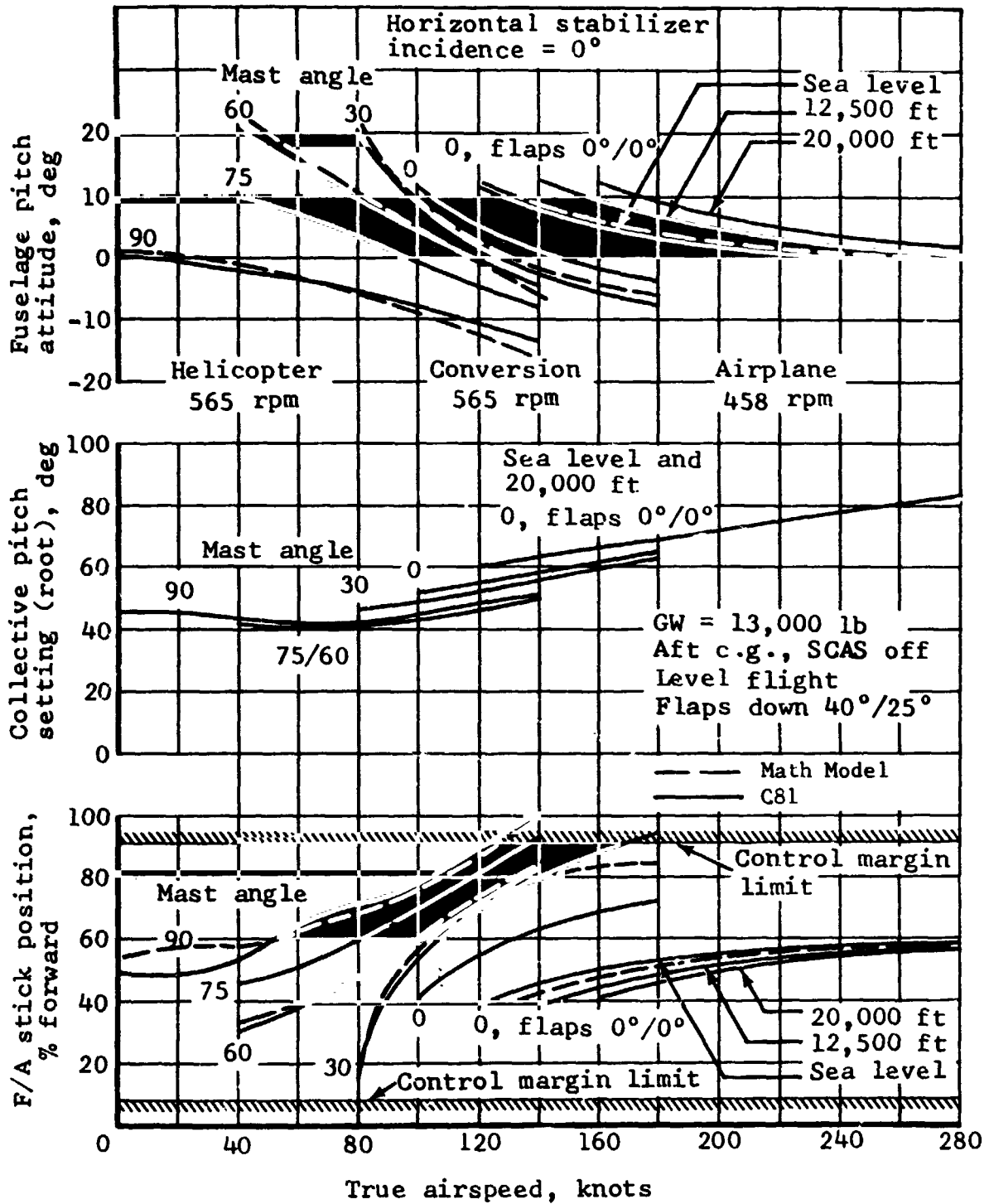


Figure III-21. Level Flight Trim Condition Correlation

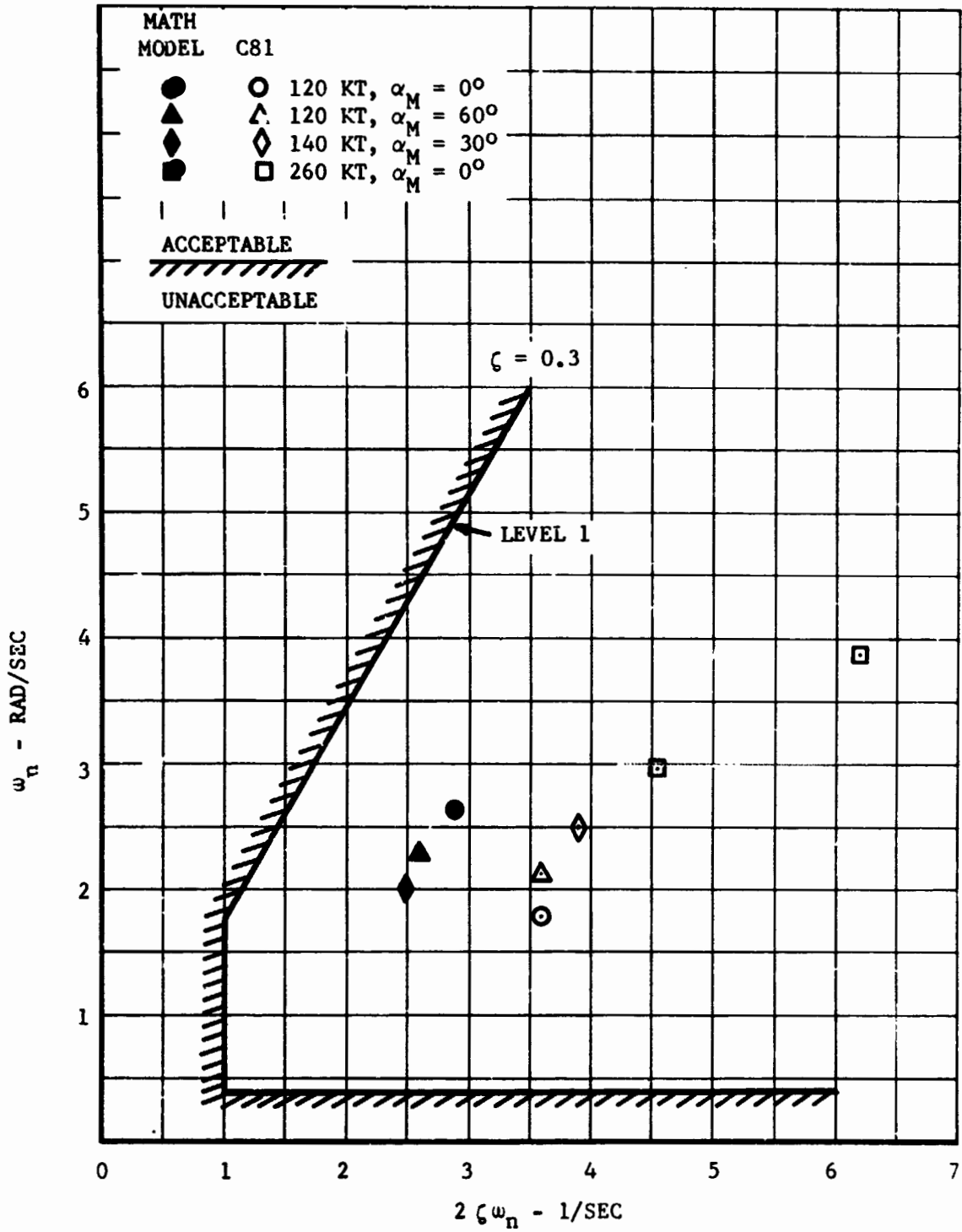


Figure III-22. Comparison of Longitudinal Oscillatory Mode Characteristics With MIL-F-83300 Response Requirements

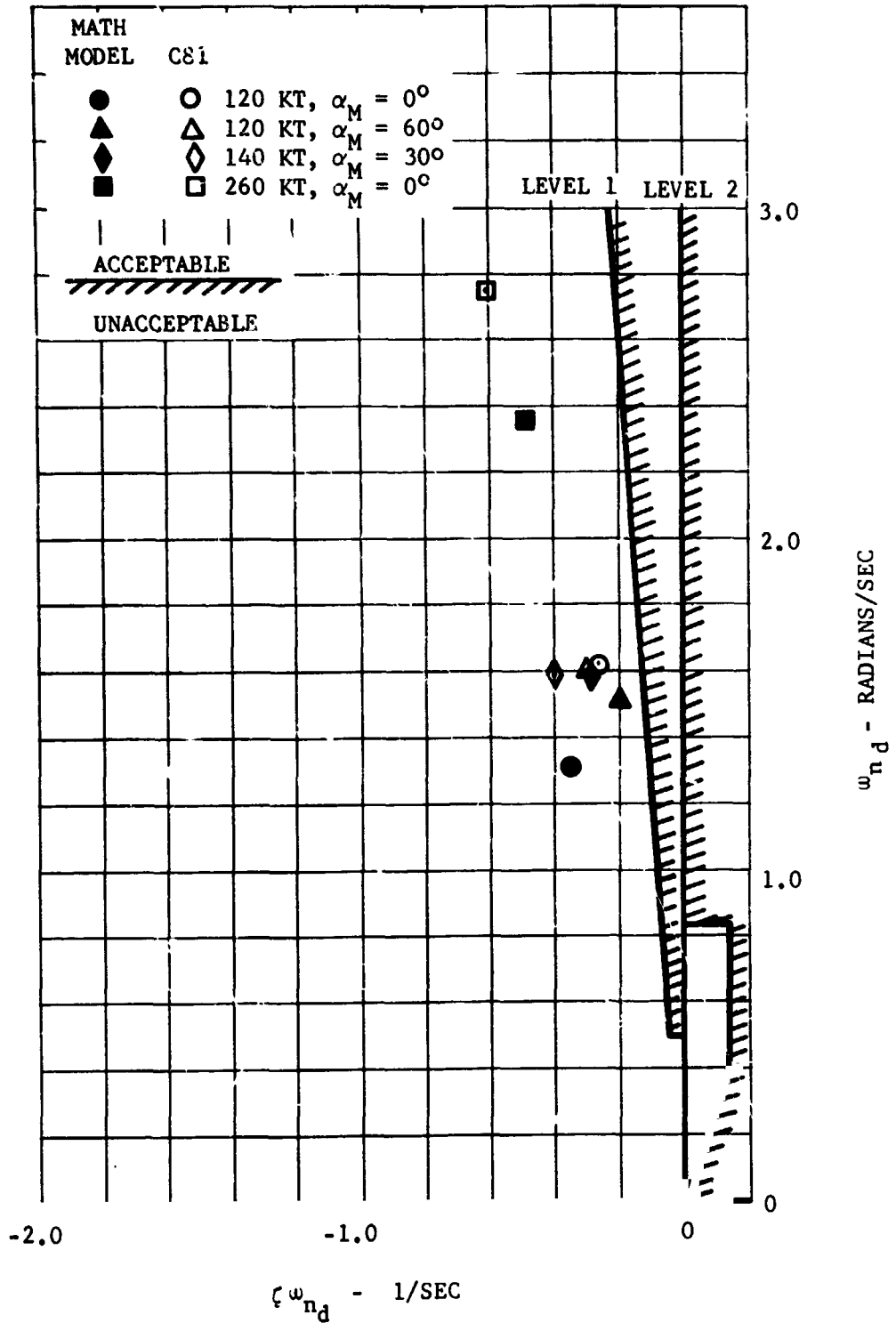


Figure III-23. Comparison of Lateral-Directional Oscillatory Mode Characteristics With MIL-F-83300 Response Requirements

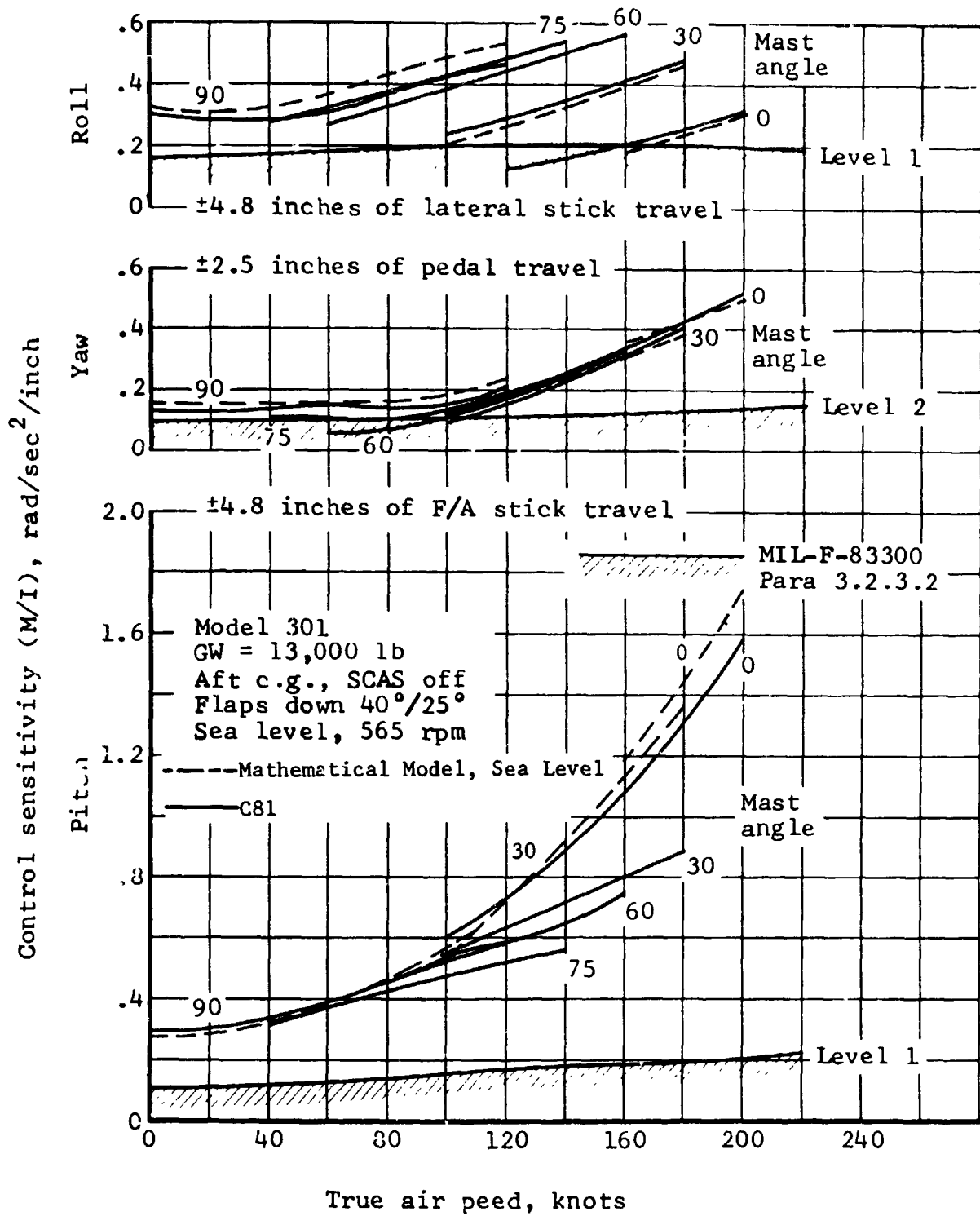


Figure III-24. Helicopter Conversion Mode Control Sensitivity Correlation

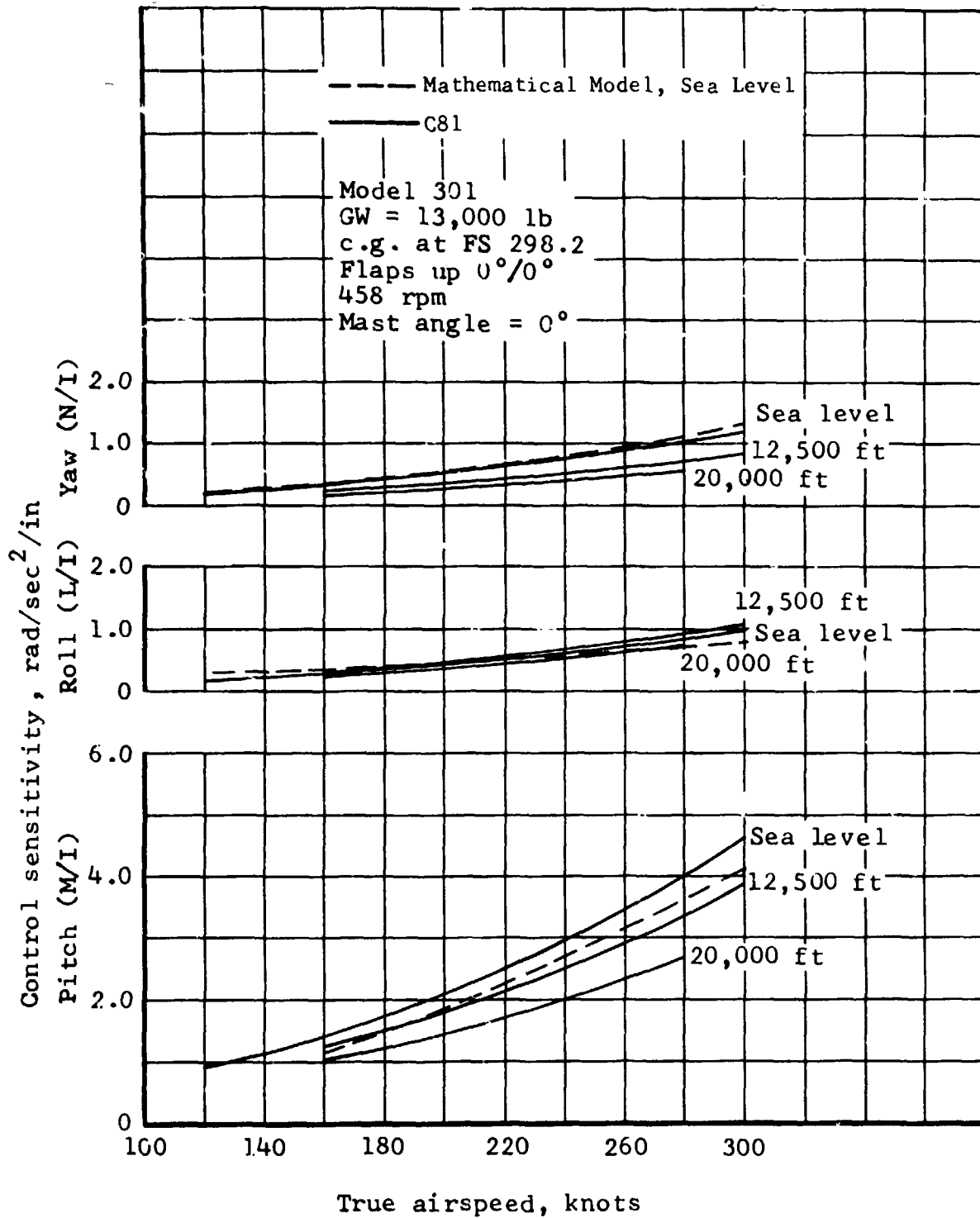


Figure III-25. Airplane Mode Control Sensitivity Correlation

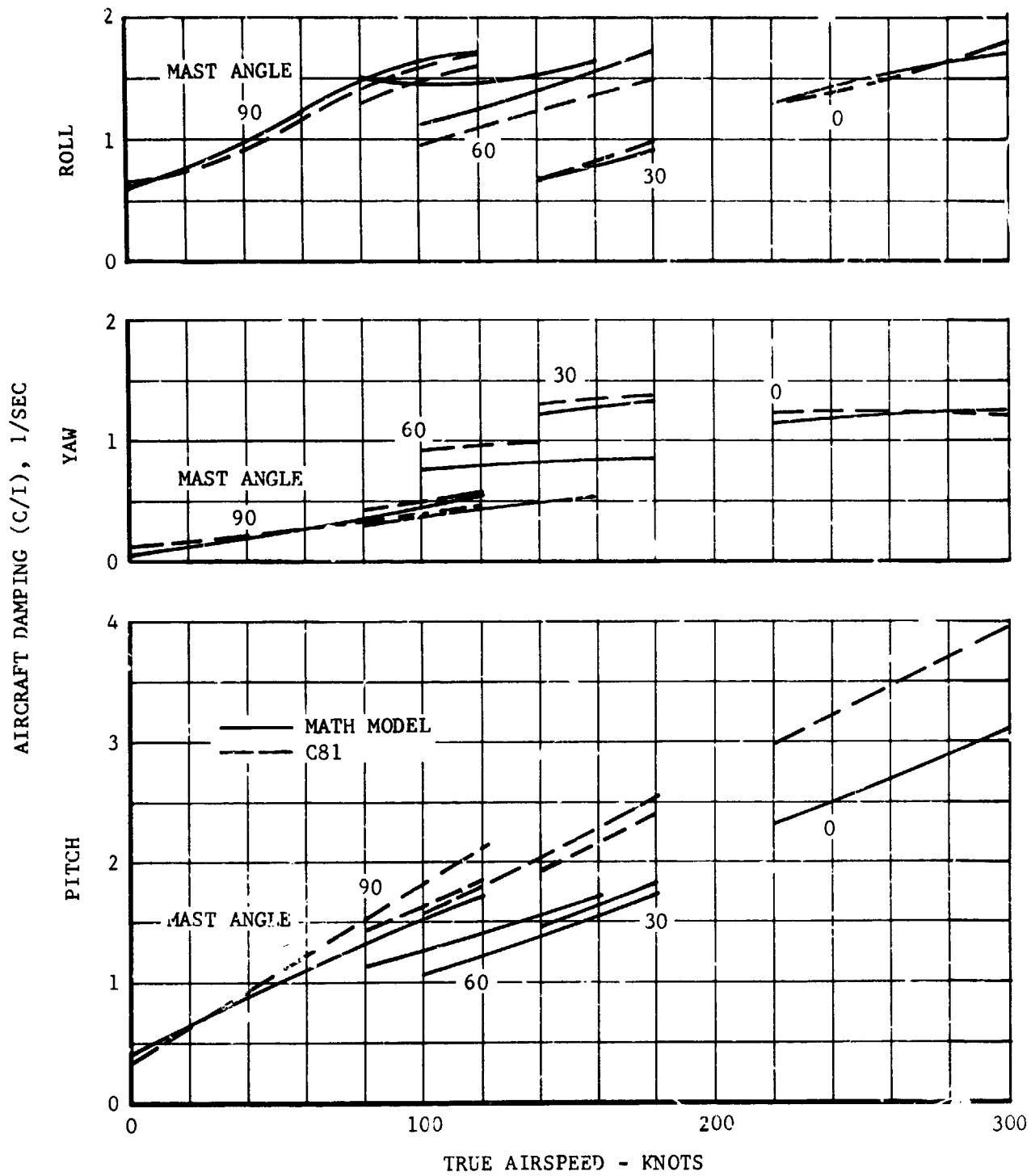


Figure III-26. Aircraft Damping Correlation

#### IV. CONTRACTOR EVALUATION OF FSAA SIMULATION

Contractor evaluation of the Model 301 research aircraft simulation on the NASA Ames FSAA to define deficiencies of the math model, computer program, and the simulator hardware was conducted prior to formal evaluation by the Army/NASA Tilt Rotor Project Office.

##### A. Math Model and Computer Program Deficiencies

- (1) The rotor math model does not include stall or compressibility. The requirement to include these effects in the math model was deleted in order to meet the schedule for the Army/NASA evaluation. However, the first order effects of stall and compressibility have been included by limiting thrust to a value determined using an analysis which includes stall and compressibility and by arbitrarily modifying the rotor power required equation to obtain the correct power at maximum speed in the helicopter and airplane modes.

The rotor flapping time constant is approximated by the transport delay lagging the response to cyclic stick inputs by the computer frame time (0.048 seconds). This gives a minimum delay of 0.048 seconds and a maximum of 0.096 seconds compared to the rotor time constant in hover of 0.08 seconds.

A uniform inflow is assumed which means that lateral flapping in helicopter mode is inaccurate, and the tandem rotor effect on rotor induced velocity in sideward flight is not represented.

- (2) Airframe aerodynamics are not adequate in rearward and sideward flight as accurate aerodynamics are limited to  $\alpha, \beta = \pm 20$  degrees. While wing and tail stall are represented, the wing downwash at the tail is believed subject to error at angles of attack above wing stall.

A change in wing flap setting produces an abrupt change in lift, drag, and pitching moment. The flap setting changes instantaneously and is therefore not representative.

Wing dihedral effects are accurate only for sideslip angles less than  $\pm 12$  degrees.

- (3) Rotor downwash at the horizontal tail is accurate only for level flight as it is not varied as a function of angle of attack or sideslip.

Wake recirculation effects at the horizontal tail in ground effect are not accounted for.

- (4) The SCAS and RPM governor systems have not been optimized. Further work on these systems is planned for the Phase II simulations.



However, their performance is satisfactory for preliminary evaluation of aircraft characteristics.

- (5) With the aircraft on the ground and at full down collective pitch, a high frequency oscillation (approximately 5 cps) occurs. This apparently results from an unstable interaction between the landing gear math model and the rotor induced velocity math model. Raising the collective approximately one inch eliminates the oscillation.
- (6) Several unexplainable computer "crashes" occurred when flying at  $\alpha_{\text{mast}} = 60$  degrees. These were not repetitive, in that flying back to the same condition did not result in a "crash". The cause of this problem has not yet been determined.

#### B. Simulator Hardware Deficiencies

- (1) The cab controls are not representative of the Model 301. The cyclic stick has too much inertia which affects the pilots ability to hover (particularly SCAS off). In hover, numerous rapid and small amplitude motions are required. The high inertia causes a lag and an overshoot, resulting in over controlling. The cyclic also has an excess amount of free play in the force detent. The cyclic, with force feel on, is too "viscous" feeling. Smooth inputs and exact return to trim are difficult. Additionally, there is a ratcheting feel when moving the cyclic fore and aft.

With the force gradient off (magnetic brake button depressed) unacceptable high inertia and viscous damping forces are present. When the brake is released there is a large stick jump.

The pedals also have a high inertia causing lag and overshoot.

- (2) The visual system has no peripheral capability making hovering flight very difficult. The resolution of the display is low causing difficulty in perception of small velocity and attitude changes further increasing the difficulty in hover.
- (3) The instrument panel is a compromise between the two contractors, consequently the panel layout is not that of the Model 301.
- (4) The absence of noise and vibration cues characteristic of rotary wing VTOL's reduces the flight condition information provided the pilot.

## V. RECOMMENDATIONS FOR FURTHER DEVELOPMENT OF MODEL 301 SIMULATION

The following modifications and additions to the mathematical model, computer program, and FSAA hardware are recommended for implementation prior to the Phase II simulations:

### A. Mathematical Model and Computer Program

#### 1. Rotor

The rotor tip-path-plane flapping degrees of freedom should be included in the mathematical model to improve prediction of transient flapping and rotor inplane shear forces. Differential equations for rotor flapping have already been derived but were not implemented because they required a large increase in computer frame time. The problem with implementing the differential equations lies with the forward precession flapping mode which has a natural frequency of approximately two-per-rev. In order to include the flapping degrees of freedom in the mathematical model it will be necessary to find a means of eliminating the forward precession mode from the solution of the differential equations.

The rotor induced velocity must be modified to include non-uniform inflow and the tandem rotor effect in sideward flight.

#### 2. Airframe

The mathematical model of the airframe aerodynamics should be extended to include  $\pm 180$  degrees of angle of attack and sideslip so that the aerodynamics are completely valid in vertical, rearward, and sideward flight. Data from the tests of the Model 301 powered aeroelastic model can be used to guide this effort.

The wing downwash representation at angles of attack above wing stall should be improved.

The computer program should be modified to interpolate between flap settings and the flap extend/retract time should be added to the mathematical model.

A representation of the airframe buffet which accompanies lifting surface stall should be added to the model to provide the pilot with a stall cue

#### 3. Rotor Wake-Airframe-Ground Interaction

The mathematical model of the rotor wake effects on the horizontal stabilizer and fins should be modified to reflect data from the test of the Model 301 powered aeroelastic model.

Ground effects on the wing and horizontal tail should be added to improve representation of STOL mode take off and landing.

4. Flight Controls and Subsystems

The mathematical model should be modified to include failures in the subsystems. This will require modeling all channels of the SCAS and RPM governor, the potential failure modes, and failure indicators.

5. Engine and Drive System

A mathematical model of the second engine and the refinements required to improve simulation of engine failure should be incorporated.

B. Simulator Hardware

1. Cab Cockpit

The FSA cockpit should be mock-up to represent the Model 301 instrument panel layout, pedal and cyclic stick inertia and free play characteristics (new hardware will probably be necessary), and the power management system. In order to use the simulation to optimize the cockpit layout it will be necessary to represent exactly the location, size and shape of the various levers manipulated by the pilot. It will also be necessary to represent the instrument location and face characteristics. Location of all switches, knobs and indicator lights should be represented.

2. Noise Cues

A noise cue generator having noise characteristics appropriate to the Model 301 should be developed. The generator should include rotor, engine and drive system noise which varies as a function of rpm, power setting, conversion angle and airspeed.

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APPENDIX A

EQUATIONS FOR TILT-ROTOR  
FLIGHT SIMULATION PROGRAM

**SUBSYSTEM NO. 1: ROTOR AERODYNAMICS**

INPUTS

A. Variables

Subsystem Block No.	Symbol	Units
12	U	Ft/Sec
↓	V	Ft/Sec
11	W	Ft/Sec
↓	P	Rad/Sec
15	q	Rad/Sec
↓	r	Rad/Sec
8a	$\rho$	Slug/Ft <sup>3</sup>
↓	$M_N$	l
2	$\beta_N$	Rad
19	$W_{iR}$	Ft/Sec
8a	$\Omega_R$	Rad/Sec
↓	$B_{1R}$	Rad
↓	$\theta_{OR}$	Rad
2	$A_{1R}$	
↓	$W_{iL}$	Ft/Sec
19	$\Omega_L$	Rad/Sec
8a	$B_{1L}$	Rad
↓	$\theta_{OL}$	Rad
↓	$A_{1L}$	
9	$h_H$	Ft

B. Coefficients

	Units
$a_0, a_1, a_2$	1/rad, 1/ $\mu$ , 1/ $\mu^2$
$\delta_0, \delta_1, \delta_2$	ND, 1/Deg, 1/Deg <sup>2</sup>
B	ND
$C_T$	ND
$X_{SS}$	ND



**EQUATIONS**

**A. Blade Twist Constants**  
**(One Time Per Rotor)**

$$K_{SO,m} = \frac{1}{\theta_1^m} \left[ \cos (\theta_1^m X_m) - \cos (\theta_1^m X_{m-1}) \right]$$

$$K_{CO,m} = \frac{1}{\theta_1^m} \left[ \sin (\theta_1^m X_{m-1}) - \sin (\theta_1^m X_m) \right]$$

$$K_{S1,m} = \frac{1}{\theta_1^m} \left[ K_{CO,m} - \left( X_{m-1} \cos (\theta_1^m X_{m-1}) - X_m \cos (\theta_1^m X_m) \right) \right]$$

$$K_{C1,m} = \frac{-1}{\theta_1^m} \left[ K_{SO,m} - \left( X_{m-1} \sin (\theta_1^m X_{m-1}) - X_m \sin (\theta_1^m X_m) \right) \right]$$

$$K_{S2,m} = \frac{2}{\theta_1^m} K_{C1,m} - \frac{1}{\theta_1^m} \left[ (X_m - 1)^2 \cos (\theta_1^m X_{m-1}) - (X_m)^2 \cos (\theta_1^m X_m) \right]$$

$$K_{C2,m} = \frac{-2}{\theta_1^m} K_{S1,m} + \frac{1}{\theta_1^m} \left[ (X_m - 1)^2 \sin (\theta_1^m X_{m-1}) - (X_m)^2 \sin (\theta_1^m X_m) \right]$$

$$K_{S3,m} = \frac{3}{\theta_1^m} K_{C2,m} - \frac{1}{\theta_1^m} \left[ (X_{m-1})^3 \cos (\theta_1^m X_{m-1}) - (X_m)^3 \cos (\theta_1^m X_m) \right]$$

$$K_{C3,m} = \frac{-3}{\theta_1^m} K_{S2,m} + \frac{1}{\theta_1^m} \left[ (X_{m-1})^3 \sin (\theta_1^m X_{m-1}) - (X_m)^3 \sin (\theta_1^m X_m) \right]$$

Where  $\theta_1^m = \text{twist rate of } m^{\text{th}} \text{ segment} = \frac{(\theta_m - \theta_{m-1})}{(X_m - X_{m-1})}$

$X_m = \text{Radial station of } m^{\text{th}} \text{ segment}$

$\theta_m = \text{Blade pitch angle at } m^{\text{th}} \text{ segment}$

$K_{Cn,m}$  = Blade twist constants

( $n = 0, 1, 2, 3$ )

$m$  = No. of segments of starting from tip ( $R = B$ )  
to root ( $R = 0$ )

Define,

$$TW1_n = \sum_1^m K_{Cn,m} \cos \Delta \theta_{Om}$$

$$TW2_n = \sum_1^m K_{Cn,m} \sin \Delta \theta_{Om}$$

$$TW3_n = \sum_1^m K_{Sn,m} \sin \Delta \theta_{Om}$$

$$TW4_n = \sum_1^m K_{Sn,m} \cos \Delta \theta_{Om}$$

Where,  $\Delta \theta_{Om} = (\theta_m - \theta_R) - X_m \theta_1^m$ ,  $\theta_R$  Blade pitch at the rotor center.

**B. Initial Transformation Equations**  
**(One Time Per Rotor)**

$$A = \pi R^2$$

$$DN' = AR^2$$

$$TD3 = TAN (\delta_3)$$

$$\sigma = \frac{n_b c_b}{\pi R}$$

$$\gamma' = \frac{cR^4}{I_b}$$

$$\gamma_m = p\gamma'$$

C. Long Term Transformations

1. Rotor Angular Velocity in Space.

$$\Omega'_R = \Omega_R + p \sin \beta_M \cos \phi_M + q \cos \beta_M \sin \phi_M - r \cos \beta_M \cos \phi_M$$

$$\Omega'_L = \Omega_L + p \sin \beta_M \cos \phi_M + q \cos \beta_M \sin \phi_M + r \cos \beta_M \cos \phi_M$$

2. 'Wind-Mast' Axis Angular Rates

Right Rotor

$$p_{WMR} = p_{HMR} \cos \xi_{WMR} + q_{HMR} \sin \xi_{WMR}$$

$$q_{WMR} = -p_{HMR} \sin \xi_{WMR} + q_{HMR} \cos \xi_{WMR}$$

where

$$p_{HMR} = p \cos \beta_M - q \sin \beta_M \sin \phi_M + r \sin \beta_M \cos \phi_M$$

$$q_{HMR} = q \cos \phi_M + r \sin \phi_M$$

$$\frac{p_{WMR}}{p} = \frac{p_{HMR}}{\Omega'_R}$$

$$\frac{q_{WMR}}{q} = \frac{q_{HMR}}{\Omega'_R}$$

$$\xi_{WMR} - \text{Wind azimuth angle defined to be equal to } \tan^{-1} \frac{V_{HMR}}{U_{HMR}}$$

Left Rotor

$$P_{WML} = P_{HML} \cos \epsilon_{WML} + q_{HML} \sin \epsilon_{WML}$$

$$q_{WML} = -P_{HML} \sin \epsilon_{WML} + q_{HML} \cos \epsilon_{WML}$$

where

$$P_{HML} = -p \cos \beta_M - q \sin \beta_M \sin \phi_M - r \sin \beta_M \cos \phi_M$$

$$q_{HML} = q \cos \phi_M - r \sin \phi_M$$

$$\Lambda_{P_{WML}} = \frac{P_{WML}}{\Omega'_L} \quad \Lambda_{q_{WML}} = \frac{q_{WML}}{\Omega'_L}$$

$$\epsilon_{WML} - \text{Wind azimuth angle defined to be equal to } \tan^{-1} \frac{V_{HML}}{U_{HML}}$$

3. Rotor Hub Velocity - Mast Axes

Right Rotor

$$U_{HMR} = U_{HBR} \cos \beta_M - V_{HBR} \sin \beta_M \sin \phi_M + W_{HBR} \sin \beta_M \cos \phi_M$$

$$V_{HMR} = V_{HBR} \cos \phi_M + W_{HBR} \sin \phi_M$$

$$W_{HMR} = -U_{HBR} \sin \beta_M - V_{HBR} \cos \beta_M \sin \phi_M + W_{HBR} \cos \beta_M \cos \phi_M$$

where,

$$\begin{array}{l} U_{HBR} = U - q * L_{ZH} - r * L_{YH} \\ V_{HBR} = V + p * L_{ZH} + r * L_{XH} \\ W_{HBR} = W + p * L_{YH} - q * L_{XH} \end{array} \left| \begin{array}{l} L_{XH} = \frac{(SL_{CG} - SL_{SP})}{12} + l_M \sin \beta_M \cos \phi_M \\ L_{YH} = \frac{(BL_{SP} - B_{CG})}{12} + l_M \sin \phi_M \\ L_{ZH} = \frac{(WL_{SP} - WL_{CG})}{12} + l_M \cos \beta_M \cos \phi_M \end{array} \right.$$

Left Rotor

$$U_{HML} = U_{HBL} \cos \beta_M + V_{HBL} \sin \beta_M \sin \phi_M + W_{HBL} \sin \beta_M \cos \phi_M$$

$$V_{HML} = -V_{HBL} \cos \phi_M + W_{HBL} \sin \phi_M$$

$$W_{HML} = -U_{HBL} \sin \theta_M + V_{HBL} \cos \theta_M \sin \phi_M + W_{HBL} \cos \theta_M \cos \phi_M$$

where,

$$U_{HBL} = U - q * L_{ZH} + r * L_{YH}$$

$$V_{HBL} = V + p * L_{ZH} + r * L_{XH}$$

$$W_{HBL} = W - p * L_{YH} - q * L_{XH}$$

#### 4. Aerodynamic Coefficients

##### Right Rotor

$$DN_R = \rho \Omega_R'^2 DN'$$

$$DNQ_R = DN_R (\Omega_R' R / 550)$$

$$\mu_R = \frac{[U_{HMR}^2 + V_{HMR}^2]^{1/2}}{\Omega_R' R}$$

$$\lambda_{OR} = - \frac{W_{HMR}}{\Omega_R' R}$$

$$\xi_{WMR} = \tan^{-1} \frac{V_{HMR}}{U_{HMR}}$$

$$a_R = a_0 + \mu_R (a_1 - a_2 \mu_R) \quad (a_0, a_1, a_2 - \text{blade lift coefficients})$$

$$C_{KFAR} = \frac{2/3 K_{FA}}{I_b \Omega_R'^2} \quad ; \quad C_{KLTR} = \frac{2/3 K_{LT}}{I_b \Omega_R'^2}$$

$$\gamma_R = \frac{\rho a_R c R^4}{I_b} \left[ 1 + \frac{\mu_R}{2} \right] = \gamma_{MR} \left[ 1 + \frac{\mu_R}{2} \right]$$

$$\gamma_{MR} = \gamma_M a_R$$

Define,

$$Q_R = 0.5 \sigma a_R (DN_R)$$

(For Left rotor replace subscript R with L)

D. Short Term Transformations  
(Every update Cycle)

1. 'Wind-Matrix' Axis Cyclic Inputs  
Right Rotor

$$\bar{A}_{1R} = A_{1R} \cos E_{WMR} - B_{1R} \sin E_{WMR}$$

$$\bar{B}_{1R} = A_{1R} \sin E_{WMR} + B_{1R} \cos E_{WMR}$$

(For left rotor replace subscript R with L)

2. Blade Pitch Constants  
Right Rotor

$$C_{SnR} = (TW1_n - TW3_n) \sin \theta_{OR} + (TW2_n + TW4_n) \cos \theta_{OR}$$

$$C_{CnR} = -(TW2_n + TW4_n) \sin \theta_{OR} + (TW1_n - TW3_n) \cos \theta_{OR}$$

(For left rotor replace subscript R with L)

3. Performance Parameters  
Right Rotor

$$\alpha_{rR} = \frac{7C_{TR}}{\sigma a_R}$$

$$C_{d_{fR}} = [\delta_0 + \alpha_{rR} (\delta_1 + \alpha_{rR} \delta_2)] / 2a_R$$

(For left rotor replace subscript R with L)

4. Thrust and Induced Velocity  
Right Rotor

$$G = (0.76 + .24 * \frac{h_H}{2R})$$

$$\text{IF } \sqrt{U^2 + V^2} > 30 \text{ OR } G > 1 \text{ SET } G = 1$$

$$\textcircled{D1} \quad Q_{3R} = \frac{T_R}{2B^2 DN_R}$$

$$Q_{5R} = 0.6(|Q_{3R}|^{1.5}) / (|Q_{3R}| + 8\lambda_R^2)$$

$$\lambda_R = \lambda_{OR} + \lambda_{iR}$$

$$\textcircled{D2} \quad Q_{iR} = \lambda_{OR} + \frac{Q_{3R} \left[ 1 + (G - 1) \left( \frac{\sqrt{U^2 + V^2}}{30} - 1 \right)^2 \right]}{(\mu_R^2 + .866 \lambda_R^2)^{.5} + \frac{Q_{5R} (|Q_{3R}| - \frac{8}{3} \lambda_R |\lambda_R|)}{(|Q_{3R}| + 8 \lambda_R^2)}}$$

$$+ \frac{x_{3S} f(\mu_R) Q_{3R}}{\mu_R}$$

$$\lambda_{iR} = Q_{1R} - \lambda_{OR}$$

If  $(|Q_{1R} - \lambda_R| > .001 |\lambda_{iR}|)$  set  $\lambda_R = Q_{1R}$  Go to (D2)  
(In IC only)

$$\lambda_R = 0.7 Q_{1R} + 0.3 \lambda_R$$

$$T_{2R} = Q_{6R} \left\{ C_{S2R} + C_{SOR} \frac{\mu_R^2}{2} - \lambda_R (C_{C1R} + C_{dFR}) \right. \\ \left. - \frac{\bar{B}_1}{2} \left[ \mu_R (C_{SOR} \lambda_R + 2 C_{C1R}) \right] \right\}$$

If  $(|T_{2R} - T_R| > .001 |T_R|)$  set  $T_R = T_{2R}$  Go to (D1)

$$T_R = T_{2R}$$

$$C_{T_R} = \frac{T_R}{DN_R}$$

$$W_{iR} = \lambda_{iR} (\Omega'_R R)$$

(For  $\lambda_{iR}$  initialization use  $\lambda_{iR} = \text{Sign } Q_{3R} * (Q_{3R})^{.5}$ )

Thrust Limit

If  $T_R > \bar{T}(\mu)$ , Set  $T_R = \bar{T}_R$

$$\bar{T}_R = C_{\bar{T}} f(\mu) * DN_R$$

(For Left Rotor replace subscript R with L)



5. Rotor Flapping (Wind-Mast Axis System)  
Right Rotor

$$\begin{bmatrix} AR(1,1) & AR(1,2) & AR(1,3) \\ AR(2,1) & AR(2,2) & AR(2,3) \\ AR(3,1) & AR(3,2) & AR(3,3) \end{bmatrix} \begin{bmatrix} a_{OR} \\ a_{1R} \\ b_{1R} \end{bmatrix} = \begin{bmatrix} BR(1) \\ BR(2) \\ BR(3) \end{bmatrix}$$

$$AR(1,1) = \frac{2}{\gamma_{MR}} + \frac{\mu_R}{2} \bar{A}_{1R} C_{S2R}$$

$$AR(1,2) = \frac{1}{2} \bar{B}_{1R} (C_{S3R} - \frac{\mu_R^2}{4} C_{S1R})$$

$$AR(1,3) = -\frac{1}{2} \bar{A}_{1R} (C_{S3R} + \frac{\mu_R^2}{4} C_{S1R}) - \mu_R \tan \delta_3 (C_{C2R} + \frac{\lambda_R}{2} C_{S1R})$$

$$AR(2,1) = \frac{\mu_R^2}{4} \bar{A}_{1R} C_{S1R}$$

$$AR(2,2) = C_{C3R} - \frac{1}{4} \mu_R^2 (C_{C1R} + C_{d_{fR}}) + \frac{1}{2} \left[ C_{d_{fR}} + \mu_R \bar{B}_{1R} C_{S2R} \right]$$

$$AR(2,3) = \frac{\mu_R}{2} \bar{A}_{1R} C_{S2R} - C_{KLTR} \\ - \tan \delta_3 (C_{C3R} + \frac{3}{4} \mu_R^2 C_{C1R} + \lambda_R C_{S2R})$$

$$AR(3,1) = -\mu_R \left[ C_{C2R} + \frac{2}{3} C_{d_{fR}} + \frac{\mu_R}{4} \bar{B}_{1R} C_{S1R} \right]$$

$$AR(3,2) = \frac{\mu_R}{2} \bar{A}_{1R} C_{S2R} + C_{KFAR} \\ + \tan \delta_3 \left( \frac{3}{4} \mu_R^2 C_{C1R} + \lambda_R C_{S2R} + C_{C3R} \right)$$

$$AR(3,3) = C_{C3R} + \frac{1}{2} \left\{ C_{d_{FR}} + \mu_R \left[ \bar{b}_{1R} C_{S2R} + \frac{\lambda}{2} \mu_R (C_{C1R} + C_{d_{FR}}) \right] \right\}$$

$$BR(1) = C_{S3R} + \frac{\mu_R^2}{2} C_{S1R} - \lambda_R C_{C2R} - \mu_R \bar{E}_{1R} (C_{C2R} + \frac{\lambda_R}{2} C_{S1R})$$

$$BR(2) = \mu_R \left[ 2 C_{S2R} - \lambda_R (C_{C1R} + C_{d_{FR}}) \right] - \bar{b}_{1R} (C_{C3R} + \frac{3}{4} \mu_R^2 C_{C1R} + \lambda_R C_{S2R}) - 4 \frac{\Lambda}{\gamma_{MR}}$$

$$BR(3) = \bar{a}_{1R} \left( \frac{\mu_R^2}{4} C_{C1R} + \lambda_R C_{S2R} + C_{C3R} \right) - \frac{4\Lambda}{\gamma_{MR}}$$

6. Rotor flapping in mast axis system

$$a_{1R} = \bar{a}_{1R} \cos \epsilon_{WMR} + \bar{b}_{1R} \sin \epsilon_{WMR}$$

$$b_{1R} = -\bar{a}_{1R} \sin \epsilon_{WMR} + \bar{b}_{1R} \cos \epsilon_{WMR}$$

(For left rotor replace R with L)

7. Rotor Inplane Forces in Wind-Mast Axis System  
(Right Rotor)

$$\bar{H}_R = Q_{6R} \left\{ \frac{1}{2} C_{S0R} \mu_R \lambda_R + C_{S2R} \left( \bar{a}_{1R} - \frac{\Lambda}{2} \right) - \frac{1}{3} C_{C1R} \lambda_R \bar{b}_{1R} \right.$$

$$\left. + C_{C1R} \left[ \frac{\mu_R}{2} (a_{OR}^2 + a_{1R}^2) - \frac{3}{2} \lambda_R \bar{a}_{1R} + \frac{\Lambda}{P} \lambda \right] \right\}$$

(Equation  
Continued on  
Next Page)

$$\begin{aligned}
 & - \frac{\bar{B}_{1R}}{2} \left[ C_{SOR} \lambda_R^2 + C_{S2R} \bar{a}_{1R}^2 + C_{C1R} (\lambda_R + \mu_R \bar{a}_{1R}) \right] \\
 & + \frac{A_{1R}}{2} \left[ \frac{1}{4} C_{SOR} \lambda_R \mu_R \bar{b}_{1R} + C_{S1R} \lambda_R a_{OR} + C_{C2R} a_{CR} \right] + C_{d_{fR}} \mu_R \left. \right\}
 \end{aligned}$$

$$\begin{aligned}
 \bar{Y}_R = Q_{6R} \left\{ C_{S2R} \left( \bar{b}_{1R} - \frac{\Lambda}{2} \right) - \frac{3}{2} C_{S1R} \mu_R a_{OR} - \frac{3}{2} C_{C1R} \lambda_R \bar{b}_{1R} \right. \\
 \left. + \frac{1}{3} C_{C1R} \lambda_R \bar{a}_{1R} - C_{C1R} \lambda_R \frac{\Lambda}{9} \right. \\
 \left. + \bar{B}_{1R} \left[ -\frac{1}{8} C_{SOR} \lambda_R \mu_R \bar{b}_{1R} + \frac{1}{2} C_{S1R} \lambda_R a_{OR} \right. \right. \\
 \left. + \frac{1}{2} C_{C2R} a_{OR} - C_{C1R} \mu_R \bar{b}_{1R} \right] \\
 \left. + \frac{\bar{A}_{1R}}{2} \left[ C_{SOR} \lambda_R (\lambda_R - \mu_R \bar{a}_{1R}) + C_{S2R} \bar{b}_{1R}^2 \right. \right. \\
 \left. \left. + C_{C1R} (\lambda_R - \mu_R \bar{a}_{1R}) \right] \right\}
 \end{aligned}$$

8. Rotor Inplane Forces in Mast Axis System

$$H_R = \bar{H}_R \cos \xi_{WMR} + \bar{Y}_R \sin \xi_{WMR}$$

$$\bar{Y}_R = -\bar{H}_R \sin \xi_{WMR} + \bar{Y}_R \cos \xi_{WMR}$$

(For Left rotor replace subscript R with L)

9. Rotor Power and Torque Required

$$\begin{aligned}
 HP_{REQ R} = (DNQ_R) \frac{\sigma a_R}{2} & \left\{ C_{C1R} (\lambda_R \mu_R \bar{a}_{1R} - \lambda_R^2) \right. \\
 & + C_{S2R} \lambda_R - \frac{1}{2} C_{C3R} (a_{1R}^2 + b_{1R}^2) + \frac{1}{2} C_{dFR} \\
 & (1 + 280 \mu_R^3 + 60 \lambda_R^3) - \bar{B}_{1R} \left[ \frac{1}{2} C_{C3R} \bar{a}_{1R} \right. \\
 & \left. + \frac{1}{2} C_{C1R} \lambda_R \mu_R + C_{S2P} \lambda_R \bar{a}_{1R} \right] \\
 & \left. + \bar{A}_{1R} \left[ \frac{1}{2} C_{C3R} \bar{b}_{1R} - \frac{1}{2} C_{C2R} \mu_R a_{OR} + C_{S2R} \lambda_R \bar{b}_{1R} \right] \right\} \\
 Q_R = \frac{(550) HP_{REQ R}}{\Omega'_R}
 \end{aligned}$$

(For Left rotor replace subscript R with L)

10. Rotor Moments in Mast Axis System

$$M_{a_{1R}} = K_{FA} a_{1R}$$

$$l_{b_{1R}} = K_{LT} b_{1R}$$

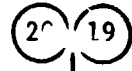
(For left rotor replace subscript R with L)

**OUTPUTS**

Subsystem Block No.

Symbol

Units

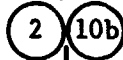


$a_{1R}$

Rad

$b_{1R}$

Rad



$T_R$

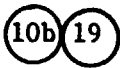
Lbs

$H_R$

Lbs

$Y_R$

Lbs



$Q_R$

Lb/Ft



$M_{a1R}$

Lb/Ft

$l_{b1R}$

Lb/Ft

$HP_{REQ R}$

HP

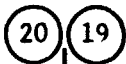


$\dot{a}_{1R}$

Rad/Sec

$\dot{b}_{1R}$

Rad/Sec

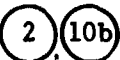


$a_{1L}$

Rad

$b_{1L}$

Rad



$T_L$

Lbs

$H_L$

Lbs

$Y_L$

Lbs

$W_{iL}$

Ft/Sec

$Q_L$

Lb/Ft



$M_{a1L}$

Lb/Ft

$l_{b1L}$

Lb/Ft



$HP_{REQ L}$

HP



$\dot{a}_{1L}$

Rad/Sec


$\dot{b}_{1L}$

Rad/Sec

**SUBSYSTEM NO. 2: ROTOR INDUCED VELOCITIES**

**INPUTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
	$T_R$	Lbs
	$H_R$	Lbs
	$Y_R$	
	$T_I$	Lbs
	$H_L$	Lbs
	$Y_L$	Lbs
	$W_{iL}$	Ft/Sec
	$\beta_M$	Rad

**B. Coefficients**

	Units
$K_0, K_1, K_2, K_3, K_4$	ND
$h_0, h_1, h_2, h_3, h_4$	ND

EQUATIONS:

Rotor Wake

$$R_{WL} = R \left\{ 0.78 + 0.22 \text{ EXP} \left[ - (0.3 + 2Z \sqrt{C_{RFL} + 60 C_{RFL}}) \right] \right\}$$

where

$$Z = \frac{(WL_{HUB} - WL_W) \beta_M}{12R} = 0$$

$$C_{RFL} = \frac{(T_L^2 + H_L^2 + Y_L^2)^{1/2}}{\rho \pi \Omega_L^2 R^4}$$

Rotor Wake at Wing, Horizontal Stabilizer and Vertical Stabilizer in Mast Axes

(For rotor wake effects on the left and right wing, on the horizontal stabilizer and on the vertical stabilizer, the values of the left rotor induced velocity will be used.)

Wing

$$W_i |_{R/WL} = (K_0 + K_1 \mu_L + K_2 \mu_L^2 + K_3 \lambda_L + K_4 \lambda_L^2) W_{iL}$$

Horizontal Stabilizer

$$W_i |_{R/H} = \left[ h_0 + h_1 \beta_M + (h_2 + h_3 \beta_M) \left( \frac{U - 168.89 + h_4 \beta_M}{168.89} \right)^2 \right] W_{iL}$$

Vertical Stabilizer

$$W_i |_{R/V} = W_i |_{R/H}$$

Components of Rotor Wake in Body Axes:

Wing:

$$U_i \Big|_{R/WL}^B = W_i \Big|_{R/WL} \sin \beta_M$$

$$W_i \Big|_{R/WL}^B = -W_i \Big|_{R/WL} \cos \beta_M$$

Horizontal Stabilizer:

$$U_i \Big|_{R/H}^B = W_i \Big|_{R/H} \sin \beta_M$$

$$W_i \Big|_{R/H}^B = -W_i \Big|_{R/H} \cos \beta_M$$

Vertical Stabilizer:

$$U_i \Big|_{R/V}^B = U_i \Big|_{R/H}^B$$

$$W_i \Big|_{R/V}^B = W_i \Big|_{R/H}^B$$



OUTPUTS

Subsystem Block No.

Symbol

Units

4

$U_i \left| \begin{matrix} B \\ R/WL \end{matrix} \right.$

Ft/Sec



$W_i \left| \begin{matrix} B \\ R/WL \end{matrix} \right.$

Ft/Sec

$R_{WL}$

Ft/Sec

5

$U_i \left| \begin{matrix} B \\ R/H \end{matrix} \right.$

Ft/Sec



$W_i \left| \begin{matrix} B \\ R/H \end{matrix} \right.$

Ft/Sec

6

$U_i \left| \begin{matrix} B \\ R/V \end{matrix} \right.$

Ft/Sec



$W_i \left| \begin{matrix} B \\ R/V \end{matrix} \right.$

Ft/Sec

SUBSYSTEM NO. 3-FUSELAGE AERODYNAMICS

INPUTS

A. Variables

Subsystem Block No.

Symbol

Units

12



$V_T$

Ft/Sec

$\alpha_F$

Deg

$\beta_F$

Deg

B. Aerodynamic Coefficients

Units

$L_0, L_1$

$Ft^2, \frac{Ft^2}{Deg}$

$D_0, D_1, D_2, D_3, D_4$

$Ft^2, \frac{Ft^2}{Deg}, \frac{Ft^2}{Deg^2}, \frac{Ft^2}{Deg}, Ft^2$

$M_0, M_1, M_2$

$Ft^3, \frac{Ft^3}{Deg}, \frac{Ft^3}{Deg}$

$Y_0, Y_1, Y_2$

$Ft^2, \frac{Ft^2}{Deg}, \frac{Ft^2}{Deg^2}$

$l_0, l_1$

$Ft^3, \frac{Ft^3}{Deg}$

$N_0, N_1$

$Ft^3, \frac{Ft^3}{Deg}$

EQUATIONS:

$$q_F = \frac{1}{2} \rho V_T^2$$

For  $\alpha_F$  and  $\beta_F \leq 20^\circ$

$$L_F = q_F (L_0 + L_1 \alpha_F)$$

$$D_F = q_F (D_0 + D_1 \alpha_F + D_2 \alpha_F^2 + D_3 |\beta_F|)$$

$$M_F' = q_F (M_0 + M_1 \alpha_F + M_2 |\beta_F|) = q_F (M_0 \alpha_F + M_1 (\alpha_F)^2 + M_2 |\beta_F|)$$

$$Y_F' = q_F (Y_0 + Y_1 \beta_F + Y_2 \beta_F |\beta_F|)$$

$$I_F' = q_F (I_0 + I_1 \beta_F)$$

$$N_F' = q_F (N_0 + N_1 \beta_F)$$

For  $\alpha_F$  and  $\beta_F > 20^\circ$

For  $\alpha_F > 20^\circ$ ,  $\beta_F < 70^\circ$

$$\left. \begin{array}{l} L_F = L_F \\ D_F = D_F \end{array} \right\} \text{ at } \alpha_F = \pm 20^\circ$$

$$M_F = M_F \text{ at } \alpha_F = \pm 40^\circ, \beta_F = \pm 20^\circ$$

$$20^\circ < \beta_F < 70^\circ$$

$$\left. \begin{aligned} Y'_F &= Y''_F \\ I'_F &= I''_F \\ N'_F &= N''_F \end{aligned} \right\} \text{At } \beta_F = \pm 20^\circ$$

For  $\beta_F > 70^\circ$ ,

$$L_F = M_F = Y'_F = I'_F = N'_F = 0 \quad D_F = q_F D_4 \quad \left. \vphantom{D_F} \right\} \text{Wind axis system}$$

OUTPUTS:

Subsystem Block No.

Symbol

Units

10a

L<sub>F</sub>

Lbs.

D<sub>F</sub>

Lbs.

Y<sub>F</sub>

Lbs.

M<sub>F</sub>

Ft-Lbs

I<sub>F</sub>

Ft-Lbs

N<sub>F</sub>

Ft-Lbs

**SUBSYSTEM NO. 4-WING-PYLON AERODYNAMICS**

**INPUTS:**

**A. Variables**

Subsystem Block No.

2

Symbol  
 $U_i \begin{matrix} | \\ B \\ R/WL \end{matrix}$

Units  
Ft/sec

↓

$W_i \begin{matrix} | \\ B \\ R/WL \end{matrix}$

Ft/sec

↓

$R_{WL}$

Ft

↓

8a

$\beta_M$

Deg

$F_X$

$F_1, F_2, F_3, F_4$

↓

$\delta_a$

Deg

↓

12

$V_T$

Ft/sec

↓

$\alpha_F$

Deg

↓

$\beta_F$

Deg

↓

U

Ft/sec

↓

V

Ft/sec

↓

W

Ft/sec

↓

$\rho$

Slug/ft<sup>3</sup>

↓

$M_N$

ND

↓

11

p

Rad/sec

↓

q

Rad/sec

↓

r

Rad/sec

**B. Aerodynamic Coefficients**

	Units
$C_{L_{WP}}, C_{L_{\delta a}}$	ND, 1/Deg
$C_{D_{WP}}$	ND
$C_{M_{O_{WP}}}$	ND
$\epsilon_{W/H}$	Deg
$X_{R/W}, X_{W/R}$	ND, ND
$C_{y\beta}  _{M_N = 0}$	1/Rad
$C_{y\beta}/C_L  _{M_N = 0}$	1/Rad
$C_{y\tau}  _{M_N = 0}$	1/Rad
$C_{l\beta}  _{C_L = 0}, C_{l\beta}/C_{L_{WP}}  _{M_N = 0}$	1/Rad, 1/Rad
$C_{lp}  _{C_L = 0}, C_{lp}  _{M_N = 0}$	1/Rad
$C_{lr}/C_{L_{WP}}  _{M_N = 0}, \Delta C_{lr}/(\frac{\partial \alpha}{\partial \delta_f}) \delta_f$	1/Rad, 1/Deg
$C_{nr}  _{M_N = 0}, C_{nr}/C_{L_{WP}}^2  _{M_N = 0}, C_{nr}/C_L  _{C_L = 0}$	1/Rad, 1/Rad
$C_{np}/C_{L_{WP}}  _{M_N = 0}$	1/Rad
$C_{nr}/C_{L_{WP}}^2, C_{nr}/C_{D_{O_{WP}}}$	1/Rad, 1/Rad
$C_{l_{\delta a}}, \delta a$	1/Deg, ND
$n_{o\delta a}, K_{n\delta a}$	1/Deg, ND
$K_{np}$	ND

7872 8-1-66

EQUATIONS

A. Wing Aerodynamics Affected by Rotor Wake

Computation of wing areas  $S_{iWR}$  and  $S_{iWL}$  under the rotor wakes.

In this subsection, K is a dummy subscript. It is replaced by R and L while computing  $S_{iWR}$  and  $S_{iWL}$  respectively.

The logical flow chart shows the sequence of computations necessary for determining  $S_{iWK}$ .

$F_1(C_{1K}, C_{2K})$  and  $F_2(C_{1K}, C_{2K})$  are procedures representing computation of a contribution to the wing-area in a rotor wake.

$F_1(C_{1K}, C_{2K})$

$$S_{iWK} = S_{iWK} + A_{1K}$$

$$XA_K = XA_K + XA_{1K}$$

$$YA_K = YA_K + XA_{1K} \sin \delta_K$$

$F_2(C_{1K}, C_{2K})$

$$S_{iWK} = S_{iWK} + \frac{1}{2} A_{1K} + \frac{R_{WYK}}{2R_{WXX}} (\sin \delta_K)(C_{2K}^2 - C_{1K}^2)$$

$$- Y_{TIPK}(C_{2K} - C_{1K})$$

$$XA_K = XA_K + \frac{1}{2} XA_{1K} + \frac{R_{WYK}}{3R_{WXX}} (\sin \delta_K)(C_{2K}^3 - C_{1K}^3)$$

$$- Y_{TIPK}(C_{2K}^3 - C_{1K}^3)$$

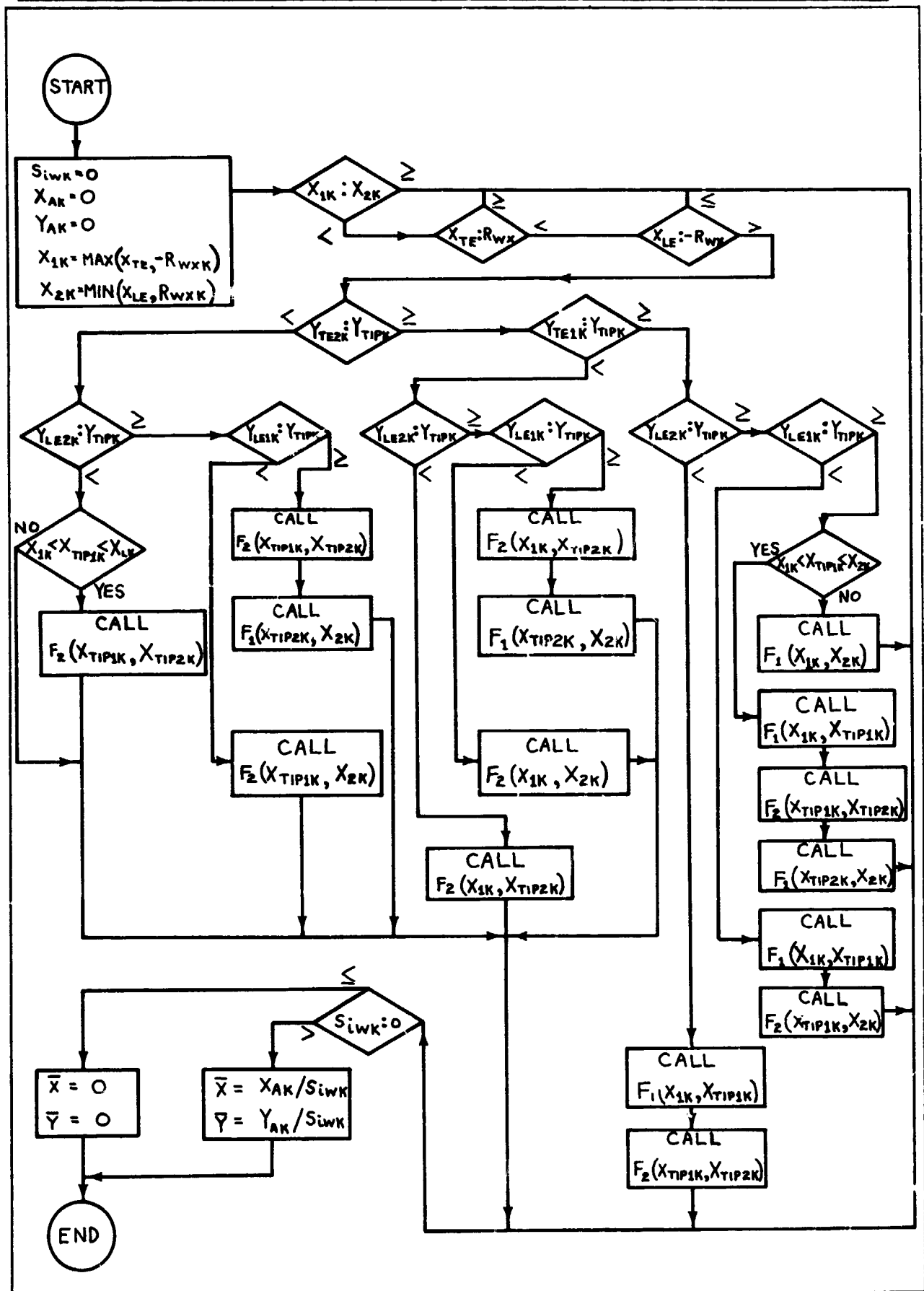
$$YA_K = YA_K + \left[ \frac{1}{2} XA_{1K} \sin \delta_K + \frac{R_{WYK}}{6R_{WXX}} (\sin^2 \delta_K - \cos^2 \delta_K) \right]$$

$$\left[ (C_{2K}^3 - C_{1K}^3) + \frac{1}{2} (R_{WXX} R_{WYK} \cos^2 \delta_K - Y_{TIPK}^2) * (C_{2K} - C_{1K}) \right] \frac{R_{WYX}}{R_{WXX}}$$

where:

$$\theta_{C_{1K}} = \sin^{-1} \frac{C_{1K}}{R_{WXX}}$$





$$\theta_{C_{2K}} = \sin^{-1} \frac{C_{2K}}{R_{WKK}}$$

$$A_{1K} = R_{WKK} R_{WYK} |\cos \delta_K| \left[ \sin \theta_{C_{2K}} \cos \theta_{C_{2K}} - \sin \theta_{C_{1K}} \cos \theta_{C_{1K}} + \theta_{C_{2K}} - \theta_{C_{1K}} \right]$$

$$XA_{1K} = -\frac{2}{3} \frac{R_{WYK}}{R_{WKK}} |\cos \delta_K| \left[ (R_{WKK}^2 - C_{2K}^2)^{3/2} - (R_{WKK}^2 - C_{1K}^2)^{3/2} \right]$$

and:

$$l' = l_M \cos \beta_M$$

$$U_{WK} = -U - W_i \left| \frac{\sin \beta_M}{R/WL} \right.$$

$$V_{WR} = V, V_{WL} = -V$$

$$W_{WK} = -W + W_i \left| \frac{\cos \beta_M}{R/WL} \right.$$

$$\text{If } \text{MAX} (|U_{WK}|, |V_{WK}|) > 100 |W_{WK}|$$

Use:

$$\frac{U_{WK}}{W_{WK}} = \frac{100 U_{WK}}{\text{MAX} (U_{WK}, V_{WK})}$$

$$\frac{V_{WK}}{W_{WK}} = \frac{100 V_{WK}}{\text{MAX} (U_{WK}, V_{WK})}$$

$\bar{\epsilon}$  is the distance forward from wing trailing edge to shaft pivot.

$$\bar{\epsilon} = \frac{(SL_{WTE} - SL_{SP})}{12}$$

$$X_{TEK} = -\bar{e} - l_M \sin \beta_M - l' \frac{U_{WK}}{W_{WK}}$$

$$X_{LEK} = c_W + X_{TEK}$$

$$Y_{T1PK} = -\frac{V_{WK}}{W_{WK}} l'$$

$$\delta_K = \tan^{-1} \left( \frac{V_{WK}}{W_{WK}} \sin \beta_M \right)$$

$$R_{WKK} = \left| R_{WK} \left( -\cos \beta_M + \frac{U_{WK}}{W_{WK}} \sin \beta_M \right) \right|$$

$$R_{WYK} = R_{WK} \sqrt{1 + \left( \frac{V_{WK}}{W_{WK}} \sin \beta_M \right)^2}$$

$$\text{If } (R^2 - X^2 \text{ or } R^2 - Y^2) \leq 0 \text{ then } \left( \sqrt{R^2 - X^2} \text{ or } \sqrt{R^2 - Y^2} \right) = 0$$

$$\text{If } X_{TE}^2 > R_{WK}^2 \text{ and } X_{TE} > 0, \text{ then } S_{iWK} = 0$$

$$Y_{TE1K} = \frac{R_{WYK}}{R_{WKK}} \left( X_{TEK} \sin \delta_K - |\cos \delta_K| \sqrt{R_{WKK}^2 - X_{TEK}^2} \right)$$

$$Y_{TE2K} = \frac{R_{WYK}}{R_{WKK}} \left( X_{TEK} \sin \delta_K + |\cos \delta_K| \sqrt{R_{WKK}^2 - X_{TEK}^2} \right)$$

$$\text{If } X_{LE}^2 > R_{WX}^2 \text{ and } X_{LE} < 0 \text{ then } S_{iWK} = 0$$

$$Y_{LE1K} = \frac{R_{WYK}}{R_{WKK}} \left( X_{LEK} \sin \delta_K - |\cos \delta_K| \sqrt{R_{WKK}^2 - X_{LEK}^2} \right)$$

$$Y_{LE2K} = \frac{R_{WYK}}{R_{W XK}} \left( X_{LEK} \sin \delta_K + |\cos \delta_K| \sqrt{R_{W XK}^2 - X_{LEK}^2} \right)$$

If  $Y_{TIP}^2 > R_{WY}^2$  and  $Y_{TIP} > 0$  then  $S_{iWK} = 0$

$$X_{T1PK} = \frac{R_{W XK}}{R_{WYK}} \left( Y_{T1PK} \sin \delta_K - |\cos \delta_K| \sqrt{R_{WYK}^2 - Y_{T1PK}^2} \right)$$

$$X_{T1P2K} = \frac{R_{W XK}}{R_{WYK}} \left( Y_{T1PK} \sin \delta_K + |\cos \delta_K| \sqrt{R_{WYK}^2 - Y_{T1PK}^2} \right)$$

$$r_{XWK} = SL_{CG} - SL_{WTE} - X_{TEK} + \bar{X}_K$$

$$r_{YWK} = BL_{SP} - BL_{CG} + Y_{T1PK} - \bar{Y}_R \text{ (for right wing)}$$

$$= -BL_{SP} + BL_{CG} - Y_{T1PK} + \bar{Y}_L \text{ (for left wing)}$$

where:  $\bar{X}_K = \frac{XA_K}{S_{iWK}}$

$$\bar{Y}_K = \frac{YA_K}{S_{iWK}}$$

If  $\beta_M > 30^\circ$ , set  $S_{iWL}, S_{iWR} = 0$

Total Velocity:

$$V_{T iWL} = \left[ (U + U_i \Big|_{R/WL}^B)^2 + v^2 + (W + w_i \Big|_{R/WL}^B)^2 \right]^{1/2}$$

Resultant angle of sideslip,

$$\beta'_F | W = \tan^{-1} \frac{v}{\left[ (U + U_i \Big|_{R/WL}^B)^2 + (W + w_i \Big|_{R/WL}^B)^2 \right]^{1/2}}$$

Resultant angle of attack

$$\alpha'_F \Big|_{WL} = \tan^{-1} \left[ \frac{W + W_i \Big|_B}{U + U_i \Big|_B} \frac{R}{WL} \right]$$

Dynamic pressure:

$$q_{iWPL} = \frac{1}{2} \rho V_{T_{iWL}}^2$$

Lift in (local) wind axis system:

$$\text{(left): } L_{iWPL} = q_{iWPL} S_{iWL} C_{L_{WP}}$$

$$\text{(right): } L_{iWPR} = q_{iWPL} S_{iWR} C_{L_{WP}}$$

Drag in (local) wind axis system:

$$\text{(left): } D_{iWPL} = q_{iWPL} S_{iWL} C_{D_{WP}}$$

$$\text{(right): } D_{iWPR} = q_{iWPL} S_{iWR} C_{D_{WP}}$$

**B. Wing Aerodynamics in freestream flow**

(Wing area subjected to freestream -  $S_{\infty W}$ )

$$S_{\infty W} = S_W - (S_{iWL} + S_{iWR})$$

Dynamic Pressure:

$$q_F = \frac{1}{2} \rho V_T^2$$

Rotor flow field effects at wing:

$$\Delta\alpha_{F/W} = 0.26 \times_{R/W} \frac{2C_{RF}}{\text{MAX}^2 (\mu_L, .15)} \quad (57.3)$$

Angle of Attack at the wing:

$$\alpha_W = \alpha_F - \Delta\alpha_{R/W}$$

Lift in (local) wind axis system:

$$L_{WP} = q_F S_{\alpha W} \left[ C_{L_{WP}} + C_{L_{\delta a}} * \delta a \right]$$

Drag:

$$D_{WP} = q_F S_{\alpha W} C_{D_{WP}} f(\alpha_W, \beta_M, F_x, M_N)$$

Note:  $C_{L_{WP}}$  here is a function of  $\alpha_W$

Pitching moment:

$$M'_{OWP} = q_F S_{\alpha W} C_W C_{M_{OWP}}$$

### C. Lateral-Directional Equations

Prandtl - Glauert compressibility factor  $C_\beta$ :

$$C_\beta = \frac{AR_W + 4 \cos (\Lambda_{c/4})_W}{AR_W B_C + 4 \cos (\Lambda_{c/4})_W}$$

where:

$$B_C = \sqrt{1 - M_N^2 \cos^2 (\Lambda_{c/4})_W}$$

for  $U > 15$  ft/sec,

Y - Force in the (local) wind axis system:

$$Y_{WP} = q_F S_{\infty W} \left[ C_{Y\beta} \beta_F + \frac{b_W}{2U} (C_{Y_p} p' + C_{Y_r} r') \right]$$

Roll and yaw moments in (local) wind axis:

$$l'_{WP} = q_F S_{\infty W} b_W \left[ C_{l\beta} \beta_F + \frac{b_W}{2U} (C_{l_p} p' + C_{l_r} r') + C_{l\delta_a} \delta_a \right]$$

$$N'_{WP} = q_F S_{\infty W} b_W \left[ C_{n\beta} \beta_F + \frac{b_W}{2U} (C_{n_p} p' + C_{n_r} r') + C_{n\delta_a} \delta_a \right]$$

where the aircraft angular velocities in wind axis system are,

$$p' = p \cos \alpha_W \cos \beta_F + q \sin \beta_F + r \sin \alpha_W \cos \beta_F$$

$$r' = -p \sin \alpha_W + r \cos \alpha_W$$

and the lateral-directional stability derivatives are,

$$1. C_{Y\beta} = \left( C_{Y\beta} \Big|_{M_N = 0} \right) (C_\beta)$$

$$2. C_{Y_p} = \left( \frac{C_{Y_p}}{C_L} \Big|_{M_N = 0} \right) (C_{L_{WP}}) (C_\beta) \left( \frac{AR_W B_C + \cos(\Lambda_{c/4W})}{AR_W + \cos(\Lambda_{c/4W})} \right)$$

$$3. C_{Y_r} = \left( C_{Y_r} \Big|_{M_N = 0} \right) (C_\beta)$$

$$4. C_{l\beta} = \left( C_{l\beta} \Big|_{\substack{C_L = 0 \\ M_N = 0}} \right) (C_\beta) + \left( \frac{C_{l\beta}}{C_{L_{WP}}} \Big|_{M_N = 0} \right) C_{L_{WP}}$$

$$\left( C_{l\beta} \Big|_{\substack{C_L = 0 \\ M_N = 0}} \right) = 0 \text{ for } |\beta_F| > 15 \text{ Deg}$$

$$5. \quad C_{1P} = \left[ C_{1P} \left| \begin{array}{l} C_L = 0 \\ M_N = 0 \end{array} \right. \right] (C_\beta) * \left[ \begin{array}{l} \frac{\partial C_{LWP}}{\partial \alpha_W} \bigg|_{\alpha_W} \\ \frac{\partial C_{LWP}}{\partial \alpha_W} \bigg|_{C_{LWP} = 0} \end{array} \right] - \frac{1}{8} \left[ C_{DWP} \quad C_{LWP}^2 \right] \bigg|_{\pi AR_W}$$

$$6. \quad C_{1r} = \left\{ \left( 1 + \frac{AR_W (1 - B_c)^2}{2B_c (AR_W B_c + 2)} \right) \frac{C_{1r}}{C_{LWP}} \bigg|_{M_N = 0} \right\} C_{LWP} + \left[ \frac{\Delta C_{1r}}{\frac{d\alpha}{d\delta_f}} \delta_f \right] \left( \frac{d\alpha}{d\delta_f} \right) \delta_f$$

If  $F_1$  then  $\delta_F = 0^\circ$

If  $F_2$  then  $\delta_F = 30^\circ$

If  $F_3$  then  $\delta_F = 60^\circ$

If  $F_4$  then  $\delta_F = -15^\circ$

$$7. \quad C_{1\delta_a} = K_{1\delta_a} * C_{1\delta_a} \bigg|_{\substack{F_1 \\ \alpha_W < 8^\circ}}$$

Where,

$$K_{1\delta_a} = f(F_x, \beta_M, \alpha_W)$$

$$8. \quad C_{n\beta} = \left( C_{n\beta} \bigg|_{\substack{C_L = 0 \\ M_N = 0}} \right) * (C_\beta) + \left( \frac{C_{n\beta}}{C_{LWP}^2} \right) \bigg|_{M_N = 0} * (C_{LWP})^2$$



$$9. \quad C_{n_p} = -C_{l_p} \alpha_w - K_{n_p} \left[ -C_{l_p} \alpha_w - \left( \frac{C_p}{\beta} \right) \frac{AR_w B_c}{AP_w} \frac{C_{np}}{C_{L_{WP}}} \right]_{\alpha_w = 0} * C_{L_{WP}}$$

$$10. \quad C_{n_r} = \left( \frac{C_{n_r}}{C_{L_{WP}}^2} \right) (C_{L_{WP}})^2 + \left( \frac{C_{n_r}}{C_{D_{0WP}}} \right) C_{D_{0WP}} \Big|_{\alpha_w = 0}$$

$$11. \quad C_{n_{\delta_a}} = K_{n_{\delta_a}} + K_{n_{\delta_a}} C_{l_{\delta_a}} C_{L_{WP}}$$

D. Wing Wake Deflection at Horizontal Tail is a function of angle of attack at the inboard section of the wing, mast tilt, flap deflection derived from test data.

$$\epsilon_{W/H} = f(\alpha_w, \beta_M, F_x)$$

**OUTPUTS:**

Subsystem Block No.	Symbol	Units
(5)	$\epsilon_{W/H}$	Deg
(10a)	$\alpha'_{F/W}, \beta'_{F/W}$	Deg
	$L_{iWPL}$	Lbs
	$L_{iWPR}$	Lbs
	$D_{iWPL}$	Lbs
	$D_{iWPR}$	Lbs
	$\alpha_W$	Deg
	$L_{WP}$	Lbs
	$D_{WP}$	Lbs
	$M'_{OWP}$	Ft-Lbs
	$Y'_{WP}$	Lbs
	$l'_{WP}$	Ft-Lbs
	$N'_{WP}$	Ft-Lbs

**SUBSYSTEM NO. 5-HORIZONTAL STABILIZER AERODYNAMICS**

**INPUTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
9	$SL_{CG}$	In
11	$q$	Rad/Sec
2	$U_i   \frac{B}{R/H}$	Ft/Sec
	$W_i   \frac{B}{R/H}$	Ft/Sec
8a	$\beta_M$	Deg
	$\delta_e$	Deg
	$i_H$	Deg
12	$U$	Ft/Sec
	$V$	Ft/Sec
	$W$	Ft/Sec
	$\dot{\alpha}_F$	Rad/Sec
	$\beta_F$	Rad
15	$M_N$	ND
	$\rho$	Slug/ft <sup>3</sup>

**B. Aerodynamic Coefficients**

$\tau_H$	ND
$K_e$	ND
$C_{L_H}, C_{L_H\beta}$	ND, 1/Deg
$C_{D_H}$	ND
$C_{M_H}$	ND
$\tau_e$	ND

EQUATIONS

Body-axes velocity components due to vector  $(\vec{v}_T + \vec{w}_i|_{R/H})$

$$U_{R/H} = U + U_i|_{R/H}^B$$

$$W_{R/H} = W + W_i|_{R/H}^B$$

$U_{R/H}$  and  $W_{R/H}$  affected by wing wake ( $\epsilon_{W/H}$ ) and body pitch rate ( $q$ ),

$$U_H = U_{R/H} \cos(\epsilon_{W/H}) + W_{R/H} \sin(\epsilon_{W/H})$$

$$W_H = -U_{R/H} \sin(\epsilon_{W/H}) + W_{R/H} \cos(\epsilon_{W/H}) + l_{XH} q$$

where,

$$\epsilon_{W/H} = f(\alpha_W, \beta_M, F_x, M_N)$$

$$l_{XH} = (SL_H - SL_{CG}) \frac{l}{12}$$

Total velocity:

$$V_{HT} = \sqrt{U_H^2 + V^2 + W_H^2}$$

Angle of attack for lift equation:

$$\alpha_{HL} = i_H + \tan^{-1} \left( \frac{W_H}{U_H} \right) \quad (\text{for } M_N < 1.0)$$

$$= i_H + \tan^{-1} \left( \frac{W_H}{U_H} \right) - (K_e \tau_e) \delta_e \quad (\text{for } M_N > 1.0)$$

Angle of attack for drag equation:

$$\alpha_{HD} = i_H + \tan^{-1} \left( \frac{W_H}{U_H} \right) - (K_e \tau_e) \delta_e$$

Dynamic pressure:

$$q_H = \frac{1}{2} \rho v_{HT}^2 (\eta_H)$$

Lift in (local) wind axes:

For  $U < 67.5$  ft/sec, set  $\dot{\alpha}_F = 0$ ; if  $U \leq 35$ , set  $U = 35$  ft/sec

$$L_H = q_H S_H \left\{ C_{L_H}(\alpha_{HL}, \delta_e, M_N) + \frac{l_{XH}}{U} \left( \frac{\partial C_{LH}}{\partial \alpha_{HL}} \right) \left( \frac{\partial \epsilon_{W/H}}{\partial \alpha_W} \right) \dot{\alpha}_F + C_{L_{HB}} |\beta_F| \right\}$$

For  $|\beta_F| > 15^\circ$ ,  $L_{H\beta} = L_{H\beta}$  AT  $|\beta_F| = 15^\circ$

Drag in (local) wind axes:

$$D_H = q_H S_H C_{D_H}(\alpha_{HD}, M_N)$$

Local angle of attack (for resolving forces)

$$\alpha_H = \tan^{-1} \left( \frac{W_H}{U_H} \right)$$

Pitching moment:

$$M'_H = q_H S_H C_H C_{M_H} f(\alpha_H, M_N)$$

**OUTPUTS:**

Subsystem Block No.

10a



Symbol

$\alpha_H$

$L_H$

$D_H$

$M'_H$

Units

Deg

Lbs

Lbs

Ft-lbs

**SUBSYSTEM NO. 6-VERTICAL STABILIZER AERODYNAMICS**

**INPUTS:**

**A. Variables**

Subsystem Block No.

Symbol

Units

12

U

Ft/sec

↓

V

Ft/sec

↓

W

Ft/sec

11

P

Rad/sec

↓

q

Rad/sec

↓

r

Rad/sec

2

$U_i \Big|_{R/V}^B$

Ft/sec

↓

$W_i \Big|_{R/V}^B$

Ft/sec

15

$M_N$

ND

↓

$\rho$

Slug/ft<sup>3</sup>

↓

$\delta_R$

Deg

8a

$\beta_M$

Deg

↓

$F_x$

$F_1, F_2, F_3, F_4$

12

$\beta_F$

Deg

↓

$\alpha_H$

Deg

9

$SL_{CG}$

In.

$WL_{CG}$

In.

**B. Aerodynamic Coefficients**

$(1 - \frac{\partial \sigma}{\partial \beta_F})$

ND

$C_{Y,V} \Big|_{\beta_M = 90}^{F_1}$

ND

$C_{D,V}$

ND

$\frac{\partial \sigma}{\partial \rho}, \frac{\partial \sigma}{\partial r}$

ND

r

ND

EQUATIONS

The H-tail is represented as a left and right fin. Velocity at the left fin,

$$V_{VL}^2 = \left\{ (U + U_i \Big|_{R/V}^B - q * l_{ZV} + r * l_{YV})^2 + (V - r * l_{XV} + p * l_{ZV})^2 \right. \\ \left. + (W - W_i \Big|_{R/V}^B + q * l_{XV} - p * l_{YV})^2 \right\}$$

$$\left. \begin{array}{l} U_i \Big|_{R/V}^B = 0 \\ W_i \Big|_{R/V}^B = 0 \end{array} \right\} \text{for } \left\{ \begin{array}{l} 5^\circ < \beta_F < 28^\circ \\ |\beta_F| > 60^\circ \end{array} \right.$$

where,

$$l_{XV} = (SL_V - SL_{CG}) \frac{1}{12}$$

$$l_{YV} = (BL_V - BL_{CG}) \frac{1}{12}$$

$$l_{ZV} = (WL_V - WL_{CG}) \frac{1}{12}$$

Velocity at right fin,

$$V_{VR}^2 = \left\{ (U + U_i \Big|_{R/V}^B - q * l_{ZV} - r * l_{YV})^2 \right. \\ \left. + (V - r * l_{XV} + p * l_{ZV})^2 \right. \\ \left. + (W - W_i \Big|_{R/V}^B + q * l_{XV} + p * l_{YV})^2 \right\}$$

$$\left. \begin{array}{l} U_i \Big|_{R/V}^B = 0 \\ W_i \Big|_{R/V}^B = 0 \end{array} \right\} \text{for } \left\{ \begin{array}{l} -5^\circ > \beta_F > -28^\circ \\ |\beta_F| > 60^\circ \end{array} \right.$$

Define sideslip angle due to vector  $(\vec{V}_T + \vec{W}_i \Big|_{R/V})$

$$\beta'_{VL,R} = \tan^{-1} \left\{ \frac{V}{\sqrt{(U + U_i \Big|_{R/V}^B)^2 + (W + W_i \Big|_{R/V}^B)^2}} \right\}$$



Define Zero Rudder Sideslip Angle, if  $(U + U_i \left| \frac{B}{R/V} \right|) \leq 10$ , set = 10.0 ft/sec

$$\beta_{VL,R_0} = \left\{ \beta_{VL,R} \left( 1 - \frac{\partial \sigma}{\partial \beta} \right) + \frac{b_W}{2(U + U_i \left| \frac{B}{R/V} \right|)} \left[ \left( \frac{2 L_{ZV}}{b_W} - \frac{\partial \sigma}{\partial \beta} \right) P - \left( \frac{2 L_{XV}}{b_W} + \frac{\partial \sigma}{\partial r} \right) r \right] \right\} \quad (57.3)$$

Sideslip Angle for Y Force Equation,

$$\begin{aligned} \beta_{VYL,R} &= \beta_{VL,R_0} \quad (\text{For } M_N < 1.0) \\ &= \beta_{VL,R_0} + (K_{R\tau R}) * \delta_R \quad (\text{for } M_N > 1.0) \end{aligned}$$

Sideslip Angle for Drag Equation,

$$\beta_{VDL,R} = \beta_{VL,R_0} + (K_{R\tau R}) * \delta_R$$

Dynamic Pressure,

$$q_{VL} = (1/2 \rho v_{VL}^2) \eta_V \quad \text{and} \quad q_{VR} = (1/2 \rho v_{VR}^2) \eta_V$$

Forces in the local wind axes are,

$$Y'_{VL} = \frac{1}{2} S_V q_{VL} C_{Y_V}$$

$$Y'_{VR} = \frac{1}{2} S_V q_{VR} C_{Y_V}$$

$$D_{VL} = \frac{1}{2} S_V q_{VL} C_{D_V}$$

$$D_{VR} = \frac{1}{2} S_V q_{VR} C_{D_V}$$

OUTPUTS:

Subsystem Block No.

10a



Symbol

$\beta_{VL'R_0}$

$Y'_{VL}$

$Y'_{VR}$

$D_{VL}$

$D_{VR}$

Units

Deg

Lbs

Lbs

Lbs

Lbs

**SUBSYSTEM NO. 7: RETRACTABLE LANDING GEAR**

**INPUTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
(8a)	$L_{LG}$	ND
(10f)	$h$	Ft
(12)	$\alpha_F$	Deg
↓	$V_T, W_G$	Ft/Sec
(15)	$\rho$	Slug/Ft <sup>3</sup>
(10c)	$\theta, \phi$	Deg

**B. Aerodynamic Coefficients**

$D_{OMGU}(t)$	$D_{OMGD}(t)$
$D_{ONGU}(t)$	$D_{ONGD}(t)$

**C. Strut Force Coefficients**

$K_{ST1}$	$K_{ST2}$	$K_{ST3}$
$D_{ST1}$	$D_{ST2}$	$D_{ST3}$
$B_{G1}$	$B_{G2}$	$B_{G3}$
$\mu_o$	$\mu_1$	$\mu_s$

Landing Gear Locations

$$X_n = SL_{CG} - SL_{Gn}$$

$$Y_n = BL_{CG} - BL_{Gn}$$

$$Z_n = WL_{CG} - WL_{Gn}$$

Butt lines positive to right. Water lines defined with zero loads in landing gears.

Where,

$$n = 1, 2, 3$$

1 = Left Main Gear

2 = Right Main Gear

3 = Nose Gear

Aerodynamic Force Equations

$$q_F = \frac{1}{2} \rho V_T^2$$

A. Gears Down

$$D_{MG} = q_F D_{MOGD}(t)$$

$$D_{NG} = q_F D_{ONGD}(t)$$

B. Gears Up

$$D_{MG} = q_F D_{OMGU}(t)$$

$$D_{NG} = q_F D_{ONGU}(t)$$

Strut Force Equations

Strut Deflection

$$h_{G\theta n} = X_n \sin \theta - Z_n \cos \theta - Y_n$$

$$h_{G\phi n} = [Y_n \sin \phi + (Z_n + Y_n) (\cos \phi - 1)] \cos \theta$$

$$h_{Tn} = (h + h_{G\theta n} - h_{G\phi n}) / (\cos \theta \cos \phi)$$

Strut Deflection Rate

$$\dot{h}_{Tn} = + W_G \left( \frac{1}{\cos \theta \cos \phi} \right) + X_n q - Y_n p$$

Strut Vertical Force

IF  $h_{Tn} > 0$ ,  $F_{GZn} = 0$

IF  $h_{Tn} < 0$ ,

$$F_{GZn} = K_{STn} h_{Tn} + D_{STn} \dot{h}_{Tn}$$

Strut Longitudinal Force

$$F_{\mu n} = - (\mu_o + \mu_i B_{Gn}) F_{GZn} \frac{U}{|U|}$$

IF  $U = 0$ ,  $F_{\mu n} = 0$ .

Strut Side Force

IF  $V = 0$ ,  $F_{Sn} = 0$

$$\text{IF } V \neq 0, F_{Sn} = -\mu_s F_{GZn} \frac{V}{|V|}$$

Force and Moment Contributions of Gear Struts

$$\Delta X_n = F_{\mu n} - F_{GZn} \theta$$

$$\Delta Y_n = F_{S_n} + F_{GZn} \phi$$

$$\Delta Z_n = F_{\mu n} \theta - F_{S_n} \phi + F_{GZn}$$

$$\Delta M_n = -\Delta Z_n X_n + \Delta X_n (Z_n + \gamma_n + h_{Tn})$$

$$\Delta I_n = -\Delta Z_n Y_n - \Delta Y_n (Z_n + \gamma_n + h_{Tn})$$

$$\Delta N_n = -\Delta X_n Y_n + X_n \Delta Y_n$$

$$\Delta X_{LG} = \sum_1^3 \Delta X_n$$

$$\Delta Y_{LG} = \sum_1^3 \Delta Y_n$$

$$\Delta Z_{LG} = \sum_1^3 \Delta Z_n$$

$$\Delta I_{LG} = \sum_1^3 \Delta I_n$$

$$\Delta M_{LG} = \sum_1^3 \Delta M_n$$

$$\Delta N_{LG} = \sum_1^3 \Delta N_n$$

OUTPUTS

Subsystem Block No.

Symbol

Units

10a

$D_{NG}$

Lbs



$D_{NG}$

Lbs

13

$\Delta X_{LG}$

Lbs



$\Delta Y_{LG}$

Lbs

$\Delta Z_{LG}$

Lbs

14

$\Delta l_{LG}$

Lb. ft



$\Delta M_{LG}$

Lb. ft

$\Delta N_{LG}$

Lb. ft

**SUBSYSTEM NO. 8a CONTROLS MIXER**

**INPUTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
(17)	$\theta_{OL/G}, \theta_{OR/G}$	Deg
(8d)	$X_{LN}$	Inch
↓	$X_{LT}$	Inch
	$X_{PD}$	Inch
(20)	$X_{COL}$	Deg
	ESAS	Deg
	RSAS	Deg
↓	ASAS	Deg
	(8d)	$X_{PM}$
↓	$X_{FL}$	$F_1, F_2, F_3, F_4$
	$X_{LG}$	UP, DN
(11)	$V_T$	Ft/Sec

**B. Control System Gearing**

$\partial B_1 / \partial X_{LN}, \partial B_1 / \partial X_{PD}$	Deg/In, Deg/In
$\partial \theta_o / \partial X_{LT}, \partial \theta_o / \partial X_{COL}, \theta_{o-LL}$	Deg/In, Deg/In, Deg
$\partial \delta_r / \partial X_{LN}, \partial \delta_r / \partial X_{PD}$	Deg/In, Deg/In
$\partial \delta a / \partial X_{LT}$	Deg, In

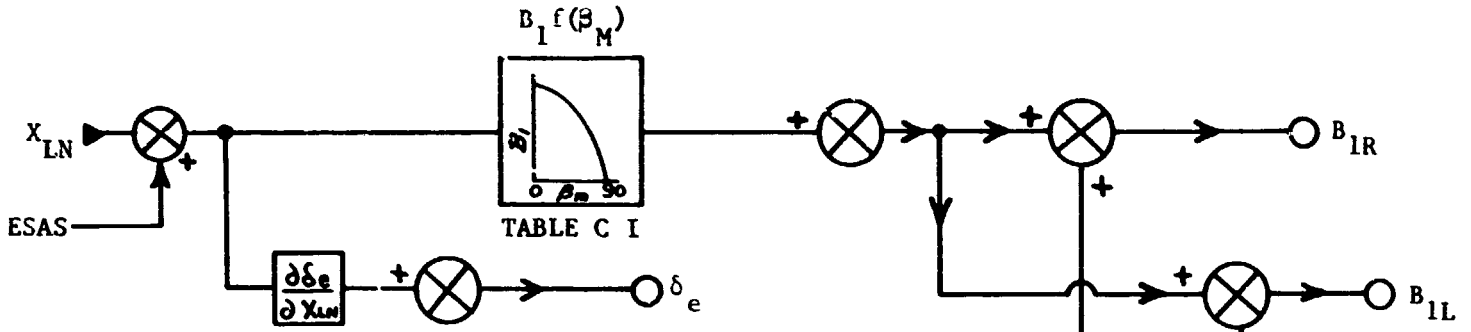


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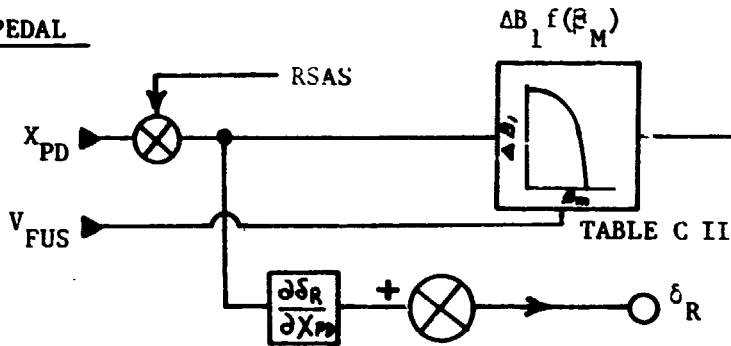
7872 55426

# CONTROL SYSTEM BLOCK DIAGRAM

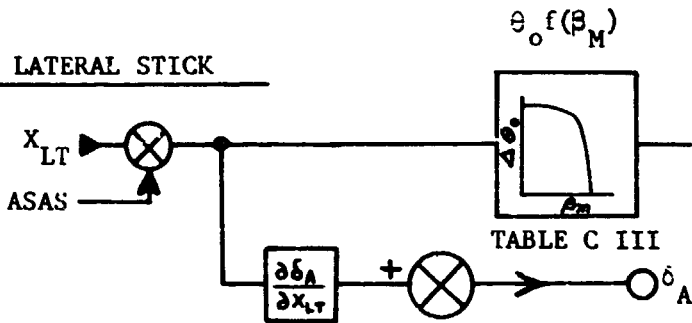
## LONGITUDINAL STICK



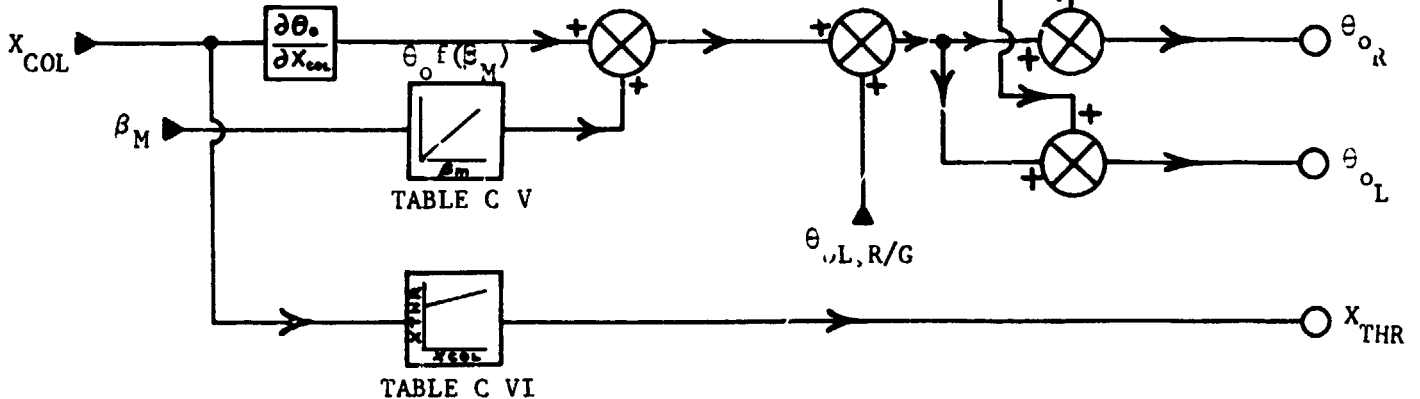
## PEDAL



## LATERAL STICK

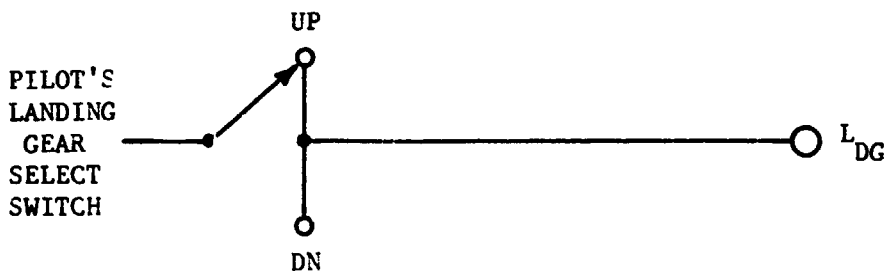
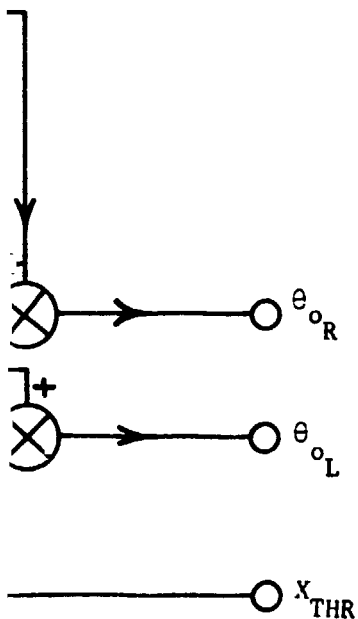
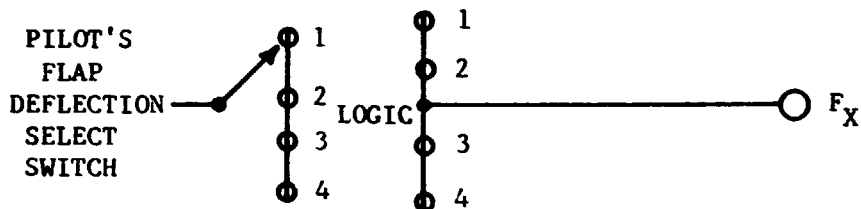
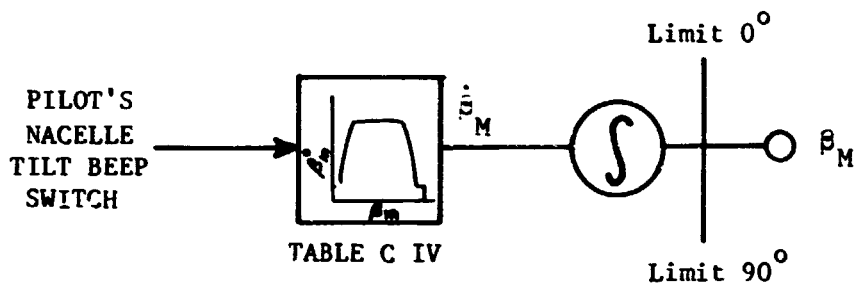
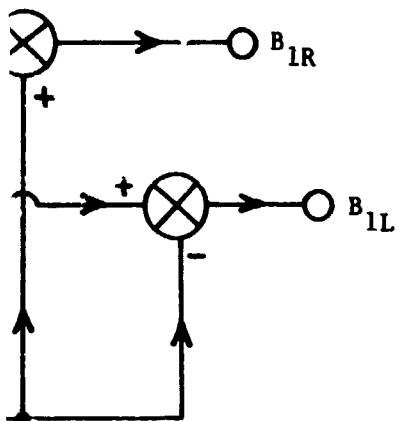


## COLLECTIVE STICK



# FOLDOUT FRAME 2

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**EQUATIONS**

**A. Collective Pitch**

$$\theta_{OR} = \left( \frac{\partial \theta_0}{\partial X_{COL}} \right) * X_{COL} + \theta_{OLL} + \theta_{OR/G}$$

$$- \left( \frac{\partial \theta_0}{\partial X_{LT}} \right) * (X_{LT} - X_{LTN} + ASAS)$$

$$\theta_{OL} = \left( \frac{\partial \theta_0}{\partial X_{COL}} \right) * X_{COL} + \theta_{OLL} + \theta_{OL/G}$$

$$+ \left( \frac{\partial \theta_0}{\partial X_{LT}} \right) * (X_{LTN} + ASAS)$$

**B. Longitudinal Cyclic**

$$B_{1R} = \left( \frac{\partial B_1}{\partial X_{LN}} \right) * (X_{LN} - X_{LNN} + ESAS)$$

$$- \left( \frac{\partial B_1}{\partial X_{PD}} \right) * (X_{PD} - X_{PDN} + RSAS)$$

$$B_{1L} = \left( \frac{\partial B_1}{\partial X_{LN}} \right) * (X_{LN} - X_{LNN} + ESAS)$$

$$+ \left( \frac{\partial B_1}{\partial X_{PD}} \right) * (X_{PD} - X_{PDN} + RSAS)$$

C. Elevator, Rudder, Aileron

$$\delta_e = \frac{\partial \delta_e}{\partial X_{LN}} * (X_{LN} - X_{LNN} + ESAS)$$

$$\delta_r = \frac{\partial \delta_r}{\partial X_{PD}} *(X_{PD} - X_{PDN} + RSAS)$$

$$\delta_a = \frac{\partial \delta_a}{\partial X_{LT}} *(X_{LT} - X_{LTN} + ASAS)$$

D. Nacelle Tilt

$$\dot{\beta}_M = f(\beta_M) * (1, 0, -1) \text{ (Fwd, Neutral, Aft)}$$

$$\beta_M = \int_0^t \dot{\beta}_M dt \quad \text{Limits } 90^\circ \leq \beta_M \leq -5^\circ$$

E. Flap Selector

Flap/Flaperon Setting is:

F <sub>1</sub>	(0,0)	
F <sub>2</sub>	(40,25)	Use to select table
F <sub>3</sub>	(75,47)	
F <sub>4</sub>	(-28, -17.5)	

F. Landing Gear Selector

$$L_{DG} = [0,1] \text{ (Up, Down)}$$

G. Variable Incidence Stabilizer

$$i_H = i_{H0} \left| \begin{array}{l} X_{LN} = X_{LNN} \\ X_{COL} = 0 \end{array} \right. + i_{H1} (X_{LN} - X_{LNN}) + i_{H2} (X_{LN} - X_{LNN})^2 + i_{H3} \left| \begin{array}{l} X_{LN} = X_{LNN} \\ X_{COL} = 0 \end{array} \right. + i_{H4} X_{COL}$$

**OUTPUTS:**

Subsystem Block No.	Symbol	Units
	$\theta_{OR}$	Deg
	$\theta_{OL}$	Deg
	$B_{1L}$	Deg
	$B_{1R}$	Deg
	$\delta_e$	Deg
	$i_H$	Deg
	$\delta_R$	Deg
	$\delta_a$	Deg
	$F_X$	( $F_1, F_2, F_3, F_4$ )
	$L_{DG}$	ND
	$\dot{\beta}_M$	Deg/Sec
	$\beta_M$	Deg

**SUBSYSTEM NO. 8b: FORCE FEEL SYSTEM**

**INPUTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
8d ↓	$X_{LN}$	Inch
	$X_{LT}$	Inch
8c ↓	$X_{PD}$	Inch
	$X_{LNT}$	Inch
	$X_{LTT}$	Inch
	$X_{PDT}$	Inch
	$V_T$	Knots

**B. Stick Force Coefficients**

$F_{LNO}, F_{LTO}, F_{PDO}$	Lb/In
$F_{LN1}, F_{LN2}$ $F_{LT1}, F_{LT2}$ $F_{PD1}, F_{PD2}$	Lb/In/Ft/Sec
$F_4$	Lb/In
$V_1, V_2$	Kts



EQUATIONS:

$$F_{LN} = \left[ F_{LN_0} + F_{LN_1} (v_T - v_1) + F_{LN_2} (v_T - v_2) \right] * | (X_{LN} - X_{LNT}) |$$

$$F_{LT} = \left[ F_{LT_0} + F_{LT_1} (v_T - v_1) + F_{LT_2} (v_T - v_2) \right] * | (X_{LT} - X_{LTT}) |$$

$$F_{PD} = \left[ F_{PD_0} + F_{PD_1} (v_T - v_1) + F_{PD_2} (v_T - v_2) \right] * | (X_{PD} - X_{PDT}) |$$

$$F_{COL} = F_4 * | X_{COL} |$$

OUTPUTS:

Subsystem Block No.

8d



Symbol

Units

F<sub>LN</sub>

Pounds

F<sub>LT</sub>

Pounds

F<sub>PD</sub>

Pounds

F<sub>COL</sub>

Pounds

SUBSYSTEM NO. 8c: CONTROL FORCE TRIM SYSTEM

INPUTS:

A. Variables

Subsystem Block No.

Symbol

Units

8d



X<sub>LN</sub>

Inch

X<sub>LT</sub>

Inch

X<sub>PD</sub>

Inch

B<sub>FT</sub>

On or Off



EQUATIONS:

If trim button is depressed ( $B_{FT}$  ON)

$$X_{LNT} = X_{LN}$$

$$X_{LTT} = X_{LT}$$

$$X_{PDT} = X_{PD}$$

If trim button is not depressed ( $B_{FT}$  OFF)

$$X_{LNT} = X_{LNT}$$

$$X_{LTT} = X_{LTT}$$

$$X_{PDT} = X_{PDT}$$

OUTPUTS:

Subsystem Block No.

Symbol

Units

8b



X<sub>LNT</sub>

Inch

X<sub>LTT</sub>

Inch

X<sub>PDT</sub>

Inch

**SUBSYSTEM NO. 8d: PILOT'S CONTROL FUNCTION**

INPUTS:

A. Variables

Subsystem Block No.	Symbol	Units
<div style="border: 1px solid black; border-radius: 50%; width: 30px; height: 30px; display: flex; align-items: center; justify-content: center;">8b</div> <div style="text-align: center; margin-top: 5px;">↓</div>	$F_{LN}$	Pounds
	$F_{LT}$	Pounds
	$F_{PD}$	Pounds
	$F_{COL}$	Pounds

EQUATIONS:

Pilot in the Loop

OUTPUTS:

Subsystem Block No.	Symbol	Control Mechanism	Units
<div style="border: 1px solid black; border-radius: 50%; width: 30px; height: 30px; display: flex; align-items: center; justify-content: center;">8a</div> <div style="text-align: center; margin-top: 5px;">↓</div> <div style="display: flex; flex-direction: column; align-items: center; margin-top: 10px;"> <div style="border: 1px solid black; border-radius: 50%; width: 30px; height: 30px; display: flex; align-items: center; justify-content: center;">17</div> <div style="border: 1px solid black; border-radius: 50%; width: 30px; height: 30px; display: flex; align-items: center; justify-content: center; margin-top: 5px;">8c</div> </div>	$X_{LN}$	Long. Stick	Inch
	$X_{LT}$	Lat. Stick	Inch
	$X_{PD}$	Pedal	Inch
	$X_{COL}$	Coll. Stick	Inch
	$X_{PM}$	Beep Switch	Fwd, Neut., Aft
	$X_{FL}$	Grip (Floor Mounted) Switch	$F_X$ Up, Down
	$RPM_p$	Beep	RPM
$B_{FT}$	Switch	ON/OFF	

Subsystem Block No.



Symbol	Control Mechanism	Units
X <sub>PSAS</sub>	Switch	ON/OFF
X <sub>RSAS</sub>	Switch	ON/OFF
X <sub>YSAS</sub>	Switch	ON/OFF

**SUBSYSTEM NO. 9-CG AND INERTIA SHIFT WITH PYLON TILT**

INPUTS

**A. Variables**

Subsystem Block No.

Symbol

Units



$\beta_M$

Deg.

$\dot{\beta}_M$

Rad/Sec.



h

Ft.

**B. Inertia Coefficients**

$K_{I1}, K_{I2}, K_{I3}, K_{I4}$

$\frac{\text{Slugs} - \text{Ft}^2}{\text{Deg.}}$



EQUATIONS

CG Displacement as a Function of Pylon Tilt Angle

$$X_{CG} = Z \sin \beta_M + X (1 - \cos \beta_M)$$

$$Z_{CG} = z (1 - \cos \beta_M) - X \sin \beta_m$$

Where,

$$X = \frac{W_P}{GW} (SL_{SP} - SL_P)$$

$$Z = \frac{W_P}{GW} (WL_{SP} - WL_P)$$

CG Location,

$$SL_{CG} = SL_{CG} \Big|_{\beta_m = 0} + X_{CG}$$

$$WL_{CG} = WL_{CG} \Big|_{\beta_m = 0} + Z_{CG}$$

Rotor Hub Height From Ground,

$$h_H = h + \left\{ l_m \cos \beta_m + \frac{(WL_{SP} - WL_{CG})}{12} \right\}$$

CG Velocity Due to Pylon Tilt Rate,

$$\dot{X}_{CG} = Z [\dot{\beta}_M * \cos \beta_M] + X [\dot{\beta}_M * \sin \beta_M]$$

$$\dot{Z}_{CG} = Z [\dot{\beta}_M * \sin \beta_M] - X [\dot{\beta}_M * \cos \beta_M]$$

CG Acceleration (Neglecting  $\ddot{\beta}_M$ )

$$\ddot{x}_{CG} = -z [\dot{\beta}_M * \sin \beta_M] + x [\dot{\beta}_M^2 * \cos \beta_M]$$

$$\ddot{z}_{CG} = z [\dot{\beta}_M^2 * \cos \beta_M] + x [\dot{\beta}_M^2 * \sin \beta_M]$$

Aircraft Inertia Change Due to Pylon Tilt

$$I_{XX} = I_{XX}|_{\beta_M = 0} - K_{I1} \beta_M$$

$$I_{YY} = I_{YY}|_{\beta_M = 0} - K_{I2} \beta_M$$

$$I_{ZZ} = I_{ZZ}|_{\beta_M = 0} + K_{I3} \beta_M$$

$$I_{XZ} = I_{XZ}|_{\beta_M = 0} - K_{I4} \beta_M$$

OUTPUTS

Subsystem Block No.



Symbol	Units
$SL_{CG}$	Inch
$WL_{CG}$	Inch
$X_{CG}$	Inch
$Z_{CG}$	Inch
$\dot{X}_{CG}$	Inch/sec
$\dot{Z}_{CG}$	Inch/sec
$\ddot{X}$	Inch/sec <sup>2</sup>
$\ddot{Z}_{CG}$	Inch/sec <sup>2</sup>
$I_{XX}$	Slug - Ft <sup>2</sup>
$I_{YY}$	Slug - Ft <sup>2</sup>
$I_{ZZ}$	Slug - Ft <sup>2</sup>
$I_{XZ}$	Slug - Ft <sup>2</sup>
$h_H$	Ft

**SUBSYSTEM NO. 10: AXES TRANSFORMATIONS**

**10a. Transformation of Airframe Aerodynamic Forces and Moments  
from Wind to Body Axes**

**INPUTS:**

Subsystem Block No.

Symbol

Units

3

$\alpha_F$

Rad

$\beta_F$

Rad

$L_F$

Lbs.

$D_F$

Lbs

$Y'_F$

Lbs

$M'_F$

Ft-Lbs

$l'_F$

Ft-Lbs

$N'_F$

Ft-Lbs

4

$\alpha'_{F/W}$

Rad

$\beta'_{F/W}$

Rad

$L_{iWPL}$

Lbs

$L_{iWPR}$

Lbs

$D_{iWPL}$

Lbs

$D_{iWPR}$

Lbs

$\alpha_W$

Rad

$L_{WP}$

Lbs

$D_{WP}$

Lbs

$M'_{ONP}$

Ft-Lbs

$Y'_{WP}$

Lbs

$l'_{WP}$

Ft-Lbs

$N'_{WP}$

Ft-Lbs

$N'_{WP}$

Ft-Lbs

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Subsystem Block No.

Symbol

Units

5

$\alpha_H$

Rad



$L_H$

Lbs

$D_H$

Lbs

$M'_H$

Ft-Lbs

6

$\beta_{VL} Ro$

Rad



$Y'_{VL}$

Lbs

$Y'_{VR}$

Lbs

$D_{VL}$

Lbs

$D_{VR}$

Lbs

7

$D_{MG}$

Lbs



$D_{NG}$

Lbs

**EQUATIONS:**

**A. General Form of Transformation**

$$\begin{bmatrix} X_i \\ Y_i \\ Z_i \end{bmatrix}_{\text{Body}} = \begin{bmatrix} \cos \alpha_i \cos \beta_i - \cos \alpha_i \sin \beta_i - \sin \alpha_i & & \\ & \sin \beta_i & \cos \beta_i & 0 \\ \sin \alpha_i \cos \beta_i - \sin \alpha_i \sin \beta_i & \cos \alpha_i & & \end{bmatrix} \begin{bmatrix} X_i \\ -Y_i \\ Z_i \end{bmatrix}_{\text{Wind}}$$

This transformation matrix is also used for the moment transformation.  $\alpha_i$  and  $\beta_i$  are the component angle of attack and sideslip angle, respectively.

**B. Transformation of Fuselage Forces and Moments**

$$X_F = -D \cos \alpha_F \cos \beta_F - Y'_F \cos \alpha_F \sin \beta_F + L_F \sin \alpha_F$$

$$Y_F = -D_F \sin \beta_F + Y'_F \cos \beta_F$$

$$Z_F = -D_F \sin \alpha_F \cos \beta_F - Y'_F \sin \alpha_F \sin \beta_F - L_F \cos \alpha_F$$

$$L_F = l'_F \cos \alpha_F \cos \beta_F - M'_F \cos \alpha_F \sin \beta_F - N'_F \sin \alpha_F$$

$$M_F = l'_F \sin \beta_F + M'_F \cos \beta_F$$

$$N_F = l'_F \sin \alpha_F \cos \beta_F - M'_F \sin \alpha_F \sin \beta_F + N'_F \cos \alpha_F$$

### C. Transformation of Wing Forces and Moments

#### 1. Forces Generated by Rotor Wake

$$X_{iWPR} = -D_{iWPR} \cos \alpha'_{F/W} \cos \beta'_{F/W} + L_{iWPR} \sin \alpha'_{F/W}$$

$$Y_{iWPR} = -D_{iWPR} \sin \beta'_{F/W}$$

$$Z_{iWPR} = -D_{iWPR} \sin \alpha'_{F/W} \cos \beta'_{F/W} - L_{iWPR} \cos \alpha'_{F/W}$$

#### 2. Forces and Moments Generated by Freestream Flow

$$X_{WP} = -D_{WP} \cos \alpha_W \cos \beta_F - Y'_{WP} \cos \alpha_W \sin \beta_F + L_{WP} \sin \alpha_W$$

$$Y_{WP} = -D_{WP} \sin \beta_F + Y'_{WP} \cos \beta_F$$

$$Z_{WP} = -D_{WP} \sin \alpha_W \cos \beta_F - Y'_{WP} \sin \alpha_W \sin \beta_F - L_{WP} \cos \alpha_W$$

$$L_{WP} = l'_{WP} \cos \alpha_W \cos \beta_F - m'_{O_{WP}} \cos \alpha_W \sin \beta_F - n'_{WP} \sin \alpha_W$$

$$M_{WP} = l'_{WP} \sin \beta_F + m'_{O_{WP}} \cos \beta_F$$

$$N_{WP} = l'_{WP} \sin \alpha_W \cos \beta_F - m'_{O_{WP}} \sin \alpha_W \sin \beta_F + n'_{WP} \cos \alpha_W$$

### D. Transformation of Horizontal Stabilizer Forces and Moments

$$X_H = -D_H \cos \alpha_H \cos \beta_F + L_H \sin \alpha_H$$

$$Y_H = -D_H \sin \beta_F$$

$$Z_H = -D_H \sin \alpha_H \cos \beta_F - L_H \cos \alpha_H$$

$$l_H = -M'_H \cos \alpha_H \sin \beta_F$$

$$M_H = M'_H \cos \beta_F$$

$$N_H = -M'_H \sin \alpha_H \sin \beta_F$$

E. Transformation of Vertical Stabilizer Forces.

$$X_{VR} = -D_{VR} \cos \alpha_H \cos \beta_{VR_0} + Y'_{VR} \cos \alpha_H \sin \beta_{VR_0}$$

$$Y_{VR} = -D_{VR} \sin \beta_{VR_0} - Y'_{VR} \cos \beta_{VR_0}$$

$$Z_{VR} = -D_{VR} \sin \alpha_H \cos \beta_{VR_0} + Y'_{VR} \sin \alpha_H \sin \beta_{VR_0}$$

F. Transformation of Landing Gear Aerodynamic Forces

$$X_{MG} = -D_{MG} \cos \alpha_F \cos \beta_F$$

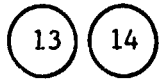
$$Y_{MG} = -D_{MG} \sin \beta_F$$

$$Z_{MG} = -D_{MG} \sin \alpha_F \cos \beta_F$$



OUTPUTS:

Subsystem Block No.



Symbol	Units
$(X, Y, Z)_F$	Lbs
$(X, Z)_{iWPL}$	Lbs
$(Z, Z)_{iWPR}$	Lbs
$(Z, Y, Z)_{WP}$	Lbs
$(X, Y, Z)_H$	Lbs
$(X, Z)_{MG}$	Lbs
$(X, Z)_{NG}$	Lbs
$(X, Y)_{VL}$	Lbs
$(X, Y)_{VR}$	Lbs
$l_F$	Ft-Lbs
$M_F$	Ft-Lbs
$N_F$	Ft-Lbs
$l_{WP}$	Ft-Lbs
$M_{WP}$	Ft-Lbs
$N_{WP}$	Ft-Lbs
$l_H$	Ft-Lbs
$M_H$	Ft-Lbs
$N_H$	Ft-Lbs

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10b. Transformation of Rotor Forces and Moments from Mast to Body Axes

INPUTS:

Subsystem Block No.	Symbol	Units
	$T_R$	Lbs
	$H_R$	Lbs
	$Y_R$	Lbs
	$Q_R$	Ft-Lbs
	$M_{a1R}$	Ft-Lbs
	$l_{b1R}$	Ft-Lbs
	$T_L$	Lbs
	$H_L$	Lbs
	$Y_L$	Lbs
	$Q_L$	Ft-Lbs
	$M_{a1L}$	Ft-Lbs
	$l_{b1L}$	Ft-Lbs
	$\theta_M$	Rad

EQUATIONS:

$$X_R = -H_R \cos \beta_M \cos \phi_M - Y_R \sin \beta_M \sin \phi_M + T_R \sin \beta_M \cos \phi_M$$

$$Y_R = H_R \sin \beta_M \sin \phi_M + Y_R \cos \phi_M + T_R \cos \beta_M \sin \phi_M$$

$$Z_R = -H_R \sin \beta_M \cos \phi_M - Y_R \cos \beta_M \sin \phi_M - T_R \cos \beta_M \cos \phi_M$$

$$X_L = -H_L \cos \beta_M \cos \phi_M - Y_L \sin \beta_M \sin \phi_M + T_L \sin \beta_M \cos \phi_M$$

$$Y_L = -H_L \sin \beta_M \sin \phi_M - Y_L \cos \phi_M - T_L \cos \beta_M \sin \phi_M$$

$$Z_L = -H_L \sin \beta_M \cos \phi_M + Y_L \cos \beta_M \sin \phi_M - T_L \cos \beta_M \cos \phi_M$$

$$L_R = l_{b1R} \cos \beta_M \cos \phi_M - M_{AIR} \sin \beta_M \sin \phi_M - Q_R \sin \beta_M \cos \phi_M$$

$$M_R = -l_{b1R} \sin \beta_M \sin \phi_M + M_{AIR} \cos \phi_M - Q_R \cos \beta_M \sin \phi_M$$

$$N_R = l_{b1R} \sin \beta_M \cos \phi_M + M_{AIR} \cos \beta_M \sin \phi_M + Q_R \cos \beta_M \cos \phi_M$$

$$L_L = -l_{b1L} \cos \beta_M \cos \phi_M - M_{AIL} \sin \beta_M \sin \phi_M + Q_L \sin \beta_M \cos \phi_M$$

$$M_L = l_{b1L} \sin \beta_M \sin \phi_M + M_{AIL} \cos \phi_M + Q_L \cos \beta_M \sin \phi_M$$

$$N_L = -l_{b1L} \sin \beta_M \cos \phi_M + M_{AIL} \cos \beta_M \sin \phi_M - Q_L \cos \beta_M \cos \phi_M$$

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OUTPUTS:

Subsystem Block No.

Symbol

Units

13

$(X, Y, Z)_R$

Lbs



$(X, Y, Z)_L$

Lbs

14

$(l, M, N)_R$

Ft-Lbs





$(l, M, N)_L$

Ft-Lbs

10c. Subsystem for Euler Angles

INPUTS:

Subsystem Block No.	Symbol	Units
	p	Rad/Sec
	q	Rad/Sec
	r	Rad/Sec
	ψ	Rad
	θ	Rad
	φ	Rad

EQUATIONS:

$$\dot{\theta} = q \cos \phi - r \sin \phi$$

$$\dot{\phi} = p + r \tan \theta \cos \phi + q \tan \theta \sin \phi$$

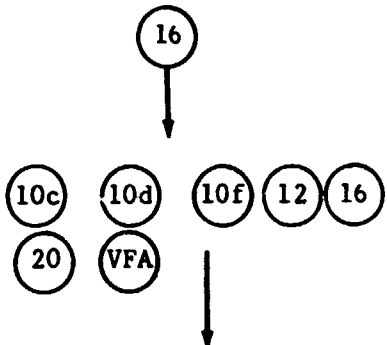
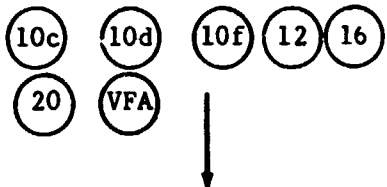
$$\dot{\psi} = \frac{r \cos \phi + q \sin \phi}{\cos \theta}$$

$$\theta = \int \dot{\theta} dt$$

$$\phi = \int \dot{\phi} dt$$


$$\psi = \int \dot{\psi} dt$$

OUTPUTS:

Subsystem Block No.	Symbol	Units
	$\dot{\psi}$	Rad/Sec
	$\dot{\theta}$	Rad/Sec
	$\dot{\phi}$	Rad/Sec
	ψ	Rad
	θ	Rad
	φ	Rad

10d. Subsystem for Earth Based Velocities

INPUTS:

Subsystem Block No.	Symbol	Units
	$\Upsilon$	Rad
	$\theta$	Rad
	$\phi$	Rad
	$U$	Ft/Sec
	$V$	Ft/Sec
	$W$	Ft/Sec

EQUATIONS:

$$U_{EB} = U \cos \Upsilon \cos \theta + V \cos \Upsilon \sin \theta \sin \phi - V \sin \Upsilon \cos \phi$$


$$+ W \cos \Upsilon \sin \theta \cos \phi + W \sin \Upsilon \sin \phi - U \text{ wind } E$$

$$V_{EB} = U \sin \Upsilon \cos \theta + V \cos \Upsilon \cos \phi + V \sin \Upsilon \sin \theta \sin \phi$$

$$+ W \sin \Upsilon \sin \theta \cos \phi - W \cos \Upsilon \sin \phi - V \text{ wind } E$$


$$W_{EB} = -U \sin \theta + V \cos \theta \sin \phi + W \cos \theta \cos \phi + W \text{ wind } E$$

OUTPUTS:

Subsystem Block No.	Symbol	Units
	$U_{EB}$	Ft/Sec
	$V_{EB}$	Ft/Sec
	$W_{EB}$	Ft/Sec

10e. Subsystem for Ground Velocity Summation

INPUTS:

Subsystem Block No.	Symbol	Units
	$U_W$	Ft/Sec
	$\theta_W$	Rad
	$\gamma_W$	Rad
	$U_{EB}$	Ft/Sec
	$V_{EB}$	Ft/Sec
	$W_{EB}$	Ft/Sec

EQUATIONS:

$$U_{WE} = U_W (\cos \gamma_W \cos \theta_W)$$

$$V_{WE} = V_W (-\sin \gamma_W)$$


$$W_{WE} = U_W (\cos \gamma_W \sin \theta_W)$$

$$U_G = U_{EB} - U_{WE}$$

$$V_G = V_{EB} - V_{WE}$$

$$W_G = W_{EB} - W_{WE}$$

OUTPUT:

Subsystem Block No.	Symbol	Units
	$U_G$	Ft/Sec
	$V_G$	Ft/Sec
	$W_G$	Ft/Sec

10f. Subsystem for Ground Reference Distances

INPUTS:

Subsystem Block No.	Symbol	Units
(10e) ↓	$U_G$	Ft/Sec
	$V_G$	Ft/Sec
	$W_G$	Ft/Sec
(9)	$WL_{CG}$	Inch

EQUATIONS:

$$N_N = \frac{1}{1.6878} \int U_G dt, \quad P_{AX} = N_N + X_0$$

$$E = \frac{1}{1.6878} \int V_G dt, \quad P_{AY} = E + Y_0$$

$$h_0 = \frac{(WL_{CG} - 11.0)}{12} + h_I (h_I - \text{Initial CG altitude})$$

$$h = -\int W_G dt + h_0, \quad P_{AZ} = h + h_0$$

$$\dot{h} = -W_G$$

OUTPUTS:

Subsystem Block No.	Symbol	Units
	$N_N$	NM
	$E$	NM
(9) (15) (16) (18) ↓	$h$	Ft
	$P_{AX}$	NM
	$P_{AY}$	NM
	$P_{AZ}$	Ft
(16)	$\dot{h}$	Ft/Sec



**SUBSYSTEM NO. 11: AIRCRAFT ANGULAR ACCELERATIONS AND VELOCITIES**

**INPUTS:**

**A. Variables**

Subsystem Block No.

Symbol

Units

13

$X_A$

Lbs.

$Z_A$

Lbs

14

$l_A$

Lb Ft

$M_A$

Lb. Ft.

$N_A$

Lb. Ft.

11

$\dot{p}$

Rad/Sec<sup>2</sup>

$\dot{q}$

Rad/Sec<sup>2</sup>

$\dot{r}$

Rad/Sec<sup>2</sup>

$p$

Rad/Sec

$q$

Rad/Sec

$r$

Rad/Sec

8a

$\beta_M$

Rad

$\dot{\beta}_M$

Rad/Sec

9

$I_{XX}$

Slug Ft<sup>2</sup>

$I_{YY}$

Slug Ft<sup>2</sup>

$I_{ZZ}$

Slug Ft<sup>2</sup>

$I_{XZ}$

Slug Ft<sup>2</sup>

$X_{CG}, Z_{CG}$

Ft

$\dot{X}_{CG}, \dot{Z}_{CG}$

Ft/Sec

$\ddot{X}_{CG}, \ddot{Z}_{CG}$

Ft/Sec<sup>2</sup>

Subsystem Block No.

Symbol

Units

9  
↓

SL<sub>CG</sub>

Inch

WL<sub>CG</sub>

Inch

EQUATIONS:

A. Aircraft CG Angular Accelerations (Body Axes)

Roll Equation:

$$I_{XX} \dot{p} = (I_{YY} - I_{ZZ}) q r + I_{XZ} (\dot{r} + pq) + l_A$$

Pitch Equation:

$$I_{YY} \dot{q} = (I_{ZZ} - I_{XX}) pr + I_{YZ} (r^2 - p^2) + M_A$$

Yaw Equation:

$$I_{ZZ} \dot{r} = (I_{XX} - I_{YY}) pq + I_{XZ} (\dot{p} - rq) + N_A$$

Angular Rate Equations:

$$p = \int \dot{p} dt$$

$$q = \int \dot{q} dt$$

$$r = \int \dot{r} dt$$

B. Pilot Station Accelerations (Body Axes)

$$a_{XPA} = \frac{X_A}{m} + (\dot{q} + pr)(z_{PA} - z_{CG}) + (q^2 + r^2)(x_{CG} - l_{PA})$$

$$+ Y_{PA}(pq - \dot{r}) - 2q \dot{z}_{CG} - \ddot{x}_{CG}$$

$$a_{YPA} = \frac{Y_A}{m} + (\dot{p} - qr)(Z_{CG} - Z_{PA}) + (\dot{r} + pq)(l_{PA} - X_{CG}) - Y_{PA}(r^2 + p^2) + 2(p\dot{Z}_{CG} - r\dot{X}_{CG})$$

$$a_{ZPA} = \frac{Z_A}{m} + (\dot{q} - pr)(X_{CG} - l_{PA}) + (p^2 + q^2)(Z_{CG} - Z_{PA}) + Y_{PA}(\dot{p} + qr) + 2q\dot{X}_{CG} - \dot{Z}_{CG}$$

C. Pilot Station Velocities (Body Axes)

$$U_{PA} = U - q * Z'_{PA} - r * Y'_{PA} - \dot{X}_{CG}$$

$$V_{PA} = V + r * l'_{PA} + p * Z'_{PA}$$

$$W_{PA} = W - q * l'_{PA} + p * Y'_{PA} - \dot{Z}_{CG}$$

where,  $l'_{PA} = (SL_{CG} - SL_{PA})/12$ ,  $l_{PA} = (SL_{SP} - SL_{PA})/12$

$$Y'_{PA} = (BL_{PA} - BL_{CG})/12, Y_{PA} = \frac{BL_{PA}}{12}$$

$$Z'_{PA} = (WL_{PA} - WL_{CG})/12, Z_{PA} = (WL_{PA} - WL_{SP})/12$$

OUTPUTS

Subsystem Block No.	Symbol	Units
(11)	$\dot{p}$	Rad/Sec <sup>2</sup>
	$\dot{q}$	Rad/Sec <sup>2</sup>
	$\dot{r}$	Rad/Sec <sup>2</sup>
(1), (4), (5), (6), (10c), (11), (12)	$p$	Rad/Sec
FSAA CAB	$q$	Rad/Sec
	$r$	Rad/Sec
FSAA CAB	$a_{XPA}$	Ft/Sec <sup>2</sup>
	$a_{YPA}$	Ft/Sec <sup>2</sup>
	$a_{ZPA}$	Ft/Sec <sup>2</sup>
	$U_{PA}$	Ft/Sec
	$V_{PA}$	Ft/Sec
	$W_{PA}$	Ft/Sec

**SUBSYSTEM NO. 12: BODY AXIS LINEAR ACCELERATIONS AND VELOCITIES**

**INPUTS:**

**A. Variables**

Subsystem Block No.

Symbol

Units

13

$X_A$

Pounds

$X_A$

Pounds

$Z_A$

Pounds

12

U

Ft/Sec

V

Ft/Sec

W

Ft/Sec

11

p

Rad/Sec

q

Rad/Sec

r

Rad/Sec

10c

$\theta$

Rad

$\phi$

Rad

EQUATIONS:

$$\dot{U} = -g \sin \theta + Vr - Wq + \frac{X_A}{m}$$

$$\dot{V} = g \cos \theta \sin \phi - rU + Wp + \frac{Y_A}{m}$$

$$\dot{W} = g \cos \theta \cos \phi + qU - pV + \frac{Z_A}{m}$$

$$N_X = \frac{X_A}{GW}$$

$$N_Y = \frac{Y_A}{GW}$$

$$N_Z = \frac{Z_A}{GW}$$

$$U = \int \dot{U} dt$$

$$V = \int \dot{V} dt$$

$$W = \int \dot{W} dt$$

$$V_T = \sqrt{U^2 + V^2 + W^2}$$

Angle of Attack,

$$\alpha_F = \tan^{-1} \frac{W}{U}$$

Angle of sideslip

$$\beta_F = \tan^{-1} \frac{V}{\sqrt{U^2 + W^2}}$$

OUTPUTS:

Subsystem Block No.	Symbol	Units
(1) (4) (5) (6) (7) (10d)	U	Ft/Sec
	V	Ft/Sec
	W	Ft/Sec
(3) (7) (8b)	$V_T$	Ft/Sec
	$\alpha_F$	Rad
	$\beta_F$	Rad
(16)	$N_X$	g's
	$N_Y$	g's
	$N_Z$	g's



**SUBSYSTEM NO. 13-FORCE SUMMATION**

**INPUTS:**

Subsystem Block No.

10a



Symbol	Units
$(X, Y, Z)_F$	Lbs.
$(X, Z)_{iWPL}$	Lbs.
$(X, Z)_{iWPR}$	Lbs.
$(X, Y, Z)_{WP}$	Lbs.
$(X, Y, Z)_H$	Lbs.
$(X, Z)_{MG}$	Lbs.
$(X, Z)_{NG}$	Lbs.
$(X, Y, Z)_L$	Lbs.
$(X, Y, Z)_R$	Lbs.
$(X, Y)_{VL}$	Lbs.
$(X, Y)_{VR}$	Lbs.
$(\Delta X, \Delta Y, \Delta Z)_{LG}$	Lbs.

EQUATIONS:

$$X_A = X_F + X_{iWPR} + X_{WP} + X_H + X_{MG} + \Delta X_{LG} + X_{NG}$$

$$+ X_L + X_R + X_{VL} + X_{VR} + X_{iWPL}$$

$$Y_A = Y_F + Y_{WP} + \Delta Y_{LG} + Y_L + Y_R + Y_{VL} + Y_{VR} + Y_H$$

$$Z_A = Z_F + Z_{iWPL} + Z_{WP} + Z_H + Z_{MG} + \Delta Z_{LG}$$

$$+ Z_{NG} + Z_L + Z_R + Z_{iWPR}$$

OUTPUTS:

Subsystem Block No.

12



Symbol

X<sub>A</sub>

Y<sub>A</sub>

Z<sub>A</sub>

Units

Lbs.

Lbs.

Lbs.

**SUBSYSTEM NO. 14-MOMENT SUMMATION**

**INEUTS:**

Subsystem Block No.



Symbol	Units
$SL_{CG}$	Inch
$WL_{CG}$	Inch
$BL_{CG}$	Inch
$(X, Y, Z)_F$	Lb.
$(X, Z)_{iWPL}$	Lb.
$(X, Z)_{iWPR}$	Lb.
$(X, Y, Z)_{WP}$	Lb.
$(X, Y, Z)_H$	Lb.
$(X, Y)_{VL}$	Lb.
$(X, Y)_{VR}$	Lb.
$(X, Z)_{MG}$	Lb.
$(X, Z)_{NG}$	Lb.
$(X, Y, Z)_L$	Lb.
$(X, Y, Z)_R$	Lb.
$(1, M, N)_F$	Ft.lb
$(1, M, N)_{WP}$	Ft.lb
$M_H$	Ft.lb
$(1, M, N)_L$	Ft.lb
$(1, M, N)_R$	Ft.lb
$(\Delta l, \Delta M, \Delta N)_{LG}$	Inch
$(r_{XW}, r_{YW})_R$	Inch
$(r_{XW}, r_{YW})_L$	Inch
$\phi$	Deg.
$h_H$	Ft.

**EQUATIONS:**

$$\begin{aligned}
 M_A = & X_F (WL_{CG} - WL_F)/12 + Z_F (SL_F - SL_{CG})/12 + (X_{iWPL} + X_{iWPR} + X_{iJP}) * \\
 & (WL_{CG} - WL_W)/12 - Z_{iWPR} * r_{XWR}/12 - Z_{iWL} * r_{XWL}/12 + Z_{WP} (SL_W - SL_{CG})/12 \\
 & + X_H (WL_{CG} - WL_H)/12 + Z_H (SL_H - SL_{CG})/12 + X_{MG} (WL_{CG} - WL_{MG})/12 \\
 & + Z_{MG} (SL_{MG} - SL_{CG})/12 + X_{NG} (WL_{CG} - WL_{NG})/12 + Z_{NG} (SL_{NG} - SL_{CG})/12 \\
 & + (X_{VL} + X_{VR}) (WL_{CG} - WL_V)/12 + (X_L + X_R) [WL_{CG} - (WL_{SP} + 12 l_M \cos \beta_M)]/12 \\
 & + (Z_L + Z_R) [(SL_{SP} - 12 l_M \sin \beta_M) - SL_{CG}]/12 + M_F + M_{WP} \\
 & + M_H + M_L + M_R + \Delta M_{LG}
 \end{aligned}$$

$$\begin{aligned}
 l_A = & Y_F (WL_F - WL_{CG})/12 + Z_{iWPL} * r_{YWL} /12 + Z_{iWPR} * r_{YWYR} /12 \\
 & + (Y_{iWPR} + Y_{iWPL} + Y_{WP}) (WL_W - WL_{CG})/12 + (Y_{VL} + Y_{VR}) (WL_V - WL_{CG})/12 \\
 & + (Y_L + Y_R) [(WL_{SP} + 12 l_M \cos \beta_M) - WL_{CG}] /12 + Z_L (BL_{SPL} - BL_{CG})/12 \\
 & + Z_R (BL_{SPR} - BL_{CG})/12 + l_F + l_{WP} + l_L + l_R + l_G * \phi + \Delta l_{LG}
 \end{aligned}$$

Where,

$$l_G = \{ l_{G0} + l_{G1} * \frac{h_H}{2R} + l_{G2} * (\frac{h_H}{2R})^2 + l_{G3} * (\frac{h_H}{2R})^3 \} * e^{-.1V_T}$$

$$\text{For } 0.5 \leq \frac{h_H}{2R} \leq 1.4$$

$$\begin{aligned}
 N_A = & Y_F (SL_{CG} - SL_F)/12 - X_{iWPL} * r_{YWL} /12 - X_{iWPR} * r_{YWR}/12 \\
 & + Y_{iWPR} * r_{XWR}/12 + Y_{iWPL} * r_{XWL}/12 + Y_{WP} (SL_{CG} - SL_W)/12 \\
 & + X_L (BL_{CG} - BL_{SPL})/12 + X_R (BL_{CG} - BL_{SPR})/12 + Y_H (SL_{CG} - SL_H)/12 \\
 & + (Y_{VL} + Y_{VR}) (SL_{CG} - SL_V)/12 + (X_{VL} - X_{VR}) (BL_{VR} - BL_{CG})/12 \\
 & + (Y_L + Y_R)[SL_{CG} - (SL_{SP} - 12 l_M \sin \beta_M)]/12 + N_F + N_{WP} + N_L \\
 & + N_R + \Delta N_{LG}
 \end{aligned}$$

OUTPUTS:

Subsystem Block No.

11



Symbol

Units

$M_A$

Ft.Lb.

$L_A$

Ft.Lb.

$N_A$

Ft.Lb.

**SUBSYSTEM NO. 15: FLIGHT ENVIRONMENT DATA**

**INPUTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
(10f)	h	Ft.
(12)	$v_T$	Ft/Sec

**B. Constants**

$T_o$	$^{\circ}C$
-------	-------------



EQUATIONS:

$$T_a = 288.15 - .0019812 h + T_o$$

$$\theta_T = \frac{T_a}{288.15}$$

$$\delta_T = \left[ \frac{T_a}{288.15} \right]^{5.255876}$$

$$\sigma' = \left\{ \frac{(1 - .00000687 h)^{5.255876}}{1 - .000687 h + \theta_T} \right\}$$

$$\rho = (.002378)(\sigma')$$

$$V_S = 661.48 (\theta_T)^{1/2}$$

$$M_N = V_T/V_S$$

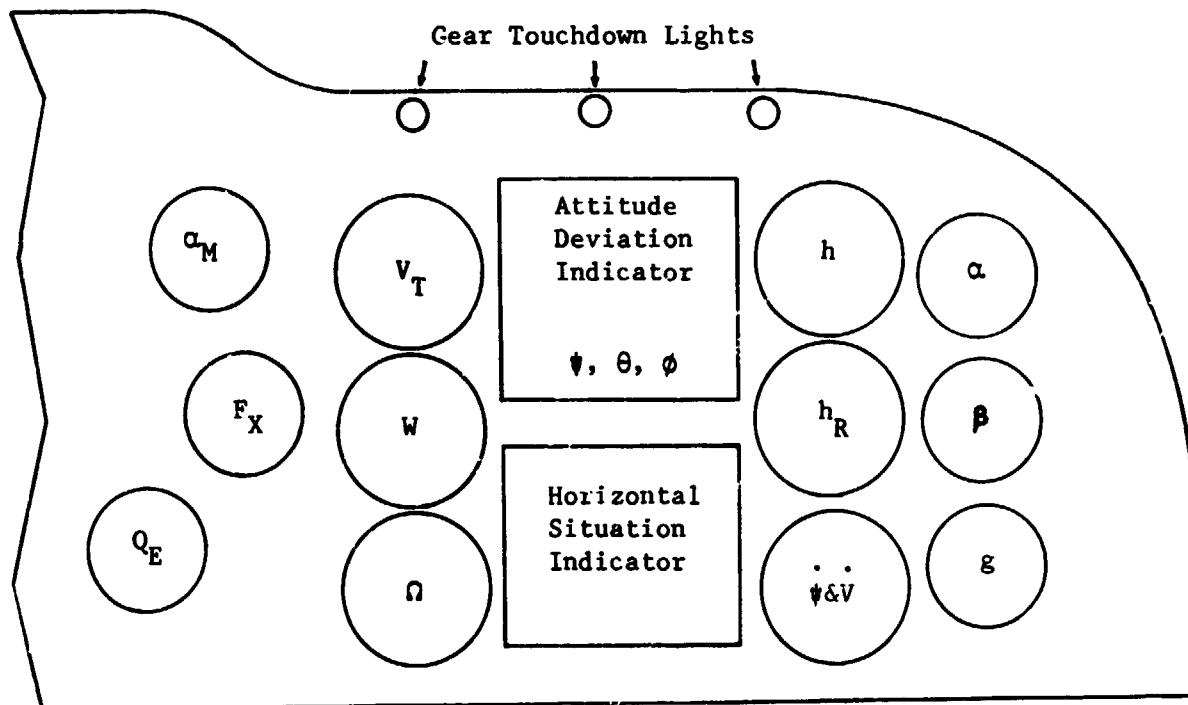
$$V_{CAS} = 661.48 \left[ 5 \left\{ \left( 1 + \delta_T \left( 1 + \frac{.2}{\theta_T} \left( \frac{V_T}{661.48} \right)^2 \right)^{7/2} - 1 \right)^{2/7} - 1 \right\} \right]^{1/2}$$

OUTPUTS:

Subsystem Block No.	Symbol	Units
(1) (3) (4) (5) (6) (7)	$\rho$	Slugs-Ft <sup>2</sup>
(16)	$V_{CAS}$	MPH
(18)	$T_a$	°K
(4) (5) (6)	$V_S$	Ft/Sec

SUBSYSTEM NO. 16: PILOT'S FLIGHT INSTRUMENTS

COCKPIT INSTRUMENT DISPLAY



$\alpha_M$  Nacelle Tilt Angle  
 $F_X$  Flap Position  
 $Q_E$  Engine Torque  
 $V_T$  Airspeed  
 $W$  Vertical Velocity Indicator  
 $\Omega$  Rotor RPM

$h$  Baro Altimeter  
 $h_R$  Radar Altimeter  
 $\psi \& V$  Turn and Slip  
 $\alpha$  Angle of Attack  
 $\beta$  Side Slip Angle  
 $g$  Normal Acceleration

FSAA - MODEL 301 PILOTS PANEL

**SUBSYSTEM NO. 17: ROTOR COLLECTIVE GOVERNOR**

**INPUTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
8d	RPM <sub>P</sub>	RPM
19	Ω <sub>INT</sub>	Rad/Sec

**B. Coefficients**

K <sub>RPG</sub>		ND
K <sub>1G</sub>		ND
$\frac{\delta\Omega_{0 INT}}{\delta\Omega_{RO}}$		ND

**EQUATIONS:**

$$\bar{\theta}_{OG} = K_{RPG} (\beta_M) * \left[ \frac{1}{s} + K_{1G} \right] \left[ \frac{\Omega_{INT} * 9.55}{\left( \frac{\delta\Omega_{INT}}{\delta\Omega_{RO}} \right)} - RPM_P \right] \quad \left( s \text{ is Laplace operator} \right)$$

$$\text{If } \bar{\theta}_{OG} > 0 \text{ then } \theta_{OL/G}, \theta_{OR/G} = \text{MIN} \left[ \bar{\theta}_{OG}, \theta_{OG_{max}} \right]$$

$$\text{If } \bar{\theta}_{OG} < 0 \text{ then } \theta_{OL/G}, \theta_{OR/G} = \text{MIN} \left[ \bar{\theta}_{OG}, \theta_{OG_{min}} \right]$$

**OUTPUTS:**

Subsystem Block No.	Symbol	Units
8a	θ <sub>OL/G</sub> , θ <sub>OR/G</sub>	Deg

**SUBSYSTEM NO. 18: ENGINES AND FUEL CONTROLS**

**INPUTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
8d	$X_{THR}$	Deg
10f	$X_{THL}$	Deg
12	$h$	Ft.
15	$V_T$	Ft/Sec
	$P_a$	Psia
	$T_a$	$^{\circ}K$
19	$\Omega_{RPT}$	Rad/Sec
	$\Omega_{LPT}$	Rad/Sec

**B. Engine Coefficients**

$K_1, K_2, K_3$	ND
$K_4, K_5, K_6$	RPM, RPM/HP, HP
$K_7$	$^{\circ}K$
$K_8, K_9, K_{10}, K_{11}, K_{13}, K_{14}$	HP/Deg <sup>2</sup> , HP/Deg, HP, 1/ $^{\circ}K$ , 1/Deg, 1/Deg <sup>2</sup> , Deg
$t_D$	Sec
pctmxs, pctmxp	%, %
$P_o$	Lb/In <sup>2</sup>
$T_o$	$^{\circ}K$

EQUATIONS:

A. Power-Turbine (N<sub>2</sub>) Governor

$$HP_{ROPTG} = (HP_{RO})_0 + \int_{t_D}^{t_{PTG}} \frac{dHP_{ROS}}{dt_{PTG}} dt_{PTG}$$

$$\epsilon_S = \left( \frac{9.55 * \Omega_{RPT}}{RPM_N} - 1 \right)$$

$$RPM_{N_{II}} = 22200$$

Integration begins when  $|\epsilon_S|$  has exceeded 0.002 for  $t_D$  seconds, and continues until  $\epsilon_S \leq 0.002$  at which time  $(HP_{RO})_0$  is reset to the current  $HP_{RO}$ , and  $t_{PTG}$  is reset to zero.

$$\frac{dHP_{ROS}}{dt_{PTG}} = \text{sign}(-\epsilon_S) * \text{MIN} \left\{ 1, \frac{100 * |\epsilon_S|}{\text{pctmxs}} \right\} * f_1(HP_{RO}, h)$$

$$HP_{ROPTG} = \text{MIN} \left\{ HP_{ROPTG}, HP_{ROTH} \right\}$$

B. Throttle Control

$HP_{ROC}$  is the commanded referred optimum HP on one engine.

$$\text{If } X_{THR} \leq 46.0 \text{ } HP_{ROC} = 150.$$

$$\text{If } T_a \leq K_7,$$

$$HP_{ROC} = K_8 X_{THR}^2 + K_9 X_{THR} + K_{10}$$

If  $T_a > K_7$ ,

$$HP_{ROC} = [1 - K_{11} (T_a - K_7)][K_{12} + K_{13} (X_{THR} - K_{14})][K_8 X_{THR}^2 + K_9 X_{THR} + K_{10}]$$

$$HP_{RC} = (HP_{RO})_0 + \int_{t_D}^{t_{HP}} \frac{dHP_{ROT}}{dt_{TH}} dt_{TH}$$

$$\epsilon_p = \frac{HP_{ROTH}}{HP_{ROC}} - 1$$

Integration begins when  $\epsilon_p$  has exceeded 0.002 for  $t_D$  seconds, and continues till  $\epsilon_p \leq 0.002$ , at which time  $(HP_{RO})_0$  is reset to the current  $HP_{RO}$ , and  $t_{TH}$  is reset to zero.

$$\frac{dHP_{ROT}}{dt_{TH}} = \text{SIGN}(-\epsilon_p) * \min \left\{ 1, \frac{100 * \epsilon_p}{\text{pctmxp}} \right\} * f_1 (HP_{RO}, h)$$

### C. Actual Power Developed

Referred optimum horsepower of one engine

$$= HP_{RO} = \min \{ HP_{ROPTG}, HP_{ROTH} \}$$

$$HP = (HP_{RO} \delta \sqrt{\theta}) \left[ K_1 \left( \frac{9.55 * \Omega \text{ RPT}}{\sqrt{\theta} \text{ RPM}_{RO}} \right)^2 + K_2 \left( \frac{9.55 * \Omega \text{ RPT}}{\sqrt{\theta} \text{ RPM}_{RO}} \right) + K_3 \right]$$

where  $\text{RPM}_{RO} = K_4 + K_5 \left\{ \text{MAX} \left[ (HP_{RO} - K_3), 0.1 \right] \right\}^{1/2}$

$$\rho = \left( \frac{P_a}{P_o} \right) \left[ 1 + \frac{.0000461 v_T^2}{T_a} \right]^{7/2}$$

$$\theta = \left( \frac{T_a}{T_o} \right) \left[ 1 + \frac{.0000461 v_T^2}{T_a} \right]$$

The actual power developed by the two engines is given by:

$$HP_R = HP * \eta_{ER}$$

$$HP_L = HP * \eta_{EL}$$

$\eta_{ER}$	$\eta_{EL}$	Comments
1	1	Both engines operating
1	0	Left engine cut
0	1	Right engine out

$$Q_{RPT} = \frac{HP_R * 550}{\Omega_{RPT}}$$

$$Q_{LPT} = \frac{HP_L * 550}{\Omega_{LPT}}$$

Initial values: (at the beginning of simulation)

$$(HP_{RO})_0 = HP_{ROC}$$



OUTPUTS:

Subsystem Block No.

Symbol

Units

19  
↓

$Q_{RPT}$

Lb. Ft.



$Q_{LPT}$

Lb. Ft.

**SUBSYSTEM NO. 19: DRIVE SYSTEM DYNAMICS**

**INPUTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
	$Q_R$	Lb-Ft
	$Q_L$	Lb-Ft
	$Q_{RPT}$	Lb-Ft
	$Q_{LPT}$	Lb-Ft

**B. Constants**

Symbol	Units
$I_1$	Slug-Ft <sup>2</sup>
$\theta_{RPT1}$	ND
$\theta_{INT1}$	ND

EQUATIONS:

Drive shaft angular acceleration,

$$\ddot{\omega}_1 = \frac{F_1(t)}{I_1}$$

$$F_1(t) = -(Q_R + Q_L) + \theta_{RPT1} (Q_{RPT} + Q_{LPT})$$

$$\Omega_R = \Omega_{R0} + \dot{\xi}$$

$$\Omega_L = \Omega_R$$

$$\Omega_{RPT} = \Omega_R * \theta_{RPT1}$$

$$\Omega_{LPT} = \Omega_{RPT}$$

$$\Omega_{INT} = \Omega_R * \theta_{INT1}$$

**OUTPUTS:**

Subsystem Block No.	Symbol	Units
1	$\Omega_R$	Rad/Sec
↓	$\Omega_L$	Rad/Sec
18	$\Omega_{RPT}$	Rad/Sec
↓	$\Omega_{LPT}$	Rad/Sec
17	$\Omega_{INT}$	Rad/Sec

**SUBSYSTEM NO. 20: STABILITY AND CONTROL AUGMENTATION SYSTEM**

**INERTS:**

**A. Variables**

Subsystem Block No.	Symbol	Units
12	$V_T$	Ft/Sec
↓	$U$	Ft/Sec
8a	$\beta_M$	Deg
8d	$X_{LN}$	In.
↓	$X_{LT}$	In.
↓	$X_{PD}$	In.
10c	$P$	Rad/Sec
↓	$q$	Rad/Sec
↓	$r$	Rad/Sec
↓	$\psi$	Rad
↓	$\theta$	Rad
↓	$\phi$	Rad

**B. SCAS Gains and Time Constants**

$K_{1P}, \dots, K_{10P}$	$[U, \beta_M]$	(Pitch Gains)
$T_{1P}, T_{2P}, T_{3P}, T_{4P}$		(Pitch Time Constants)
$K_{1Y}, \dots, K_{5Y}$	$[U, \beta_M]$	(Yaw Gains)
$T_{1Y}, T_{2Y}, T_{3Y}, T_{4Y}$		(Yaw Time Constants)

$K_{1R} [U, \beta_M], \dots, K_{10R} [U, \beta_M]$

(Roll Gains)

$T_{1R}, T_{2R}, T_{3R}$

(Roll Time Constants)

EQUATIONS:

A. Pitch SCAS Electronics

$$\begin{aligned}
 PSCAS = & q * \left[ K_{3P} + \frac{s}{(sT_{2P})(sT_{3P} + 1)} K_{2P} \right] \\
 & + (X_{LN} - X_{LNN}) * \left[ \frac{s K_{1P} * K_{2P}}{(sT_{1P} + 1)(sT_{2P} + 1)(sT_{3P} + 1)} \right] \\
 & + PHOLD \\
 & (s \text{ is Laplace operator})
 \end{aligned}$$

where,

$$PHOLD = SIGN(\overline{PHOLD}) * MIN \left[ |\overline{PHOLD}|, PHOLD_{max} \right]$$

$$\begin{aligned}
 \overline{PHOLD} = & K_{5P} * \{ 57.3 * \theta(t) - \theta_{REFD}(t) \} \\
 & + K_{8P} * \{ -U(t) + U_{REFD}(t) \} + K_{9P} * 57.3 * \theta(t)
 \end{aligned}$$

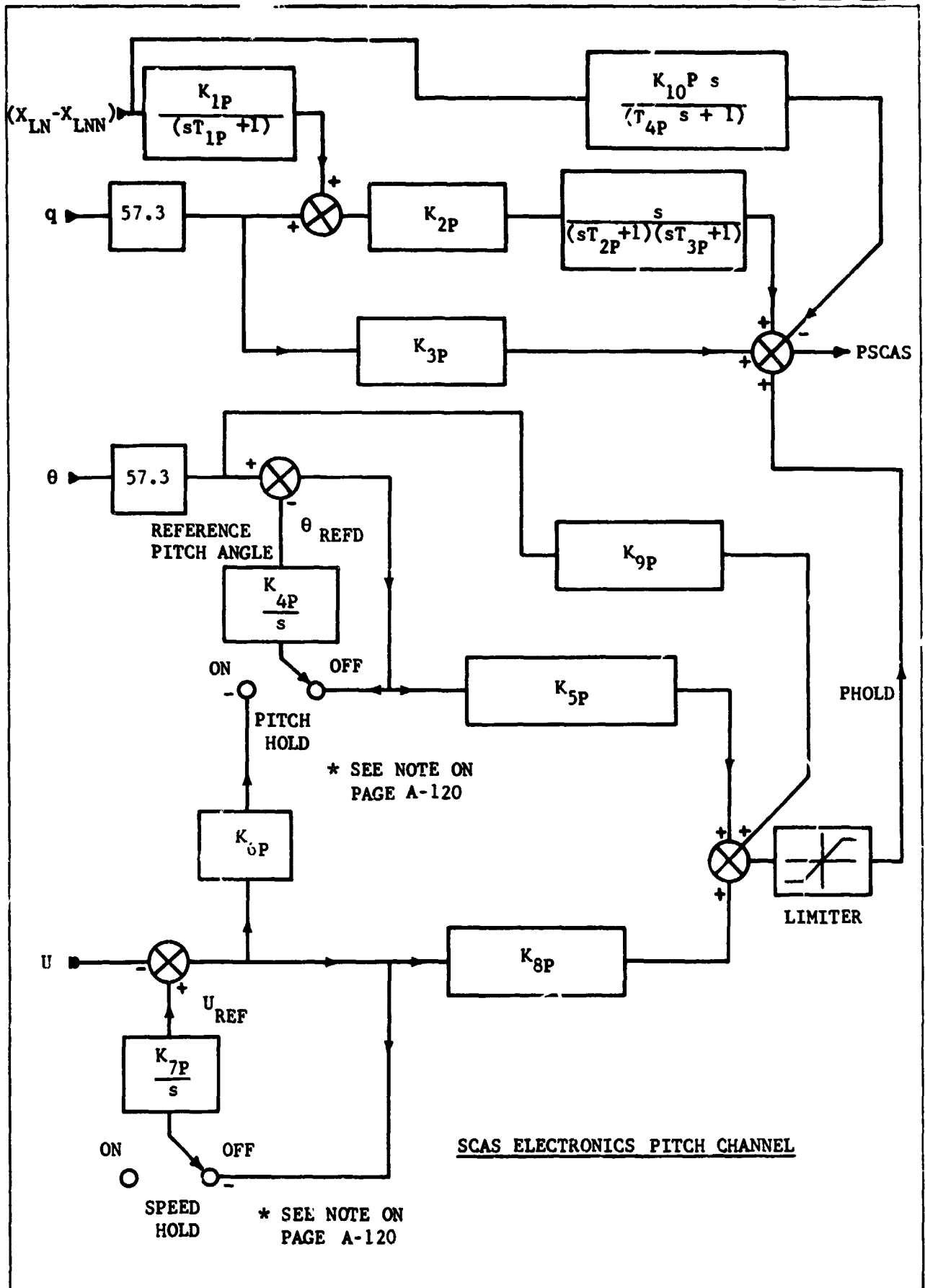
$$\begin{aligned}
 \theta_{REFD}(t) = & \int_0^t K_{4P} \{ K_{6P} * IFPH * [U(t) - U_{REF}(t)] \\
 & + [1 - IFPH(t)] * [57.3 * \theta(t) - \theta_{REFD}(t)] \} dt
 \end{aligned}$$

$$\theta_{REFD}(0) = 57.3 * \theta(0)$$

$$U_{REF}(t) = \int_0^t K_{7P} * [1 - IFUH(t)] * [U(t) - U_{REF}(t)] dt$$

$$U_{REF}(0) = U(0)$$

PHOLD is the output from the pitch-attitude-hold and airspeed-hold circuits.  
IFUH and IFPH defined on page A-115.





IFPH = 0, If pitch-attitude-hold is Off.

= 1, If pitch-attitude-hold is Operative

IFUH = 0, If airspeed-hold is Off.

= 1, If airspeed-hold is Operative and if  $U \geq 102$  Ft/Sec

When the airspeed-hold is tripped off due to U dropping below 102 FPS, it can be re-engaged only manually after the aircraft has reached a new trim-point.

Pitch SCAS Parameters

$$K_{1P} = K_{1PU} * K_{1PB_M}$$

$$K_{2P} = K_{2PU} * K_{2PB_M}$$

$$K_{3P} = K_{3PU} * K_{3PB_M}$$

If IFUH = 0,  $K_{5P} = K_{5PU0} * K_{5PB_M0}$

If IFUH = 1,  $K_{5P} = K_{5PU1} * K_{5PB_M1}$

$$K_{8P} = K_{8PU} * K_{8PB_M}$$

$$K_{9P} = K_{9PU} * K_{9PB_M}$$

$$K_{10P} = K_{10PB_M}$$

**B. Yaw SCAS Electronics**

$$\begin{aligned}
 Y_{SCAS} = & -r * \left[ K_{3Y} + \frac{s K_{2Y}}{(sT_{2Y} + 1)(sT_{3Y} + 1)} \right] \\
 & + (X_{PD} - X_{PDN}) * \left[ \frac{s (sT_{4Y} + 1) K_{1Y}}{(sT_{1Y} + 1)(sT_{2Y} + 1)(sT_{3Y} + 1)} \right] \\
 & + Y_{HOLD}
 \end{aligned}$$

where,

$$Y_{HOLD} = \text{SIGN}(\overline{Y_{HOLD}}) * \text{MIN} \left[ |\overline{Y_{HOLD}}|, Y_{HOLD_{max}} \right]$$

$$\overline{Y_{HOLD}} = K_{5Y} * \{ \Psi_{REFD}(t) - 57.3 * \Psi(t) \}$$

$$\Psi_{REFD}(t) = \int_0^t K_{4Y} * [1 - IFYH(t)] * [57.3 * \Psi(t) - \Psi_{REFD}(t)] dt$$

$$\Psi_{REFD}(0) = 57.3 * \Psi(0)$$

$Y_{HOLD}$  is the output from the yaw-attitude-hold circuit.

$IFYH = 0$ , If yaw-attitude-hold is Off.

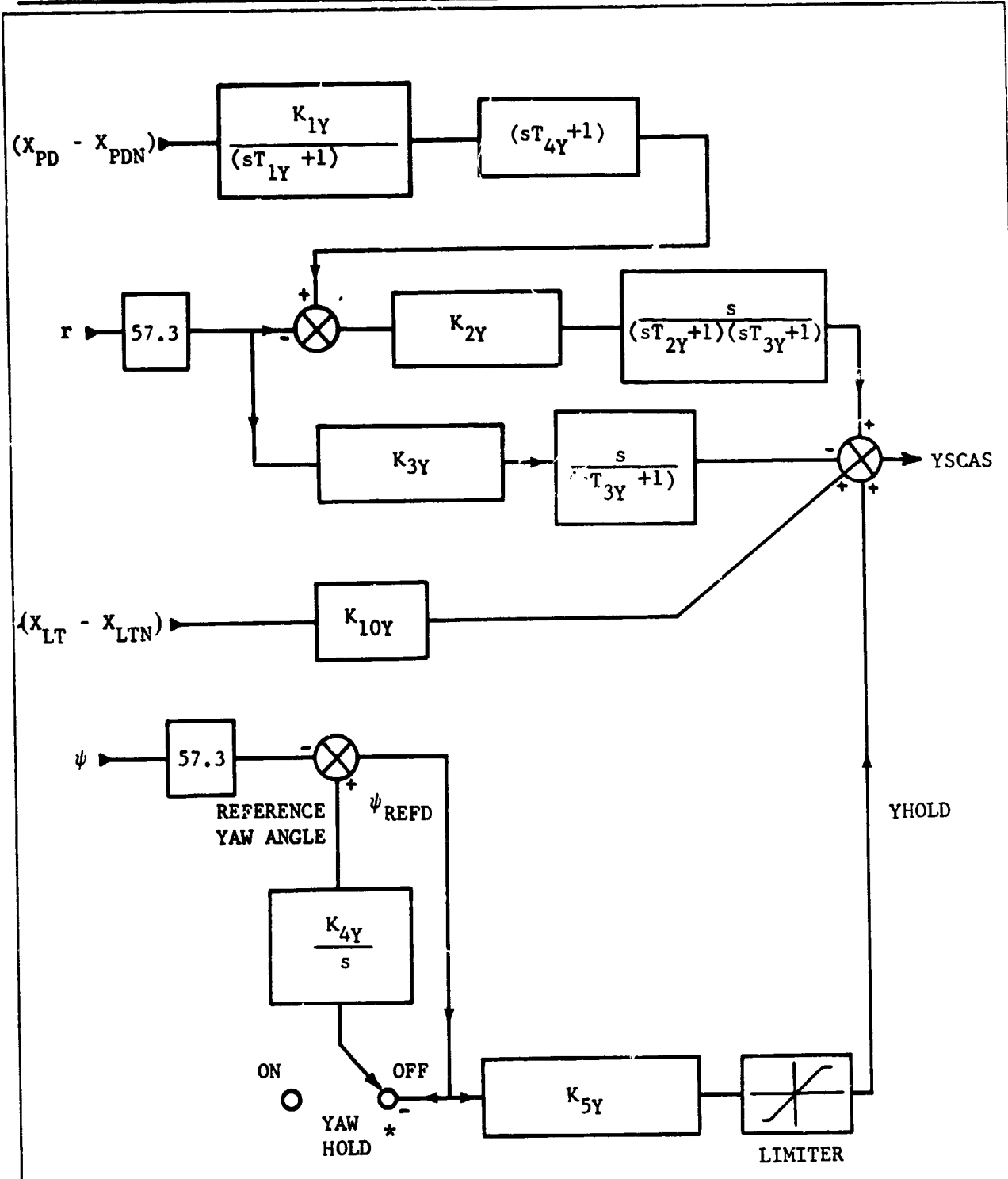
$= 1$ , If yaw-attitude-hold is Operative.

**Yaw SCAS Parameters**

$$K_{1Y} = K_{1YU} * K_{1Y\beta_M}$$

$$K_{2Y} = K_{2YU} * K_{2Y\beta_M}$$

$$K_{3Y} = K_{3YU} * K_{3Y\beta_M}$$



**SCAS ELECTRONICS**  
**YAW CHANNEL**

\*See Note on page A-120

$$\text{If IFUH} = 0, K_{5Y} = K_{5YU0} * K_{5Y\beta_M}^0$$

$$\text{If IFUH} = 1, K_{5Y} = K_{5YU1} * K_{5Y\beta_M}^1$$

$$K_{10Y} = K_{10Y\beta_M}$$

Roll SCAS Electronics

$$\begin{aligned} \text{RSCAS} = & -p * \left[ K_{3R} + \frac{s K_{2R}}{(sT_{2R} + 1)(sT_{3R} + 1)} \right] \\ & + \left( X_{LTN} - X_{LTN} \right) * \left[ \frac{s K_{1R} * K_{2R}}{(sT_{1R} + 1)(sT_{2R} + 1)(sT_{3R} + 1)} \right] \\ & + \text{RHOLD} \end{aligned}$$

where,

$$\text{RHOLD} = \text{SIGN}(\overline{\text{RHOLD}}) * \text{MIN} \left[ |\overline{\text{RHOLD}}|, \text{RHOLD}_{\text{max}} \right]$$

$$\overline{\text{RHOLD}} = K_{5R} * \{ \phi_{\text{REFD}}(t) - 57.3 * \phi(t) \} - K_{9R} * 57.3 * \phi(t)$$

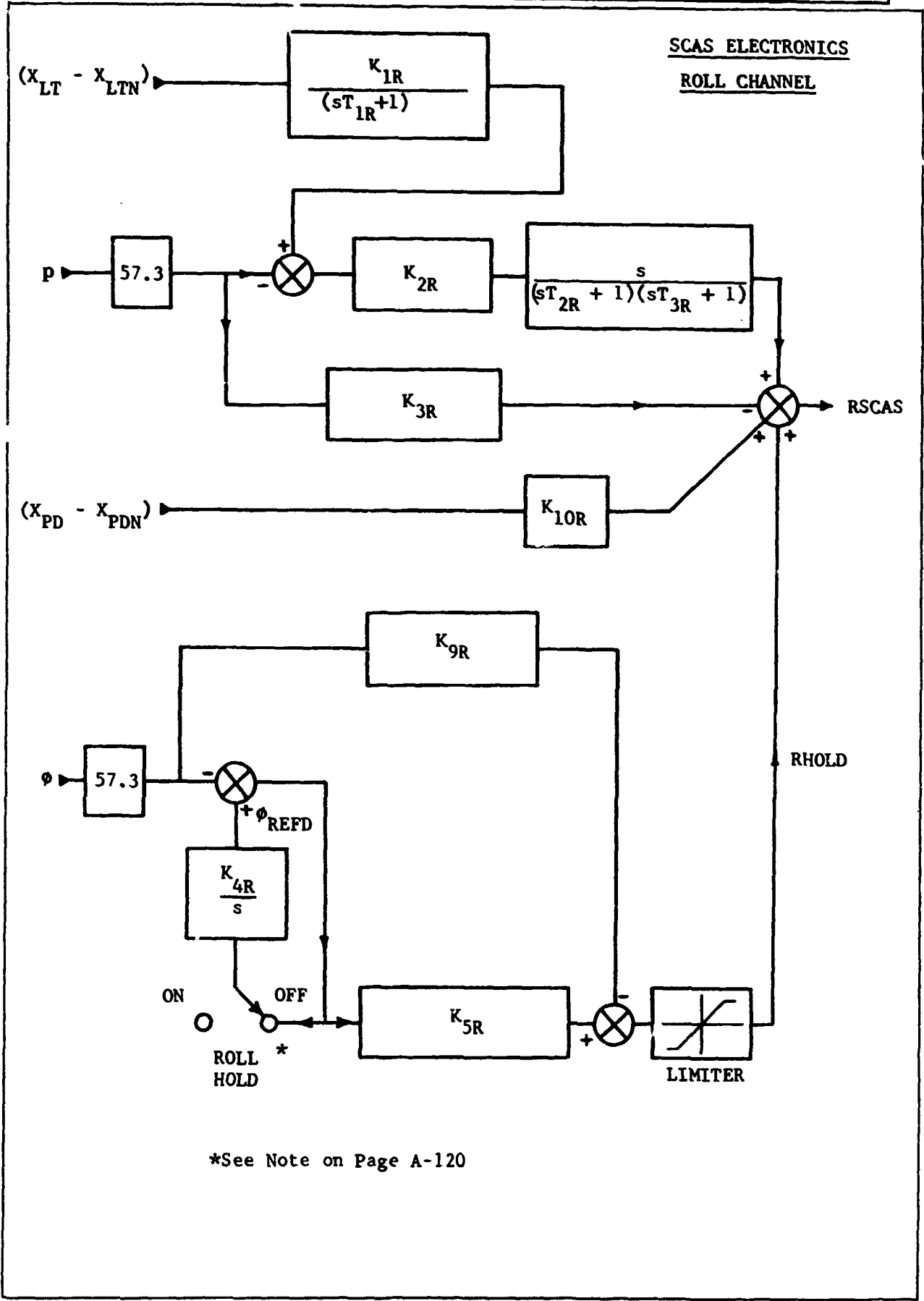
$$\phi_{\text{REFD}}(t) = \int_0^t K_{,R} * [1 - \text{IFRH}(t)] * [57.3 * \phi(t) - \phi_{\text{REFD}}(t)] dt$$

$$\phi_{\text{REFD}}(0) = 57.3 * \phi(0)$$

RHOLD is the output from the roll-attitude-hold circuit.

IFRH = 0, If roll-attitude-hold is Off.

= 1, If roll-attitude-hold is Operative.



Roll SCAS Parameters

$$K_{1R} = K_{1RU} * K_{1R\theta_M}$$

$$K_{2R} = K_{2RU} * K_{2R\theta_M}$$

$$K_{3R} = K_{3RU} * K_{3R\theta_M}$$

$$K_{9R} = K_{9RU} * K_{9R\theta_M}$$

If IFUH = 0,  $K_{5R} = K_{5RUC} * K_{5R\theta_M 0}$

If IFUH = 1,  $K_{5R} = K_{5RU1} * K_{5R\theta_M 1}$

$$K_{10R} = K_{10R}$$

$$ESAS = \text{SIGN}(\overline{PSCAS}) * \text{MIN } ESAS_{\text{max}}, \text{ABS}(\overline{PSCAS})$$

$$RSAS = \text{SIGN}(\overline{RSCAS}) * \text{MIN } RSAS_{\text{max}}, \text{ABS}(\overline{RSCAS})$$

$$ASAS = \text{SIGN}(\overline{ASCAS}) * \text{MIN } ASAS_{\text{max}}, \text{ABS}(\overline{ASCAS})$$

\*NOTE

ATTITUDE/AIRSPEED HOLD SWITCH LOGIC

1. Turn all holds to Off if Mag brake release depressed
2. Turn pitch and airspeed holds Off if long stick is out of force detent.
3. Turn roll and yaw holds Off if Lat stick or pedal is out of force detent.

OUTPUTS:

Subsystem Block No.

8a



Symbol

ESAS

RSAS

ASAS

Units

Inch

Inch

Inch

APPENDIX B

INPUT DATA FOR MODEL 301  
TILT ROTOR RESEARCH AIRCRAFT



**TABLE I - MODEL 301 CG AND INERTIA DATA**

DESIGN GW = 13000 lbs

	Aft CG	Fwd CG
$SL_{CG}   \beta_m = 0$	301.2	291.7
$BL_{CG}   \beta_m = 0$	0	0
$WL_{CG}   \beta_m = 0$	81.65	81.65
$I_{XX}   \beta_m = 0$	42379	42379
$I_{YY}   \beta_m = 0$	14230	14351
$I_{ZZ}   \beta_m = 0$	49459	49580

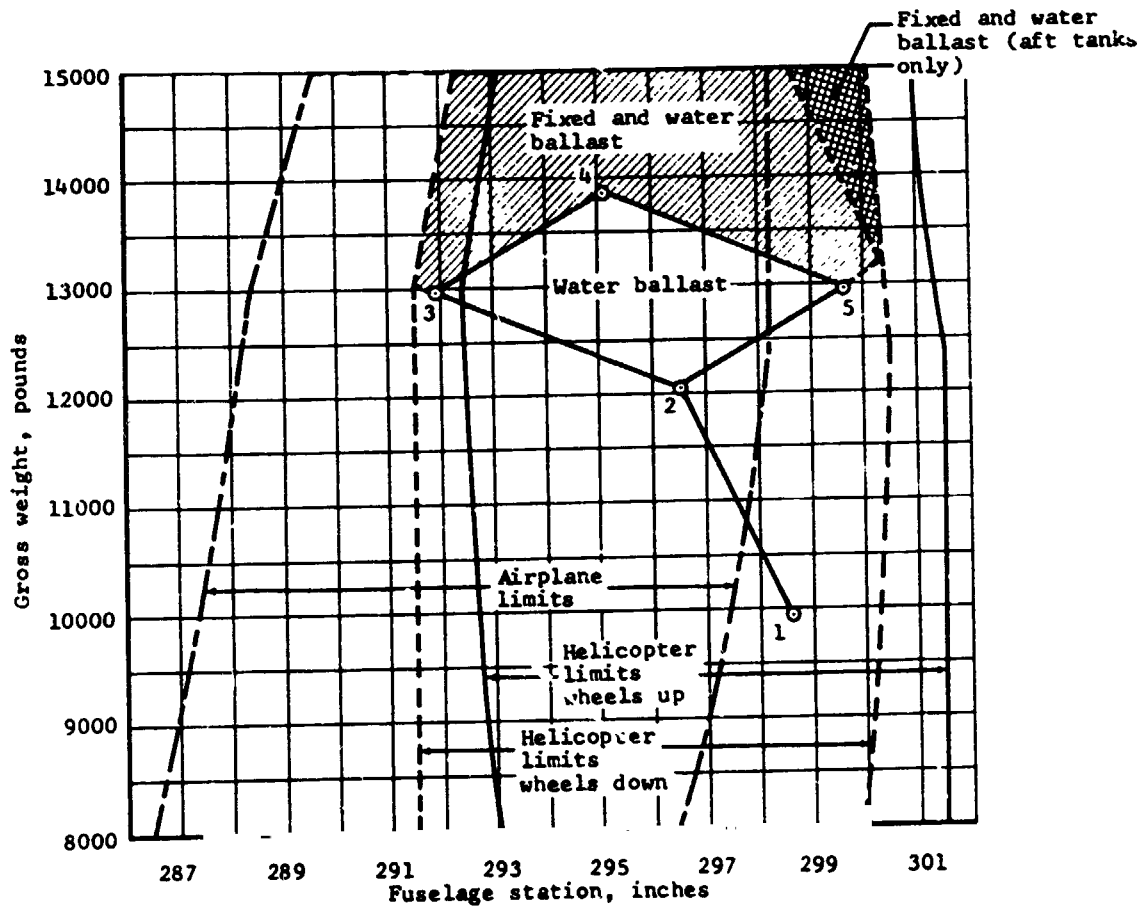


TABLE II-AIRCRAFT DESIGN DATA

ITEM	SYMBOL	M301 VALUE
<u>Fuselage</u>		
Center of pressure	$SL_F$	293.0 In.
	$BL_F$	0.0 In.
	$WL_F$	84.0 In.
<u>Wing-Pylon</u>		
Center of pressure	$SL_{WP}$	291.17 In.
	$BL_{WP}$	102.5 In.
	$WL_{WP}$	95.85 In.
Area	$S_W$	181.0 Ft <sup>2</sup>
Span	$b_W$	32.17 Ft.
Chord	$c_W$	5.225 Ft.
Sweep	$\Lambda c/4W$	-6.5 Deg.
Aspect Ratio	$AR_W$	5.7
Trailing Edge	$SL_{WTE}$	
<u>Horizontal Stabilizer</u>		
Center of pressure	$SL_H$	560.0 In.
	$BL_H$	0.0 In.
	$WL_H$	103.0 In.
Area	$S_H$	50.25 Ft. <sup>2</sup>
Span	$b_H$	12.83 Ft.
Chord	$c_H$	3.92 Ft.
Leading Edge	$SL_{HLE}$	548.25 In.
<u>Vertical Stabilizer</u>		
Center of pressure	$SL_V$	570.02 In.
	$BL_V$	77.0 In.
	$WL_V$	115.69 In.
Number of Panels		2.
Area (Per Panel)	$S_V$	25.25 Ft. <sup>2</sup>

ITEM	SYMBOL	M301 VALUE
<u>Vertical Stabilizer (Continued)</u>		
Span	$b_V$	7.68 Ft.
Chord	$c_V$	3.725 Ft.
Leading Edge	$SL_{VLE}$	555.1 In.
<u>Rotors</u>		
Location of Shaft Pivot Point	$SL_{SP}$	300.0 In.
	$BL_{SP}$	193.0 In.
	$WL_{SP}$	100.0 In.
Number of Blades Per Rotor	$n_b$	3
Radius	$R$	12.5 Ft.
Chord	$c_b$	1.167 Ft.
Mast Length	$l_M$	4.667 Ft.
Pitch-Flap Coupling	$\delta_3$	-15.0 Deg.
Solidity	$\sigma$	0.089
Lock Number	$\gamma$	3.83
Direction of Rotation		
-inboard tip motion-helicopter/airplane		Aft/Up
Rotor RPM		
Helicopter		565 RPM
Conversion		565 RPM
Airplane		458 RPM
Blade Flapping Limits		$\pm 12$ Deg.
Flapping Inertia per Blade	$I_b$	102.5 Slug-Ft. <sup>2</sup>
Flapping Spring Rate/Rotor	$K_{FA}/K_{LAT}$	225.0 Ft-L./Deg.
Blade Twist Distribution:	$x_M, \theta_M$	
	$x_{M0}, \theta_{M0}$	1.0, 0.0
	$x_{M1}, \theta_{M1}$	0.6, 10.2
	$x_{M2}, \theta_{M2}$	0.5333, 12.3
	$x_{M3}, \theta_{M3}$	0.4667, 14.5
	$x_{M4}, \theta_{M4}$	0.4000, 17.75

ITEM	SYMBOL	M301 VALUE	
<u>Rotors (Continued)</u>			
	$X_{M5}, \theta_{M5}$	0.3333, 21.90	
	$X_{M6}, \theta_{M6}$	0.2667, 26.15	
	$X_{M7}, \theta_{M7}$	0.2000, 30.65	
	$X_{M8}, \theta_{M8}$	0.1333, 34.65	
	$X_{M9}, \theta_{M9}$	0.0667, 38.00	
	$X_{M10}, \theta_{M10}$	0.0 , 40.90	
<u>Angle of outboard tilt of mast axis</u>			
Helicopter	$\phi_M$	1.0* Deg.	
Airplane		0 Deg.	
Conversion range	$\beta_M$	-5 to + 90 Deg.	
<u>Pylon</u>			
Center of Gravity	$\left\{ \begin{array}{l} SL_P \\ BL_P \\ WL_P \end{array} \right.$	291.7 In. 193.0 In. 118.0 In.	
	Weight (Two pylons)	$W_P$ 3986 Lbs.	
	Inertia (Per pylon)	$I_X$	81.4 Slug-Ft. <sup>2</sup>
$I_Y$		431.0 Slug-Ft. <sup>2</sup>	
$I_Z$		380.0 Slug-Ft. <sup>2</sup>	
<u>Landing Gear</u>			
Main Gear Coordinates	$\left\{ \begin{array}{l} SL_{MG} \\ BL_{MG} \\ WL_{MG} \end{array} \right.$	324.0 In. 54.75 In. 7.40 In.	
	Nose Gear Coordinates	$\left\{ \begin{array}{l} SL_{NG} \\ BL_{NG} \\ WL_{NG} \end{array} \right.$	139.0 In. 0.0 In. 4.95 In.

\*The builtin dihedral of the pylon is 2.5 degrees; in hover elastic deformation reduces the dihedral to 1.0 degree. 0° was erroneously used during Phase I simulation.

7872 55426

Item	Symbol	M301 Value
<u>Pilot Control Limits</u>		
Collective Stick	X <sub>COL</sub>	12.0 In.
Longitudinal Stick	X <sub>LN</sub>	±4.8 In.
Lateral Stick	X <sub>LT</sub>	±4.8 In.
Pedal	X <sub>PD</sub>	±2.5 In.
Blade Pitch Governor Lever		7.5 In.
<u>SCAS Actuator Limits</u>		
Longitudinal	ESAS <sub>max</sub>	1.54 In.
Lateral	ASAS <sub>max</sub>	1.54 In.
Pedal	RSAS <sub>max</sub>	0.8 In.
Pitch Attitude Hold	PHOLD <sub>max</sub>	0.77 In.
Yaw Attitude Hold	YHOLD <sub>max</sub>	0.4 In.
Roll Attitude Hold	RHOLD <sub>max</sub>	0.77 In.
<u>Engine Ratings</u>		
2 Min. Contingency		1760 HP
10 Min. Takeoff		1550 HP
30 Min. Military		1400 HP
Normal Rated		1250 HP
<u>Pilot Station Coordinates</u>		
	SL <sub>PA</sub>	215.25 In.
Pilot Station	BL <sub>PA</sub>	17.00 In.
	WL <sub>PA</sub>	50.5 In.

**SUBSYSTEM NO. 1: MODEL 301 ROTOR AERODYNAMIC DATA**

Equation	Constant	M301 Value
$a_{R,L}$	$a_0$	5.88
	$a_1$	9.20
	$a_2$	20.0
$C_{d_{FL,R}}$	$\delta_0$	0.002
	$\delta_1$	-.01
	$\delta_2$	0.50
$Q_{3R,L}$	B	0.97

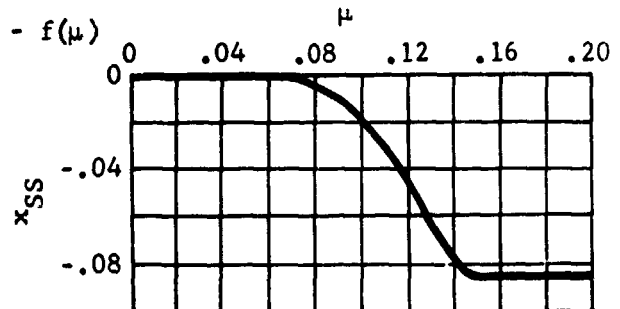
$\bar{T}_{R,L}$

$C_T - f(\mu)$

$\mu$	$C_T$
0	.0146
.057	.0146
.114	.0144
.171	.0138
.228	.0132
.285	.0127
.342	.0125
.399	.0123
.456	.0122

$Q_{R,L}$

$x_{SS} - f(\mu)$



Equation	Constant	M301 Value
Blade Twist Constants $K_{ijm}$	$X_{mi}, \theta_{mi}$	See Table II
TD3	TAN $\delta_3$	0.268
Y'	$I_b$	102.7 Slug-Ft <sup>2</sup>
$C_{KFAR}$	$K_{FA}$	225 Ft-Lb/Deg
$C_{KLTR}$	$K_{LT}$	225 Ft-Lb/Deg

**SUBSYSTEM NO. 2: MODEL 301 AERODYNAMIC DATA**

Equation	Constant	M301 Value
$W_i   R/W$	$K_0$	1.6
	$K_1$	0
	$K_2$	↓
	$K_3$	
	$K_4$	
$W_i   R/H$	$h_0$	- .5838
	$h_1$	.01158
	$h_2$	.5967
	$h_3$	.002547
	$h_4$	-2.2519



**SUBSYSTEM NO. 3: MODEL 301 FUSELAGE AERODYNAMIC DATA**

Equation	Constant	Model 301 Value
$L_F$	$L_0$	7.23
	$L_1$	0.905
$D_F$	$D_0$	1.56
	$D_1$	0.
	$D_2$	0.036 For Flaps Up Otherwise = 0
	$D_3$	0.0023
	$D_4$	125.
$M'_F$	$M_0$	-146.6
	$M_1$	17.8 ( $\alpha_F < 8^\circ$ )
	$M_2$	9.5
	$M_\alpha - f(\alpha_F)$	(Table F-1)
$Y'_F$	$Y_0$	0.
	$Y_1$	-1.44
	$Y_2$	0.
$l'_F$	$l_0$	0.
	$l_1$	-7.5
$N'_F$	$N_0$	0.
	$N_1$	-23.5

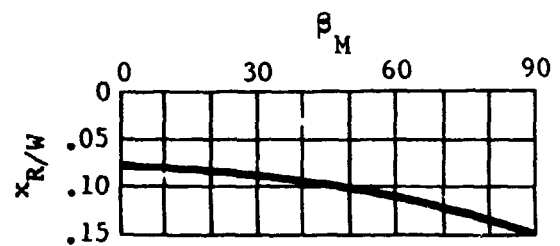
TABLE FI FUSELAGE PITCHING MOMENT COEFFICIENT ( $M_\alpha$ )

$\alpha$ F	$M_\alpha$
-40	-282
-36	-282
-32	-282
-28	-282
-24	-282
-20	-282
-16	-282
-12	-320
-8	-287
-4	-217
0	-146
4	-75
8	-11.3
12	-56.5
16	-94.1
20	-112.9
24	-112.9
28	-103.5
32	-75.3
36	-75.3
40	-94.1

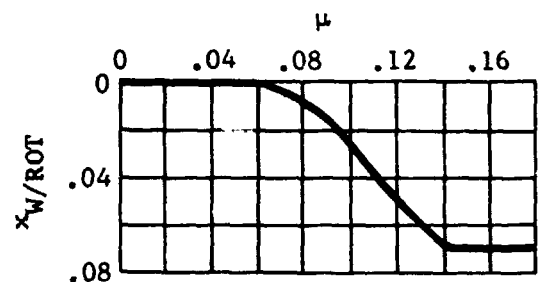
**SUBSYSTEM NO. 4: MODEL 301 WING-PYLON AERODYNAMIC DATA**

Equation	Constant	M301 Value								
$L_{iWPL}$	$C_{L_{WP}} = f(\alpha'_{F W} / \alpha_W \beta_M, F_x, M_N)$	Tables WI, WII								
$L_{iWPL}$										
$L_{WP}$										
$L_{WP}$	$C_{L_{\delta a}} = f(F_x)$	Table A III								
$D_{iWPL}$	$C_{D_{WP}} = f(\alpha'_{F W} / \alpha_W \beta_M, F_x, M_N)$	Tables W III, W IV								
$D_{iWPR}$										
$D_{WP}$										
$M_{OWP}$	$C_{M_{OWP}} = f(F_x)$	<table border="0"> <tr> <td>+0.051</td> <td><math>F_1(0/0)</math></td> </tr> <tr> <td>-0.057</td> <td><math>F_2(40/25)</math></td> </tr> <tr> <td>-0.20</td> <td><math>F_3(75/47)</math></td> </tr> <tr> <td>+0.085</td> <td><math>F_4(-28/-17.5)</math></td> </tr> </table>	+0.051	$F_1(0/0)$	-0.057	$F_2(40/25)$	-0.20	$F_3(75/47)$	+0.085	$F_4(-28/-17.5)$
+0.051	$F_1(0/0)$									
-0.057	$F_2(40/25)$									
-0.20	$F_3(75/47)$									
+0.085	$F_4(-28/-17.5)$									

$\Delta\alpha_{R/W} \quad x_{R/W} = f(\beta_M)$



$\Delta\alpha_{W/ROT} \quad x_{W/ROT} = f(\mu)$



Equation	Constant	M301 Value
$\epsilon_{W/H}$	$\epsilon_{W/H} = f(\alpha'_F   W / \alpha_W \beta_M, F_X)$	Table W V
$Y'_{WP}$	$C_{Y\beta}   M_N = 0$	0
	$\frac{C_{Yp}}{C_L}   M_N = 0$	0
	$C_{Yr}   M_N = 0$	0
$l'_{WP}$	$C_{l\beta}   C_L = 0, M_N = 0 = f(F_X, \beta_F, \beta_M)$	Table W VI
	$\frac{C_{l\beta}}{C_{lWP}}   M_N = 0 = f(F_X, \beta_F, \beta_M)$	Table W VII
	$C_{lp}   C_L = 0, M_N = 0$	- .774
	$\frac{C_{lr}}{C_{lWP}}   M_N = 0$	0.27
	$\left[ \frac{\Delta C_{lY}}{\left( \frac{\partial \alpha}{\partial \delta_F} \right) \delta_F} \right]$	-.0016
	$\partial \alpha / \partial \delta_F$	0.45

Equation	Constant	M301 Value
$N'_{WP}$	$C_{n\beta} \left  \begin{array}{l} C_L = 0 \\ M_N = 0 \end{array} \right.$	-0.015
	$\left( \frac{C_{n\beta}}{2 C_{L_{WP}}} \right) \left  M_N = 0 \right.$	0.
	$\left( \frac{C_{nr}}{2 C_{L_{WP}}} \right)$	-0.016
	$\left( \frac{C_{nr}}{C_{D_{0_{WP}}}} \right)$	-0.32
$l'_{WP}$	$K_{l\delta_a} = f(\alpha_W, \beta_M, F_x)$	Table A II
	$C_{l\delta_a} \left  \begin{array}{l} F_1 \\ \alpha_W < 3^\circ \end{array} \right.$	0.005
$N'_{WP}$	$\frac{C_{np}}{C_{L_{WP}}} \left  M_N = 0 \right.$	-0.06
	$K_{np}$	1.0
	$K_{n0\delta_a} = f(F_x, \beta_M)$	Table A IV
	$K_{n\delta_a} = f(F_x, \beta_M)$	Table A IV

TABLE WI WING-PYLON LIFT COEFFICIENT ( $C_{LWP}$ )

FLAP SETTING ( $F_Y$ )		$F_1$ (0/0)				
MAST ANGLE ( $\beta_M$ )		AIRPLANE ( $90^\circ$ )			HELICOPTER ( $0^\circ$ )	
MACH NUMBER ( $M_N$ )		0-.2	.4	.5	.6	0-.4
	$\alpha'_{F/W}/\alpha'_W$					
-40	-40	-.93				-.68
-36	-36	-.84				-.58
-32	-32	-.84	Not			-.57
-28	-28	-.89	Defined	Not		-.64
-24	-24	-1.00		Defined	Not	-.72
-20	-20	-1.15	-.84		Defined	-.88
-19.5	-19.5	-1.15	-.86			-.88
-16	-16	-.95	-.94	-.675	-.49	-.73
-15.5	-15.5	-.91	-.945	-.680	-.49	-.70
-13.0	-13.0	-.75	-.85	-.805	-.50	-.57
-12	-12	-.67	-.772	-.800	-.50	-.50
-11	-11	-.60	-.68	-.795	-.50	-.45
-8	-8	-.383	-.418	-.445	-.46	-.272
-4	-4	-.0625	-.075	-.094	-.105	-.045
0	0	.257	.260	.272	.290	.183
4	4	.577	.610	.630	.690	.412
8	8	.88	.960	1.00	.925	.640
11	11	1.10	1.23	1.10	.945	.800
12	12	1.19	1.28	1.09	.940	.870
13	13	1.26	1.29			.930
16	16	1.48	1.23			1.095
17	17	1.50			Not	1.100
20	20	1.38		Not	Defined	.98
24	24	1.22	Not	Defined		.80
28	28	1.20	Defined			.78
32	32	1.27				.86
36	36	1.40				.98
40	40	1.46				1.06

TABLE VII WING-PYLON LIFT COEFFICIENT ( $C_{LWP}$ )

FLAP SETTING ( $F_x$ )		$F_2$ (40/25)		$F_3$ (75/47)		$F_4$ (-28/-17.5)	
MAST ANGLE ( $\beta_M$ )		90°	0°	90°	0°	90°	0°
MACH NUMBER ( $M_N$ )		$M_N = 0 - .40$					
	$\alpha'_F / \alpha'_W$						
	-90	0	0	0	0		
	-80	-.325	-.245	-.235	-.190		
	-70	-.520	-.400	-.385	-.305		
	-60	-.610	-.480	-.450	-.333		
	-50	-.590	-.420	-.390	-.220		
	-40	-.410	-.265	-.240	-.105	-1.17	-.96
	-36	-.400	-.250	-.220	-.090	-1.08	-.83
	-32	-.425	-.260	-.240	-.095	-1.06	-.805
	-28	-.515	-.300	-.275	-.120	-1.11	-.85
	-24	-.660	-.380	-.340	-.160	-1.21	-.945
	-21.5	-.690	-.440	-.400	-.210	-1.31	-1.03
	-21.0	-.680	-.440	-.400	-.210	-1.33	-1.06
	-20	-.640	-.395	-.367	-.188	-1.37	-1.10
	-19.2	-.580	-.360	-.310	-.140	-1.38	-1.12
	-16	-.320	-.165	-.048	.040	-1.23	-.95
	-12	0	.0628	.272	.268	-.95	-.73
	-8	.320	.291	.591	.496	-.68	-.502
	-4	.64	.581	.910	.724	-.3843	-.275
	0	.96	.749	1.237	.952	-.0643	-.0460
	4	1.24	.975	1.460	1.170	.2557	.182
	8	1.49	1.205	1.680	1.390	.5757	.408
	11.0	1.68	1.380	1.80	1.50	.770	.590
	12	1.75	1.433	1.79	1.47	.840	.638
	13.6	1.80	1.500	1.71	1.39	.950	.730
	16	1.70	1.400	1.60	1.24	1.10	.864
	18.4	1.56	1.260	1.49	1.16	1.20	.950
	20	1.51	1.200	1.46	1.14	1.15	.890
	24	1.48	1.15	1.46	1.16	.93	.680
	28	1.48	1.20	1.54	1.29	.87	.630
	32	1.69	1.32	1.69	1.38	.91	.680
	36	1.76	1.41	1.78	1.44	1.05	.80
	40	1.80	1.47	1.80	1.48	1.14	.89

TABLE VIII WING-PYLON DRAG COEFFICIENT ( $C_{DWP}$ )

FLAP SETTING ( $F_x$ )		$F_I$ (0/0)				
MAST ANGLE ( $\beta_M$ )		90°				0°
MACH NUMBER ( $M_N$ )		0-.2	.4	.5	.6	0-.2
$\alpha_F / \alpha_W$						
-40	.575					.685
-36	.505		Not			.635
-32	.425		Defined	Not		.580
-28	.327			Defined	Not	.522
-24	.230	.312			Defined	.450
-20	.150	.175		.275		.370
-16	.089	.089		.135	.240	.295
-12	.042	.042		.050	.110	.246
-8	.025	.0250		.025	.052	.219
-4	.0170	.0170		.0170	.040	.212
0	.0170	.0170		.0170	.040	.212
4	.0353	.0353		.0353	.0555	.231
8	.0602	.0602		.0700	.1150	.262
12	.1000	.1100		.1500	.2500	.300
16	.1620	.1850		.2800		.354
20	.247	.3050				.436
24	.354	.500			Not	.512
28	.493			Not	Defined	.580
32	.600	Not		Defined		.642
36	.660	Defined				.698
40	.705					.748



TABLE WIV WING-PYLON COEFFICIENT ( $C_{DWP}$ )

FLAP SETTING ( $F_X$ )	$F_2$ (40/25)		$F_3$ (75/47)		$F_4$ (-28/-17.5)		
MAST ANGLE ( $\beta_M$ )	90°	0°	90°	0°	90°	0°	
MACH NUMBER ( $M_N$ )	$M_N = 0 - .4$						
	$\alpha'_{FW}/\alpha_W$						
-90		1.18	.93	1.145	.90		
-80		1.10	.91	1.050	.878		
-70		.93	.855	.89	.822		
-60		.705	.775	.67	.740		
-50		.565	.665	.507	.640		
-40		.430	.540	.450	.550	.622	.734
-36		.335	.468	.400	.525	.566	.682
-32		.245	.405	.350	.500	.500	.632
-28		.180	.352	.309	.480	.430	.575
-24		.130	.310	.278	.462	.345	.502
-20		.090	.282	.260	.450	.255	.422
-16		.065	.263	.243	.440	.186	.350
-12		.058	.253	.246	.445	.130	.298
-8		.063	.254	.261	.464	.086	.267
-4		.081	.282	.288	.494	.055	.248
0		.109	.313	.328	.536	.047	.239
4		.148	.356	.378	.590	.0471	.243
8		.200	.410	.444	.658	.0653	.258
12		.275	.490	.530	.728	.0950	.284
16		.380	.566	.642	.789	.138	.332
20		.528	.630	.730	.839	.195	.402
24		.630	.690	.790	.880	.287	.476
28		.710	.748	.838	.920	.414	.540
32		.764	.800	.883	.955	.510	.600
36		.805	.845	.950	.985	.583	.648
40		.865	.888	1.025	1.015	.640	.695

TABLE WV WING WAKE DEFLECTION ON HORIZONTAL STABILIZER ( $\epsilon_{W/H}$ )

MACH NUMBER ( $M_N$ )		$M_N = 0 - .2$			
FLAP SETTING ( $F_X$ )		$F_1(0/0)$	$F_2(40/25)$	$F_3(75/47)$	$F_4(-28/7.5)$
	$\alpha_W$				
AIRPLANE MODE ( $\beta_M = 90^\circ$ )					
	-90	0	0	0	0
	-25.0	0	0	0	0
	-17.94	0	0	2.40	0
	-8.82	0	2.00	5.48	0
	-2.353	1.92	4.38	7.70	0
	0.0	2.52	5.25	8.50	.800
	12	6.00	9.90	12.58	4.88
	13.8	6.50	10.75	12.80	5.70
	16	7.18	11.40	12.35	6.24
	16.5	7.40	11.40	12.10	6.40
	18.8	7.20	10.60	11.10	6.80
	20	6.7	10.00	10.40	6.55
	24	4.20	6.50	7.10	4.00
	28	0	0	0	0
	40	0	0	0	0
HELICOPTER MODE ( $\beta_M = 0^\circ$ )					
	-90	0	0	0	0
	-28.65	0	0	0	0
	-22.15	0	0	0	0
	-17.8	0	0	0	0
	-15.5	0	0	0	0
	-12.41	0	1.2	2.0	0
	-7.6	0	3.2	3.75	0
	-2.94	2.0	5.0	5.4	0
	0.0	3.1	6.15	6.3	1.00
	8	6.3	9.25	9.2	3.72
	12	7.8	11.00	10.7	5.08
	13.0	8.30	11.40	11.0	5.40
	14.8	9.00	11.95	11.5	6.00
	16.0	9.6	11.90	11.4	6.44

TABLE WV WING WAKE DEFLECTION ON HORIZONTAL STABILIZER (CONT)

	$F_1$ (0/0)	$F_4$ (40/25)	$F_3$ (75/47)	$F_4$ (-28/-17.5)
$\alpha_W$				
HELICOPTER MODE (CONT)				
18.0	10.00	11.20	11.2	7.00
20	9.30	10.20	10.20	6.70
24	6.00	6.80	6.80	4.20
28-40	0	0	0	0

TABLE W VI - WING-PYLON DIHEDRAL STABILITY

AT  $C_{L_{WP}} = 0$   $\left\{ \begin{array}{l} C_{1\beta_{WP}} \\ C_{M_N} \end{array} \right\} \left. \begin{array}{l} C_L = 0 \\ M_N = 0 \end{array} \right\}$  PER RAD.

FLAP	$F_1(0/0) \& F_4(-28/-17.5)$			$F_2(40/25)$			$F_3(75/47)$		
SIDESLIP/ $\beta_F$	0	5	10	0	5	10	0	5	10
$\beta_M$									
0	-.06	-.065	+.045	-.2	-.094	-.045	-.223	-.015	-.125
30	-.032	-.061	-.02	-.079	-.074	-.147	-.085	-.087	-.225
60	-.021	-.051	-.052	.067	-.052	-.148	-.008	-.088	-.222
90	-.025	-.025	+.035	-.11	-.025	-.025	+.033	-.04	-.08

TABLE W VII - WING-PYLON DIHEDRAL STABILITY

AS A FUNCTION OF WING LOADING  $[C_{1\beta_{WP}}/C_{L_{WP}}]$

FLAP	$F_1(0/0) \& F_4(-28/-17.5)$			$F_2(40/25)$			$F_3(75/47)$		
SIDESLIP/ $\beta_F$	0	5	10	0	5	10	0	5	10
$\beta_M$									
0	0	0	0	+.105	+.013	+.035	+.104	-.045	+.110
30	+.049	+.079	-.016	+.069	+.062	+.062	+.049	+.047	+.107
60	+.039	+.071	-.018	+.032	+.061	+.058	-.013	+.055	+.082
90	-.065	-.065	0	0	-.085	-.065	-.043	-.045	0

TABLE AI AILERON EFFECTIVENESS ( $C_{l\delta a}$ )

$$C_{l\delta a} = .005/\text{deg} \quad \alpha_w < 8^\circ$$

$$\text{Where } \delta_a = (\delta_l - \delta_r)/2$$

$$\text{for } \delta_F = 0/0; (F_1)$$

$$\beta_M = 90^\circ; (\text{Airplane})$$

TABLE AII AILERON EFFECTIVENESS CORRECTION  
FOR FLAP AND MAST ( $K_{l\delta a}$ ) FOR  $\alpha_w < |8^\circ|$

Flap Setting	Mast Angle	$K_{l\delta a}$ (1)
F <sub>1</sub>	0°	.68
	90°	1.00
F <sub>2</sub>	0°	.66
	90°	.73
F <sub>3</sub>	0°	.45
	90°	.34
F <sub>4</sub>	0°	.67
	90°	.90

$$(K_{l\delta a} = 0 @ \alpha_w = \pm 25^\circ)$$

(1) Straight Line Variation with Mast Angle

TABLE AIII AILERON EFFECT ON WING LIFT ( $C_{L\delta a}$ )

Flap Setting	Mast Angle	$C_{L\delta a}$ (per Deg)
F <sub>1</sub>	All	0
F <sub>2</sub>	All	0
F <sub>3</sub>	All	0
F <sub>4</sub>	All	0

TABLE AIV AILERON YAW ( $C_{n_{\delta a}}$ )

$$C_{n_{\delta a}} = K_{n_{o_{\delta a}}} + K_{n_{c_{\delta a}}} C_L C_{l_{\delta a}}$$

Where:

Flap Setting	Mast Angle	$K_{n_{c_{\delta a}}}$	$K_{n_{o_{\delta a}}}$
F <sub>1</sub>	0°	.00046	-.0348
	30°	.00092	-.0354
	90°	.00143	-.0238
F <sub>2</sub>	0°	.00046	-.0137
	30°	.00109	-.0052
	90°	.00103	-.0137
F <sub>3</sub>	0°	-.00003	-.0232
	30°	.00035	-.0050
	90°	.00029	-.0157
F <sub>4</sub>	0°	0	-.0205
	30°	0	-.0301
	90°	0	-.0215

SUBSYSTEM NO. 5: MODEL 301 HORIZONTAL TAIL AERODYNAMIC DATA

Equations	Constant	M301 Value
$\alpha_{HL}$ (For lift)	$\tau_e$	0.0 for $M_N < 1.0$ 0.565 for $M_N > 1.0$
$\alpha_{HD}$ (For drag)	$\tau_e$	0.565
$q_H$	$\eta_H$	1 [The effect is included in calculating $\epsilon_{w/H}$ ]
$L_H$	$C_{L_H} = f(\alpha_H, \delta_e, M_N)$	Tables HI, HII
$D_H$	$C_{D_H} = f(\alpha_H, M_N)$	-0.0017 Table H IV
$M'_H$	$C_{M_H} = f(\alpha_H, M_N)$	0
$\alpha_H$	$K_e = f(\delta_e, M_N)$	Table H III

TABLE HI HORIZONTAL STABILIZER LIFT COEFFICIENT ( $C_{LH}$ )

ELEVATOR ANGLE	0°	10°	15°	20°	-10°	-15°	-20°
MACH NUMBER	$M_N = 0 - .2$						
$\alpha_H$							
-90	0	0	0	0	0	0	0
-80	-.425	-.560	-.580	-.600	-.360	-.285	-.220
-70	-.720	-.865	-.890	-.920	-.600	-.490	-.380
-60	-.900	-1.060	-1.090	-1.120	-.770	-.640	-.510
-50	-1.002	-1.175	-1.205	-1.240	-.890	-.745	-.600
-40	-1.050	-1.240	-1.260	-1.300	-.960	-.800	-.640
-36	-1.030	-1.230	-1.255	-1.290	-.890	-.735	-.600
-28	-1.010	-1.210	-1.240	-1.280	-.840	-.680	-.560
-24	-.980	-1.185	-1.220	-1.260	-.780	-.615	-.500
-20	-.930	-1.160	-1.198	-1.235	-.690	-.500	-.420
-18.4	-.920	-1.200	-1.210	-1.240	-.660	-.540	-.480
-17.5	-.930	-1.260	-1.250	-1.250	-.710	-.565	-.450
-16.8	-.990	-1.310	-1.290	-1.310	-.740	-.550	-.420
-16.0	-1.12	-1.40	-1.330	-1.330	-.710	-.510	-.380
-15.6	-1.10	-1.44	-1.380	-1.350	-.700	-.480	-.350
-14.2	-1.01	-1.40	-1.55	-1.450	-.610	-.400	-.270
-12.5	-.880	-1.31	-1.49	-1.60	-.480	-.280	-.150
-12	-.852	-1.260	-1.464	-1.59	-.444	-.240	-.110
8	.568	.160	-.044	-.18	.976	1.180	1.330
12.0	.850	.442	.240	.100	1.250	1.420	1.500
12.2	.860	.450	.260	.120	1.270	1.430	1.480
13.0	.920	.520	.330	.170	1.30	1.370	1.450
15.0	1.0	.650	.450	.290	1.200	1.270	1.360
15.0	.98	.690	.475	.320	1.160	1.240	1.320
16.8	.94	.700	.490	.340	1.150	1.200	1.320
18.0	.89	.680	.500	.370	1.130	1.220	1.340
20	.88	.600	.465	.380	1.180	1.280	1.380
24	.935	.660	.455	.330	1.300	1.380	1.440
28	1.00	.730	.500	.380	1.370	1.440	1.500
32	1.05	.780	.540	.400	1.430	1.490	1.540
36	1.08	.820	.560	.410	1.470	1.535	1.570
40	1.10	.840	.570	.410	1.510	1.560	1.590



TABLE III HORIZONTAL STABILIZER LIFT COEFFICIENT ( $C_{LH}$ )

ELEVATOR ANGLE		$\alpha_e = 0^\circ$			
MACH NUMBER ( $M_N$ )		0 - .2	.4	.5	.6
$\alpha_H$	-20	-.930	Not Defined	Not Defined	Not Defined
	-16	-1.12	Defined	Defined	Defined
	-12.2	-.870	-.92		
	-12.0	-.852	-.92		
	-8.8	-.630	-.680	-.72	
	-8.0	-.568	-.620	-.660	-.50
	-6.5	-.460	-.500	-.540	-.55
	-4.0	-.284	-.310	-.330	-.350
	0	0	0	0	0
	4.0	.284	.310	.330	.350
	4.2	.310	.330	.340	.390
	8.0	.568	.620	.600	.440
	16.5	.820	.800	.570	
	12.0	.850	.798	.560	Not Defined
	15.5	1.00	Not Defined	Not Defined	Defined
	16.0	.980	Defined	Defined	
	20.0	.880			

TABLE IIII - ELEVATOR/RUDDER EFFECTIVENESS ( $\tau_e/\tau_r$ )

CORRECTED FOR MACH NO. AND DEFLECTION EFFECTS

ELEVATOR ANGLE - DEG	MACH NUMBER	$K_e$ OR $K_r$
$\pm 15^\circ$	0-.2	1.0
	.4	.965
	.5	.950
	.6	.930
$\pm 15^\circ$ $\pm 20^\circ$	0-.2	1.0
		.920

TABLE HIV HORIZONTAL STABILITY DERIVATIVE DRAG COEFFICIENT ( $C_{DH}$ )

ELEVATOR ANGLE		$\delta_e = 0^\circ$			
MACH NUMBER ( $M_N$ )		0-.2	.4	.5	.6
	$\alpha_H$				
	-90	.92			
	-80	.91			
	-70	.87			
	-60	.81	NOT		
	-50	.72	DEFINED	NOT	
	-40	.60		DEFINED	NOT
	-36	.54			DEFINED
	-32	.47			
	-28	.39			
	-24	.30			
	-20	.20			
	-16	.115	.135		
	-12	.068	.068	.088	
	-8	.035	.035	.035	.045
	-4	.015	.015	.015	.015
	0	.00875	.00875	.00875	.00875
	4	.015	.015	.015	.015
	8	.035	.035	.045	.065
	12	.068	.075	.105	
	16	.115	.145		
	20	.20			NOT
	24	.34		NOT	DEFINED
	28	.48	NOT	DEFINED	
	32	.61	DEFINED		
	36	.72			
	40	.80			

**SUBSYSTEM NO. 6: MODEL 301 VERTICAL FIN AERODYNAMIC DATA**

Equations	Constant	M301 Value
$\beta_{VL,R_0}$	$\left(1 - \frac{\partial \sigma}{\partial \beta_F}\right) = f(\beta_F, \beta_M, F_X, \alpha_F)$	Tables V IV, V V, V VI
$\beta_{VYL,R} \text{ \& } \beta_{VDL,R}$	$K_R = f(\delta_r, M_N)$	Table H III
	$\tau_r$	.385
	$\frac{\partial \sigma}{\partial \beta}$	-0.1
	$\frac{\partial \sigma}{\partial r}$	0.0
$q_{VL,R}$	$\eta_V$	1
$Y_{VL,R}$	$C_{Y_V} = f(\beta_V, \delta_r, M_N)$	Tables VI, VII
$D_{VL,R}$	$C_{D_V} = f(\beta_V, \delta_r, M_N)$	Table V III

TABLE VI VERTICAL STABILIZER LIFT COEFFICIENT ( $C_{Y_V}$ )

RUDDER ANGLE	0°	10°	20°	-10°	-20°
MACH NUMBER	$M_N = 0 - .2$				
$\beta_V$					
-90	0	0	0	0	0
-40	-1.0	-.96	-.92	-1.07	-1.11
-32	-.93	-.84	-.82	-1.02	-1.12
-28	-.94	-.90	-.87	-1.02	-1.09
-26	-.98	-.93	-.91	-1.07	-1.11
-24	-1.03	-.98	-.92	-1.11	-1.15
-22	-1.05	-.98	-.85	-1.14	-1.19
-20	-1.05	-.88	-.74	-1.15	-1.22
-18	-.96	-.77	-.64	-1.14	-1.24
-16	-.86	-.66	-.52	-1.05	-1.18
-12	-.635	-.45	-.30	-.84	-.99
-8	-.425	-.225	-.09	-.63	-.76
8	.425	.625	.76	.23	.10
12	.635	.84	.99	.44	.31
16	.86	1.05	1.78	.66	.52
18	.96	1.14	1.24	.77	.64
20	1.05	1.15	1.22	.88	.74
22	1.05	1.14	1.19	.98	.85
24	1.03	1.11	1.15	.98	.92
26	.98	1.07	1.11	.93	.91
28	.94	1.02	1.09	.90	.87
32	.93	1.02	1.12	.84	.82
40	1.0	1.07	1.11	.96	.92
90	0	0	0	0	0

TABLE VII VERTICAL STABILIZER LIFT COEFFICIENT ( $C_{Y_V}$ )

RUDDER ANGLE		$\delta_r = 0^\circ$			
MACH NUMBER ( $M_N$ )		0-.2	.4	.5	.6
	$\beta_V$				
	-24	-1.05	NOT DEFINED	NOT DEFINED	NOT DEFINED
	-22	-1.05			
	-20	-.96			
	-16	-.86	-.60	-.45	
	-12	-.635	-.60	-.45	-.32
	-10	-.52	-.575	-.445	-.34
	-8	-.425	-.52	-.44	-.34
	-6	-.33	-.43	-.405	-.33
	-4	-.22	-.30	-.32	-.30
	0	0	0	0	0
	4	.22	.30	.32	.30
	6	.33	.43	.405	.33
	8	.425	.52	.44	.34
	10	.52	.575	.445	.34
	12	.635	.60	.45	.32
	16	.86	.60	.45	
	20	.96	NOT DEFINED	NOT DEFINED	NOT DEFINED
	22	1.05	DEFINED	DEFINED	DEFINED
	24	1.05			

TABLE VIII VERTICAL STABILIZER DRAG COEFFICIENT ( $C_{Dv}$ )

RUDDER ANGLE		$\delta_r = 0^\circ$			
MACH NUMBER	( $M_N$ )	0 - .2	.4	.5	.6
	$\beta_v$				
	-90	.800			
	-40	.600			
	-32	.550	Not Defined		
	-28	.435		Not Defined	Not Defined
	-24	.290			
	-20	.162			
	-16	.080	.165		
	-12	.040	.0750	.100	.160
	- 8	.0120	.0220	.0350	.060
	- 4	.0070	.0070	.0075	.010
	0	.00355	.00355	.00355	.00355
	4	.0070	.0070	.0075	.010
	8	.0120	.0220	.0350	.060
	12	.040	.0750	.100	.160
	16	.080	.165		
	20	.162			
	24	.290		Not Defined	Not Defined
	28	.435			
	32	.550	Not Defined		
	40	.600			
	90	.800			

TABLE VIV - SIDEWASH FACTOR  $(1 - \delta\sigma / \delta\beta_F)$  FOR FLAPS  $F_1$  &  $F_4$

FLAPS		$F_1$ (0/0) AND $F_4$ (-28/-17.5)						
		SIDESLIP ANGLE $ \beta_F $ - DEG						
$\beta_M$ DEG	$\alpha_F$ DEG	0	4	8	12	16	20 TO 50	>50
0	<-12	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3	1.0	1.1	1.05	1.015	0.985	1.01	1.0
	0.4	1.0	1.038	1.044	0.965	0.933	1.0	1.0
	6.8	1.0	0.863	0.810	0.772	0.787	0.958	1.0
	13.2	1.0	0.524	0.517	0.474	0.491	0.673	1.0
	> 28	1.0	1.0	1.0	1.0	1.0	1.0	1.0
30	<-10.5	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.	1.0	1.13	1.072	0.97	1.025	1.0	1.0
	0.5	1.0	1.248	1.093	0.977	1.015	1.056	1.0
	6.9	1.0	0.995	0.961	0.865	0.845	0.953	1.0
	13.3	1.0	0.677	0.595	0.523	0.526	0.681	1.0
	> 28	1.0	1.0	1.0	1.0	1.0	1.0	1.0
60	<-9	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.	1.0	1.15	1.05	1.0	1.06	1.1	1.0
	0.5	1.0	1.21	1.08	0.975	1.02	1.025	1.0
	6.9	1.0	0.945	0.975	0.9	0.88	0.92	1.0
	13.3	1.0	0.69	0.645	0.585	0.59	0.7	1.0
	> 28	1.0	1.0	1.0	1.0	1.0	1.0	1.0
90	<-8	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.	1.0	1.09	1.10	1.18	1.15	1.04	1.0
	0.2	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	7.0	1.0	0.834	0.865	0.866	0.842	0.924	1.0
	13.5	1.0	0.659	0.676	0.622	0.642	0.680	1.0
	> 28	1.0	1.0	1.0	1.0	1.0	1.0	1.0

TABLE VV SIDEWASH FACTOR  $(1 - \frac{\partial \sigma}{\partial \beta_F})$  FOR FLAPS  $F_2$

FLAPS		$F_2 - (40/25)$						
		SIDESLIP ANGLE $ \beta_F  - \text{DEG}$						
$\beta_M$ DEG	$\alpha_F$ DEG	0	4	8	12	16	20 to 50	>50
0	$\leq -22.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.2	1.0	1.315	1.26	1.185	1.1	1.059	1.0
	1.0	1.0	1.228	1.208	1.12	1.045	1.0	1.0
	7.3	1.0	0.89	0.91	0.86	0.809	0.86	1.0
	13.5	1.0	0.535	0.59	0.396	0.443	0.678	1.0
	$\geq 28.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
30	$\leq -21.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.1	1.0	1.065	1.027	1.055	1.14	1.055	1.0
	1.1	1.0	1.1	1.115	1.052	1.12	1.065	1.0
	7.4	1.0	1.025	.935	.972	.882	.86	1.0
	13.6	1.0	.884	0.82	0.629	0.456	0.689	1.0
	$\geq 28.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
60	$\leq -19.5$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.0	1.0	1.02	1.03	1.08	1.117	1.04	1.0
	1.2	1.0	0.945	1.07	0.998	1.05	1.09	1.0
	7.4	1.0	1.03	0.985	1.015	0.95	0.908	1.0
	13.6	1.0	0.915	0.9	0.8	0.61	0.745	1.0
	$\geq 26.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
90	$\leq -18.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.1	1.0	1.07	1.121	1.16	1.072	1.04	1.0
	1.2	1.0	0.984	1.15	1.09	1.05	1.064	1.0
	5.4	1.0	0.982	1.035	1.03	0.993	1.015	1.0
	14.0	1.0	0.842	0.74	0.77	0.695	0.725	1.0
	$\geq 28.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0



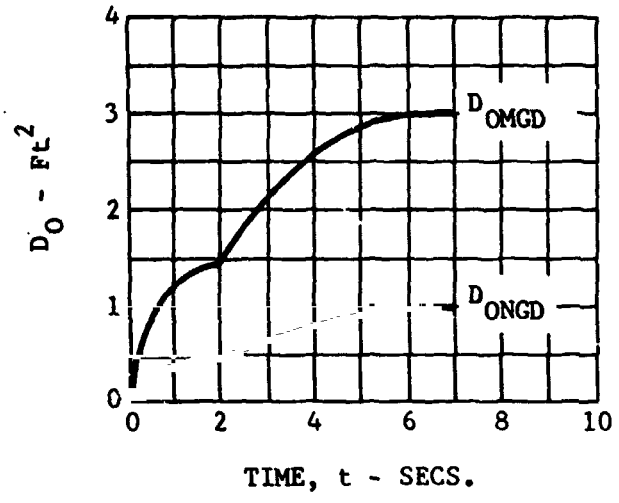
TABLE VVI SIDEWASH FACTOR  $(1 - \frac{\alpha}{\alpha_F})$  FOR FLAPS F<sub>3</sub>

FLAPS		F3 (75/47)						
$\beta_M$ DEG	$\alpha_F$ DEG	SIDE SLIP ANGLE $ \beta_F $ -					DEG	
		0	4	8	12	16	20 to 50	>50
0	$\leq -30.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.1	1.0	1.8	1.22	1.155	1.105	1.048	1.0
	1.2	1.0	1.128	1.185	1.125	1.045	1.01	1.0
	5.3	1.0	0.846	0.99	0.948	0.92	0.862	1.0
	13.5	1.0	0.535	0.65	0.44	0.51	0.65	1.0
	$\geq 28.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
30	$\leq -29.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.0	1.0	1.05	1.072	1.14	1.135	1.08	1.0
	1.3	1.0	0.979	1.01	1.105	1.1	1.035	1.0
	5.5	1.0	0.82	0.862	0.998	0.986	0.932	1.0
	13.5	1.0	0.65	0.72	0.7	0.52	0.78	1.0
	$\geq 28.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
60	$\leq -28.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-3.0	1.0	1.025	1.082	1.15	1.115	1.075	1.0
	1.3	1.0	0.915	1.005	1.1	1.1	1.04	1.0
	5.5	1.0	0.855	0.90	1.015	0.955	0.995	1.0
	13.5	1.0	0.73	0.8	0.8	0.61	0.82	1.0
	$\geq 28.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
90	$\leq -26.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0
	-2.9	1.0	1.058	1.15	1.145	1.075	1.04	1.0
	1.3	1.0	1.0	1.12	1.12	1.05	1.088	1.0
	5.6	1.0	0.905	1.005	1.03	0.99	1.028	1.0
	13.5	1.0	0.782	0.76	0.72	0.66	0.71	1.0
	$\geq 28.0$	1.0	1.0	1.0	1.0	1.0	1.0	1.0

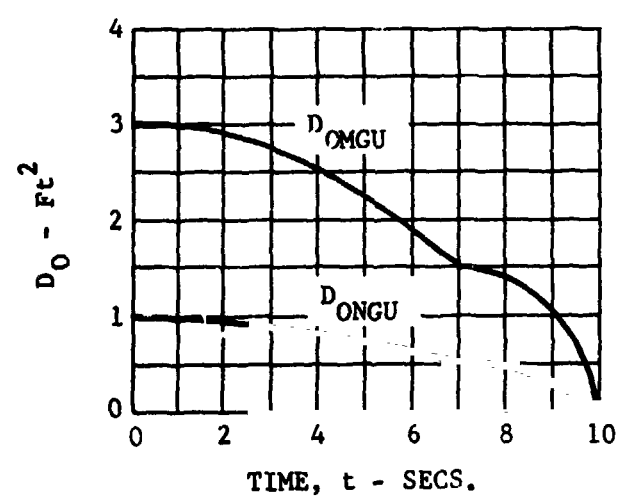
**SUBSYSTEM NO. 7: MODEL 301 LANDING GEAR DATA**

Equation	Constant
$D_{MG}$	$D_{OMGD} f(t)$
$D_{NG}$	$D_{ONGD} f(t)$

M301 Value



$D_{MG}$	$D_{OMGU} f(t)$
$D_{NG}$	$D_{ONGU} f(t)$



SUBSYSTEM NO. 8(a): MODEL 301 CONTROL SYSTEM DATA

Equation	Constant	M301 Value
$\theta_{OL}, \theta_{OR}$	$\frac{\partial \theta_0}{\partial X_{COL}}$	Table C IV
	$\theta_{0LL} = f(\beta_M)$	Table C IV
	$\frac{\partial \theta_0}{\partial X_{LT}} = f(\beta_M)$	Table C III
$B_{1R}, B_{1L}$	$\frac{\partial B_1}{\partial X_{LN}} = f(\beta_M)$	Table C I
	$\frac{\partial B_1}{\partial X_{PD}} = f(\beta_M, V_T)$	Table C II
$\delta_e$	$\frac{\partial \delta_e}{\partial X_{LN}}$	-4.16
$\delta_R$	$\frac{\partial \delta_R}{\partial X_{PD}}$	8.0
$\delta_a$	$\frac{\partial \delta_a}{\partial X_{LT}}$	+ 3.93
$\dot{\beta}_M$	$\dot{\beta}_M = f(\beta_M)$	Table C V
$i_H$	$i_{H0}  _{X_{LN}} = X_{LNN}$	$\begin{bmatrix} 0 \end{bmatrix}$
	$i_{H1}$	
	$i_{H2}$	
	$i_{H3}  _{X_{COL}} = 0$ $i_{H4}$	

TABLE CI FORE AND AFT CYCLIC PITCH ( $\delta B_1 / \delta X_{LN}$ )

MAST ANGLE (DEG)	$\frac{\delta B_1}{\delta X_{LN}}$ ( $^{\circ}/in$ )
0	2.1
10	2.09
20	1.98
30	1.81
40	1.60
50	1.35
60	1.04
70	.71
80	.362
90	0

$(X_{LN} = \pm 4.8 \text{ in}) \quad (B_1 = 10.0625^{\circ} @ \beta_M = 0^{\circ})$

TABLE CII DIFFERENTIAL CYCLIC PITCH ( $\delta B_1 / \delta X_{PD}$ )

MAST ANGLE (DEG)	$\frac{\delta B_1}{\delta X_{PD}}$ ( $^{\circ}/in$ )		
	0 - 60 KTS	80 KTS	100 KTS
0	1.6	1.04	.40
10	1.58	1.025	.394
20	1.51	.975	.375
30	1.39	.90	.345
40	1.225	.795	.305
50	1.035	.67	.257
60	.803	.52	.200
70	.55	.325	.137
80	.28	.18	.069
90	0	0	0

$X_{PD} = \pm 2.5 \text{ in}$

$\Delta B_1 = \pm 4.0^{\circ} \text{ 0-60}$   
 $\quad \quad \quad \pm 2.6^{\circ} \text{ 80}$   
 $\quad \quad \quad \pm 1.0^{\circ} \text{ 100}$

TABLE CIII DIFFERENTIAL COLLECTIVE PITCH

MAST ANGLE (DEG)	$\delta\theta_o/\delta X_{LT}$ (°/in)
0	0.625
10	.606
20	.575
30	.541
40	.50
50	.438
60	.365
70	.293
80	.209
90	.121

$(X_{LT} = \pm 4.8 \text{ in } )$

$(\Delta\theta_o = \pm 3^\circ \text{ @ } \beta_M = 0)$

TABLE CIV COLLECTIVE PITCH ( $\delta\theta_c/\delta X_{Col}$ ;  $\theta_{oLL}$ )

MAST ANGLE (DEG)	$\delta\theta_c/\delta X_{Col}$ (°/in)	$\theta_{oLL}$ (DEG)
0	1.17	35.5
10	1.12	36.0
20	1.08	37.1
30	1.00	39.5
40	0.85	42.7
50	0.69	46.6
60	0.50	50.4
70	0.31	53.5
80	0.13	55.9
90	0	57.5

TABLE CV CONVERSION RATE ( $\dot{\beta}_M$ )

MAST ANGLE (DEG)	$\dot{\beta}_M$ (°/Sec)
-5	3.0 <sup>(1)</sup>
0	3.0 <sup>(1)</sup>
2	3.0 <sup>(1)</sup>
2.5	15.0 <sup>(2)</sup>
19	14.25
20	13.3
30	12.45
40	11.7
50	11.4
60	11.4
70	11.63
80	12.5
87	14.0 <sup>(2)</sup>
87.5	2.8 <sup>(1)</sup>
90	2.8 <sup>(1)</sup>

(1) When conversion starts at mast angle of  
-5, 0 or 90;  $\dot{\beta}_M = 2.8$  or 3.0. When conversion  
stops at mast angle of -5, 0, or 90;  $\dot{\beta}_M = 0$

(2)  $\Delta t$  from quarter rate to max rate;  $\Delta t = .05$  sec

TABLE CVI  $X_{THR}$  VS  $X_{COL}$

$X_{COL}$ (INCH)	$X_{THR}$ (DEG)
0	46.0
1.0	51.0
2.0	56.2
3.0	61.2
4.0	65.8
5.0	70.5
6.0	75.0
7.0	79.3
8.0	83.7
9.0	88.0
10.0	92.3
11.0	96.2
12.0	100.0

TABLE CVII  $\beta$  GOVERNOR GAIN

$\beta_M$ (DEG)	$K_{RPG}$
0	0.1
15	0.1
30	0.1
45	0.1
60	0.1
75	0.1
90	0.1

TABLE CVIII  $\beta$  GOVERNOR LIMIT

$\beta_M$	$\theta_{CG} \text{ Max}$
0	-6, +32
15	-6, +32
30	-6, +32
45	-6, +32
60	-6, +32
75	-6, +32
90	-6, +32



**SUBSYSTEM 8(b): MODEL 301 FORCE GRADIENT DATA**

Equation	Constant	M301 Values
$F_{LN}$	$F_{LNO}$	2
	$F_{LN1}$	0 for $V_T < V_1$
		.0264 for $V_T \geq V_1$
	$F_{LN2}$	0 for $V_T < V_2$
		.056 for $V_T \geq V_2$
	$V_1$	35
$V_2$	175	
$F_{LT}$	$F_{LTO}$	1
	$F_{LT1}$	0 for $V_T < V_1$
		.0107 for $V_T \geq V_1$
	$F_{LT2}$	0 for $V_T < V_2$
		.0336 for $V_T \geq V_2$
	$V_1$	35
$V_2$	175	
$F_{PD}$	$F_{PDO}$	6
	$F_{PD1}$	0 for $V_T < V_1$
		.148 for $V_T \geq V_1$
	$F_{PD2}$	0 for $V_T < V_2$
		1.21 for $V_T \geq V_2$
	$V_1$	0
$V_2$	135	

$F_{COL}$

$F_4$

0

NOTE: Breakout forces are adjusted by the pilot by varying the friction.

**SUBSYSTEM NO. 9: M301 INERTIA COEFFICIENT DATA (AFT CG)**

Equation	Constant	M301 Value
$I_{XX}$	$K_{I1}$	20.5 Slugs - Ft <sup>2</sup> /deg
$I_{YY}$	$K_{I2}$	11.24 Slugs - Ft <sup>2</sup> /deg
$I_{ZZ}$	$K_{I3}$	9.26 Slugs - Ft <sup>2</sup> /deg
$I_{XZ}$	$K_{I4}$	1.76 Slugs - Ft <sup>2</sup> /deg

**SUBSYSTEM NO. 14: M301 GROUND EFFECT ROLLING MOMENT COEFFICIENT DATA**

Equation	Constant	M301 Value
$l_A$	$l_{G0}$	$-8270 \frac{\text{Ft. lbs.}}{\text{Deg.}}$
	$l_{G1}$	$26186 \frac{\text{Ft. lbs.}}{\text{Deg.} \cdot \text{Ft.}}$
	$l_{G2}$	$-23369 \frac{\text{Ft. lbs.}}{\text{Deg.} \cdot \text{Ft}^2}$
	$l_{G3}$	$6336 \frac{\text{Ft. lbs.}}{\text{Deg.} \cdot \text{Ft}^3}$

SUBSYSTEM NO. 17: M301 COLLECTIVE GOVERNOR DATA

Equation

Constant

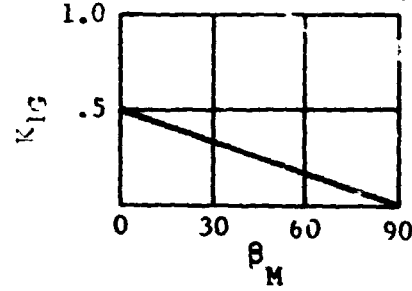
M301 Value

$$\bar{\theta}_{OG}$$

$$\frac{\omega_{OINT}}{\omega_{RO}}$$

11.3

$$K_{IG} (\beta_M)$$



$$K_{RPG} (\beta_M)$$

Table C VII

$$\left. \begin{array}{l} \theta_{OG \max} \\ \theta_{OG \min} \end{array} \right\}$$

Table C VIII

**SUBSYSTEM NO. 18: MCDL 301 ENGINE DATA**

Equations	Constant	M301 Value
HP	$K_1$	-0.94
	$K_2$	1.02
	$K_3$	0
RPM <sub>RO</sub>	$K_4$	13100
	$K_5$	235
	$K_6$	475
$T_a$	$K_7$	288
HP <sub>ROC</sub>	$K_8$	0.266
	$K_9$	-8.97
	$K_{10}$	0
	$K_{11}$	0.0032
	$K_{12}$	0.875
	$K_{13}$	0.00125
	$K_{14}$	0
$\epsilon_s$	$t_D$	0.1
$\frac{dHP_{ROS}}{dt_{PTG}}$	pctmxs	6.0
$\frac{dHP_{ROS}}{dt_{TH}}$	pctmxp	6.0

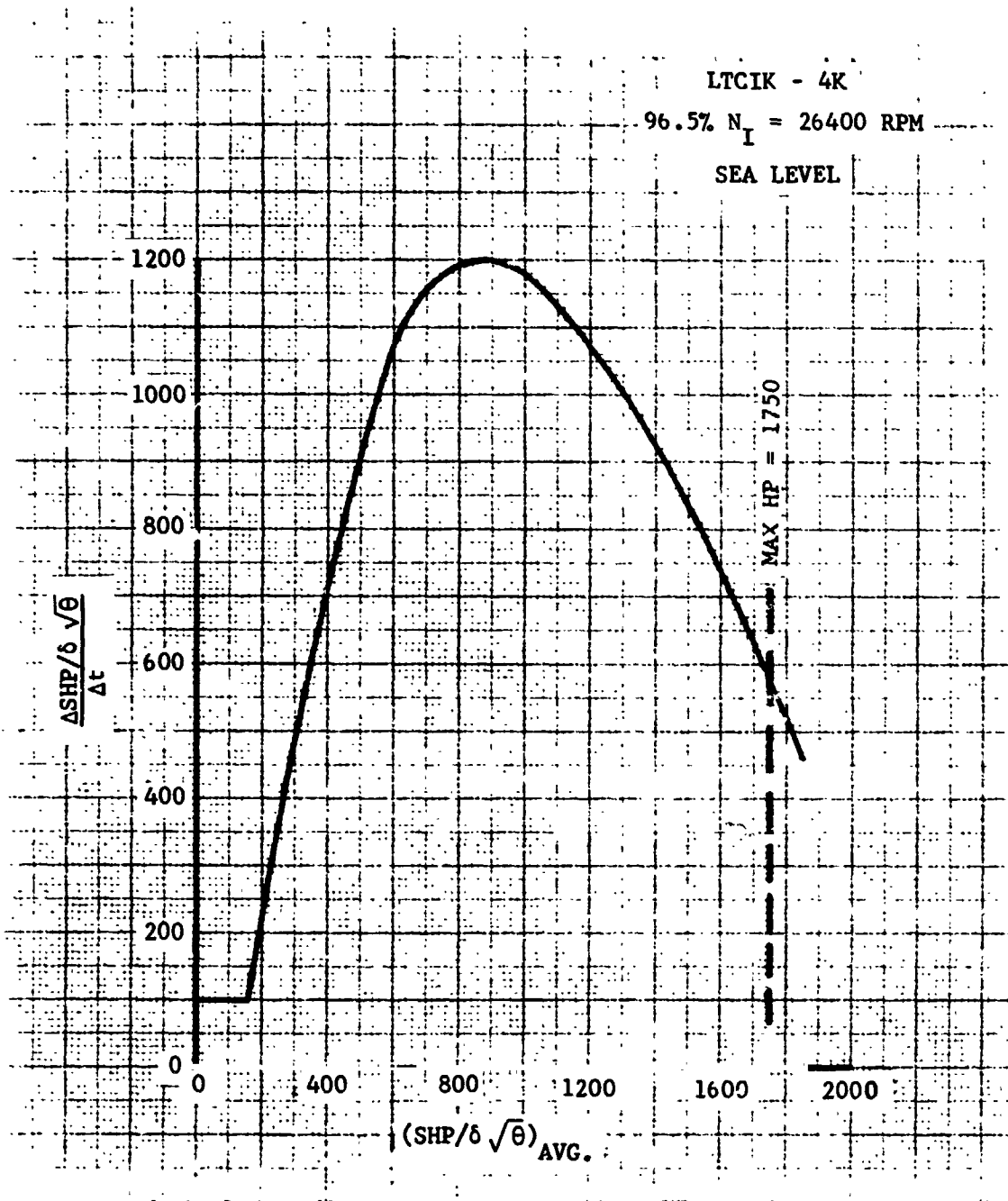


Figure B1. T53 Engine Acceleration Characteristics

SUBSYSTEM NO. 19: MODEL 301 DRIVE SYSTEM DATA

Equation	Constant	M301 Value
$\ddot{\xi}$	$I_1$	824.0 Slug-Ft <sup>2</sup>
$F_1(t), \Omega_{RPT}$	$\theta_{RPT1}$	35.2
$\Omega_{INT}$	$\theta_{INT1}$	11.3
$\Omega_R$	$\Omega_{RO}$	



SUBSYSTEM NO. 20: M301 SCAS DATA

Equation	Constant	M301 Value
PSCAS	$K_{1PU}$ (U)	Table SAS I
	$K_{9PU}$ (U)	
	$K_{1P\beta_M}$ ( $\beta_M$ )	Table SAS I
	$K_{10P\beta_M}$ ( $\beta_M$ )	
	$T_{1P}$	0.5 Sec
	$T_{2P}$	3.15 Sec
	$T_{3P}$	3.15 Sec
	$T_{4P}$	1.0 Sec
YSCAS	$K_{1YU}$ (U)	Table SAS II
	$K_{5YU}$ (U)	
	$K_{1Y\beta_M}$ ( $\beta_M$ )	Table SAS II
	$K_{5Y\beta_M}$ ( $\beta_M$ )	
	$T_{1Y}$	0.6 Sec
	$T_{2Y}$	2.7 Sec
	$T_{3Y}$	2.7 Sec
	$T_{4Y}$	1.0 Sec
RSCAS	$K_{1RU}$ (U)	Table SAS I
	$K_{9RU}$ (U)	

Equation	Constant	M301 Value
	$K_{1R\beta_M} (\beta_M)$	Table SAS III
	$K_{10R\beta_M} (\beta_M)$	
	$T_{1R}$	1.0 Sec
	$T_{2R}$	3.0 Sec
	$T_{3R}$	3.0 Sec
ESAS	ESAS <sub>max</sub>	1.54 Inch
RSAS	RSAS <sub>max</sub>	0.80 Inch
ASAS	ASAS <sub>max</sub>	1.54 Inch
PHOLD	PHOLD <sub>max</sub>	0.77 Inch
YHOLD	YHOLD	0.4 Inch
RHOLD	RHOLD <sub>max</sub>	0.77 Inch

**TABLE SAS 1 - Gains for Pitch SCAS Electronics**

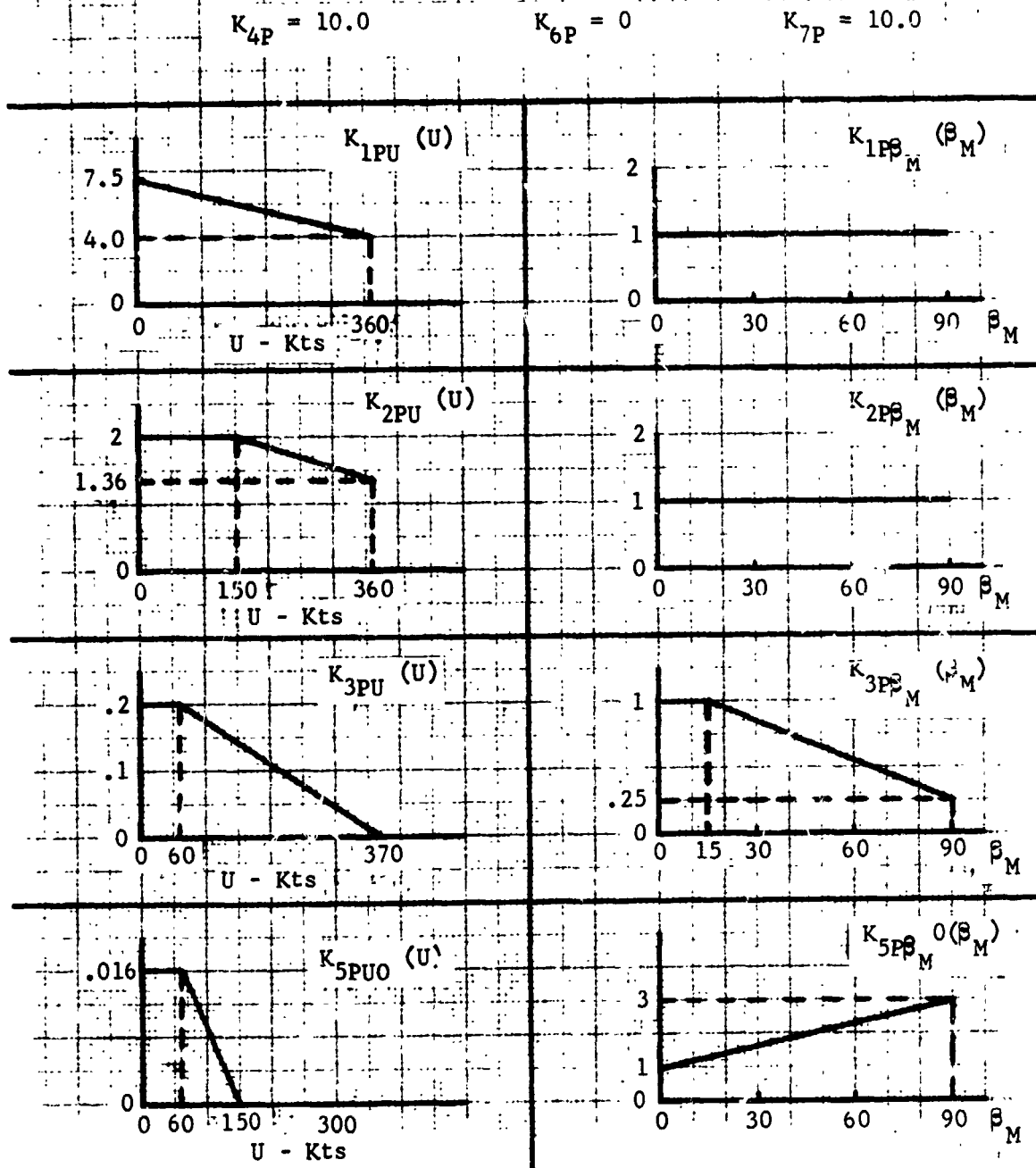


TABLE SAS I (Contd)

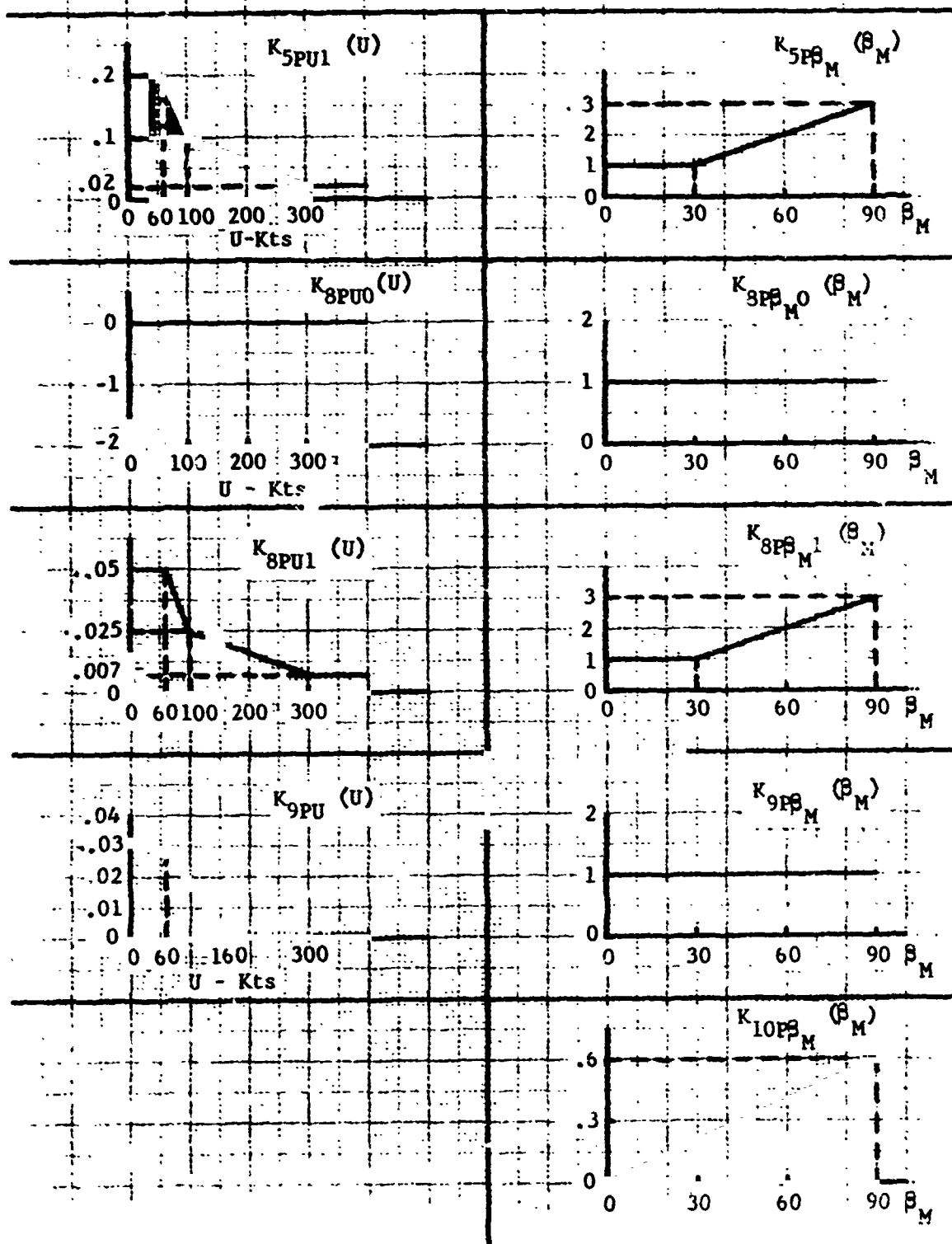


TABLE SAS II - Gains for Yaw SCAS Electronics

$K_{4Y} = 10.0$

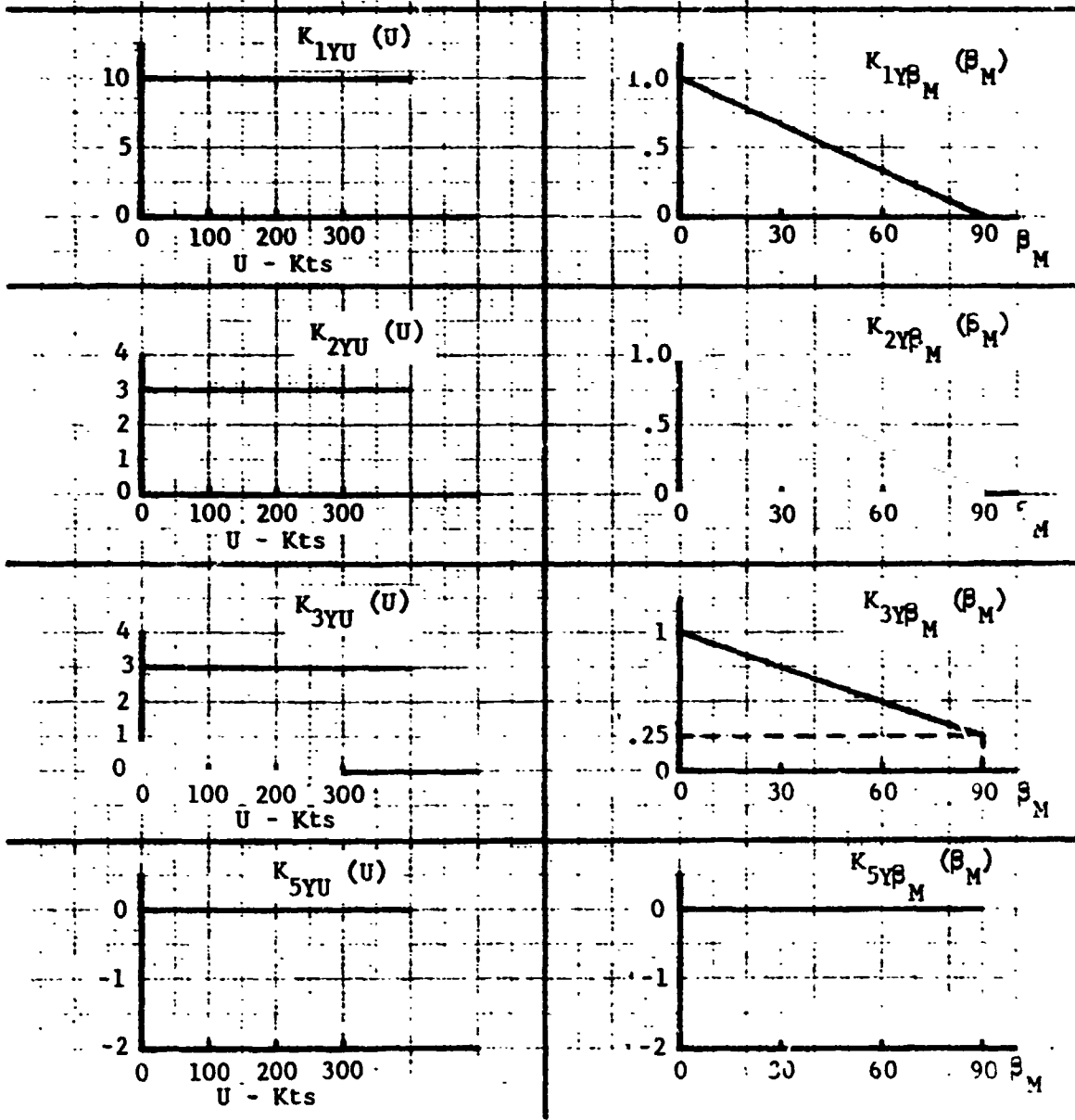
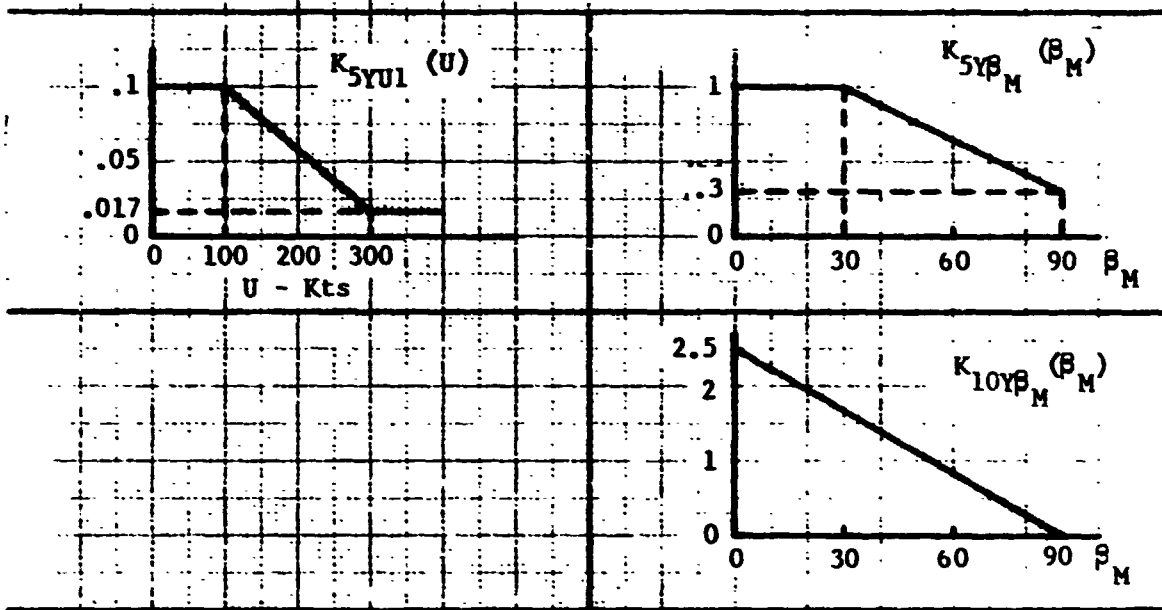


TABLE SAS II - Contd



**TABLE SAS III - Gains for Roll SCAS Electronics**

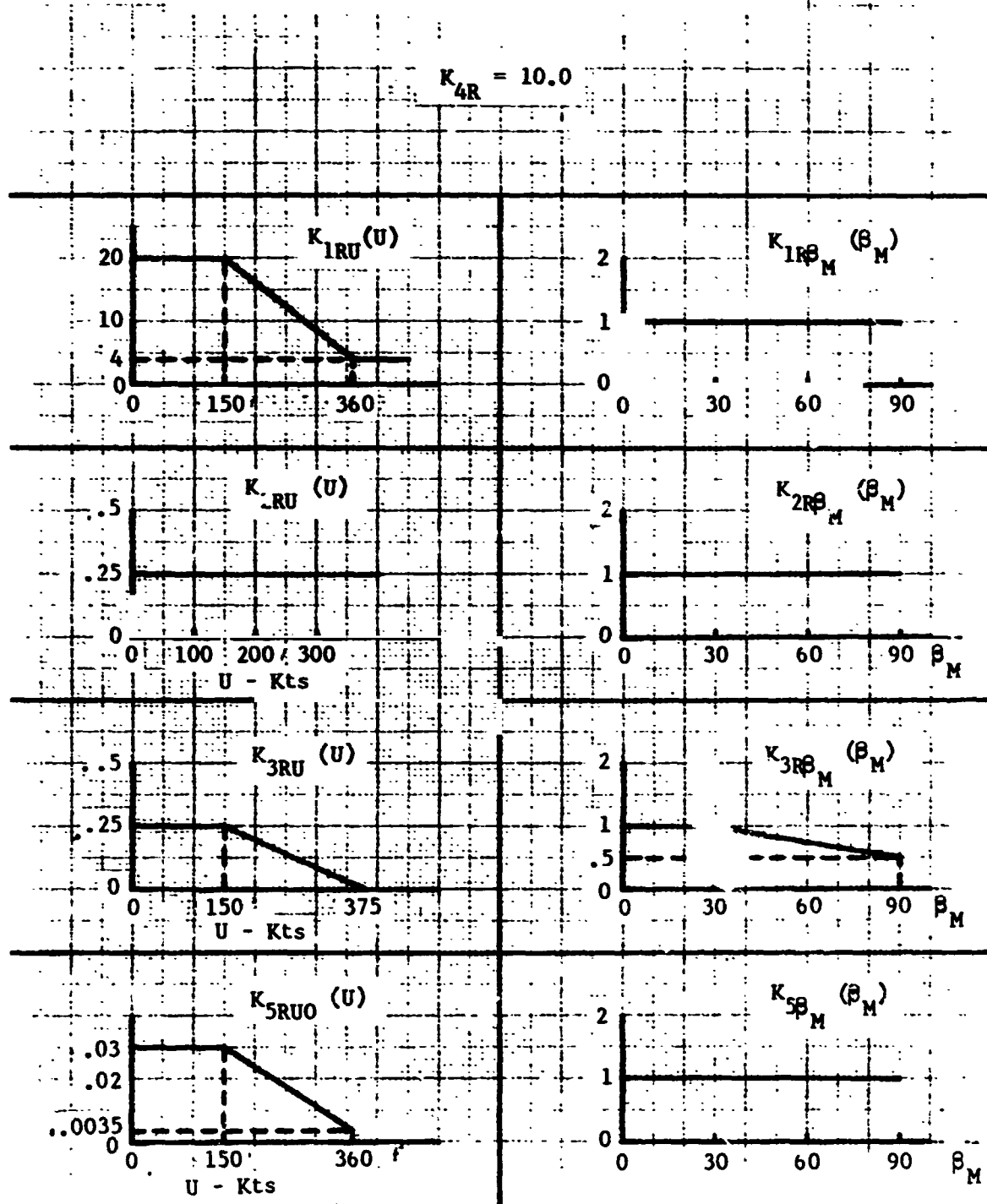


TABLE SAS III - Contd

