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# SIMULATOR MODEL SPECIFICATION FOR THE AUGMENTOR WING JET STOL RESEARCH AIRCRAFT

by

P. C. Rumsey and R. E. Spitzer

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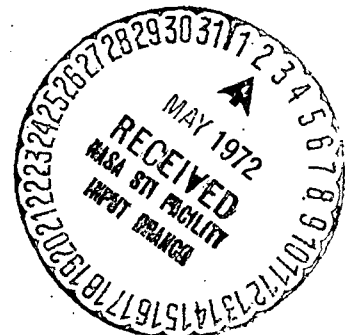
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# N O T I C E

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$W_{x,y,z}$  Steady winds in earth axes, ft/sec

$W_{LCG}$  Water line position of the c.g., ins

$V_A$  Total airplane velocity, ft/sec, true airspeed

$X_i$  Force in X-direction, lbs

$Y_i$  Force in Y-direction, lbs

$Z_i$  Force in Z-direction, lbs

where  $i$  = STAB, Aero forces in stability axes

$i$  = A, Aero forces in stability axes

$i$  = LG, gear forces in body axes

$i$  = ENG, Engine forces in body axes

$i$  = B, Total forces in body axes

$i$  = E, total forces in earth axes

$Y_{CP}$  Effective distance of center of pressure of asymmetric lift from body centerline, ft.

$Y_E$  Distance of engine centerline from body centerline, ft

$Z_V$  Moment arm from vertical tail 1/4 chord to wing 1/4 chord, in body axes, ft

$\alpha, \alpha_{wing}$  Wing angle of attack, degs.

$\alpha_F = \alpha_{wing} - 2.5^\circ$ , fuselage angle of attack, degs

$\alpha_{ot}$  Zero lift angle of attack of the horizontal tail

$\Delta\alpha_Q = \frac{l_T}{V_A} \cdot Q_s$ , Angle of attack induced at the tail by a pitch rate, rads

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# GENERAL NOTATION

Most symbols used in the text are defined there. This list includes those that are not plus those in general use.

$C_{L\alpha_t}$  Linearized lift curve slope of the horizontal tail, per degree

BSCG Body station of the c.g., inches

BW Wing span, ft.

CW Wing reference chord length, ft.

$C_i = \frac{i}{\bar{Q} \cdot S_w}$ , where  $i = L, Y, D$ , Aerodynamic FORCE coefficients in STABILITY axes

$C_m = \frac{M}{\bar{Q} \cdot S_w \cdot CW}$ , Aerodynamic pitching moment coefficient

$C_i = \frac{i}{\bar{Q} \cdot S_w \cdot BW}$ , where  $i = l, n$ , Aerodynamic MOMENT coefficients in STABILITY axes

$C_{i_{wb}}$   $i = L, M, D$  Aerodynamic force and moment coefficients due to airplane wing and body alone

$C_{i_{GE}}$   $i = L, M, D$  Ground effect on wing-body aerodynamic coefficients

$C_{i_{\hat{P}}} = \frac{\partial C_i}{\partial \hat{P}}$ , where  $i = l, n, Y$ ;  $\hat{P} = \frac{P_s \cdot BW}{2 \cdot V_A}$

$C_{i_{\hat{r}}} = \frac{\partial C_i}{\partial \hat{r}}$ , where  $i = l, n, Y$ ;  $\hat{r} = \frac{R_s \cdot BW}{2 \cdot V_A}$

$C_{i_{\beta}} = \frac{\partial C_i}{\partial \beta}$ , where  $i = l, n, Y$

$C_{i_{\hat{\beta}}} = \frac{\partial C_i}{\partial \hat{\beta}}$ , where  $i = l, n, Y$ ;  $\hat{\beta} = \frac{\beta \cdot BW}{2 \cdot V_A}$

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$$C_{i\hat{\alpha}} = \frac{\partial C_i}{\partial \hat{\alpha}}, \text{ where } i = L, M; \hat{\alpha} = \frac{\alpha \cdot CW}{2 \cdot V_A}$$

$$C_{i\hat{Q}} = \frac{\partial C_i}{\partial \hat{Q}}, \text{ where } i = L, M; \hat{Q} = \frac{QS \cdot CW}{2 \cdot V_A}$$

$$C_J \quad \text{Momentum blowing coefficient} = \frac{\text{Nozzle Thrust}}{\bar{Q} \cdot S_w}$$

$D_{RAM}$  Ram drag, lbs

$F_i$  Forces on pilot's controls, lbs, where  $i = S, W, P$  for stick, wheel and pedals respectively ( $F_S$  positive for aft column,  $F_W$  positive for right wheel,  $F_P$  positive for right pedal).

$i_T$  Stabilizer incidence angle, degs

$I_{ij}$  Moments of inertia in body axes, slug-ft<sup>2</sup>, where  $i, j = X, Y, Z$

$l_T$  Body axes moment arm from horizontal tail 1/4 chord to wing 1/4 chord

$l_V$  Body axes moment arm from vertical tail 1/4 chord to wing 1/4 chord

$L_i$  Aerodynamic rolling moment, ft lbs

$M_i$  Aerodynamic pitching moment, ft lbs

$N_i$  Aerodynamic yawing moment, ft lbs

} where  $i = S$ , stability axes  
 $i = A$ , body axes

$L, M, N$

Roll, pitch and yaw moments, where  $\begin{cases} i = B, \text{ in body axes including gear and engine} \\ i = E, \text{ engine moment only in body axes} \end{cases}$

$m$  Airplane Mass, slugs

$M_{RAM}$  Ram drag pitching moment, ft lbs

$\dot{m}$  Mass flow into engines, lb wt/sec

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|             |                                                                |                                                                                                                                                           |
|-------------|----------------------------------------------------------------|-----------------------------------------------------------------------------------------------------------------------------------------------------------|
| RPM         | Engine speed, revolutions per minute                           |                                                                                                                                                           |
| $P_i$       | Roll rate, rads/sec                                            | } where $i = S$ , stability axes<br>$i = B$ , body axes<br>$i = BN$ , due to random gusts in body axes<br>$i = BT$ , total gust and airplane in body axes |
| $Q_i$       | Pitch rate, rads/sec                                           |                                                                                                                                                           |
| $R_i$       | Yaw rate, rads/sec                                             |                                                                                                                                                           |
| $\dot{P}_i$ | Roll acceleration, rads/sec <sup>2</sup>                       | } where $i = B$ , body axes<br>$i = DG$ , accelerations due to special gust model, body axes                                                              |
| $\dot{Q}_i$ | Pitch acceleration, rads/sec <sup>2</sup>                      |                                                                                                                                                           |
| $\dot{R}_i$ | Yaw acceleration, rads/sec <sup>2</sup>                        |                                                                                                                                                           |
| $\bar{Q}$   | Dynamic pressure, lb/ft <sup>2</sup>                           |                                                                                                                                                           |
| $S_T$       | Reference area, Horizontal tail, ft <sup>2</sup>               |                                                                                                                                                           |
| $S_V$       | Reference area, Vertical tail, ft <sup>2</sup>                 |                                                                                                                                                           |
| $S_W$       | Reference area, Wing tail, ft <sup>2</sup>                     |                                                                                                                                                           |
| $U_i$       | Longitudinal velocity, ft/sec                                  | } where $i = B$ , body axes<br>$i = BN$ , due to random gust, body axes<br>$i = E$ , earth axes                                                           |
| $V_i$       | Transverse velocity, ft/sec                                    |                                                                                                                                                           |
| $W_i$       | Downward velocity, ft/sec                                      |                                                                                                                                                           |
| $\dot{U}_E$ | Longitudinal acceleration, ft/sec <sup>2</sup> , in earth axes |                                                                                                                                                           |
| $\dot{V}_E$ | Transverse acceleration, ft/sec <sup>2</sup> , in earth axes   |                                                                                                                                                           |
| $\dot{W}_E$ | Downward acceleration, ft/sec <sup>2</sup> in earth axes       |                                                                                                                                                           |

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|                 |                                                                                                                 |
|-----------------|-----------------------------------------------------------------------------------------------------------------|
| $\beta$         | Angle of sideslip, degs                                                                                         |
| $\delta_p$      | Pedal deflection, inches                                                                                        |
| $\delta_w$      | Wheel deflection, degs                                                                                          |
| $\delta_c, x_s$ | Stick deflection (positive aft), $\delta_c$ -deg, $x_s$ -inches                                                 |
| $\delta_{FL}$   | Flap lever deflection, degs                                                                                     |
| $\delta_N$      | Nozzle vector lever deflection, degs                                                                            |
| $\delta_{TH}$   | Throttle lever deflection, degs                                                                                 |
| $\delta_F$      | Flap deflection, degs                                                                                           |
| $\delta_e$      | Elevator angle, degrees, positive trailing edge down                                                            |
| $\delta_a$      | Aileron deflection, degrees, positive trailing edge down                                                        |
| $\delta_R$      | Rudder deflection, degrees, positive trailing edge left                                                         |
| $\delta_{SP}$   | Spoiler deflection, degs, positive trailing edge up                                                             |
| $\delta_{CH}$   | Choke deflection, positive when closing                                                                         |
| $E$             | Downwash angle at the horizontal tail, degs                                                                     |
| $E^*$           | Downwash angle at the horizontal tail, degs at time $t = \Delta t$<br>where $\Delta t = \frac{l_T}{V_R}$ , secs |
| $\nu$           | Hot gas nozzle deflection, downwards from the fuselage reference line                                           |

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$\phi$  Euler roll angle

$\theta$  Euler pitch angle

$\psi$  Euler heading angle

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## 1.0 INTRODUCTION

NASA in conjunction with deHavilland of Canada has been developing the augmentor wing jet flap for a number of years. The work has included wind tunnel tests, preliminary configuration studies, and piloted simulator studies. In July, 1970 The Boeing Company began work under NASA Contract NAS2-6025 to modify a deHavilland C-6A "Buffalo" airplane into an augmentor wing, jet-flap STOL research vehicle. Figure 1-1 shows a 3-view drawing of the Modified C-6A. Augmentor-wing jet flaps, blown and drooped ailerons and leading-edge slats have been added. Wing span has been shortened to increase wing loading. Two Rolls Royce Spey 801PF jet engines provide blowing air to the flaps and ailerons via ducts. Engine hot thrust is directed through vectorable Pegasus nozzles. Further descriptive information may be found in Reference 1\*.

The total simulator model includes a number of component parts tied together to provide as an end product realistic visual, aural, tactile, vestibular and kinesthetic cues for the pilot to assess the predicted behavior of the real airplane. To produce these cues requires the following models to be accurately programmed on the simulator:

1. Pilot's controls and control system characteristics
2. Aerodynamic forces and moments
3. Propulsive forces and moments
4. Airplane mass and inertia characteristics
5. Equations of rigid body motion in the earth's gravitational field, including atmospheric disturbances
6. Accelerations felt at the pilot's station
7. Outside world visual presentation to the pilot

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8. Pilot's instruments and cab environment
  9. Stability augmentation system
  10. Readout to engineering staff for analysis and check-out

Each of these modules is dealt with in detail below. A flow chart of the total simulation is shown in Figure 1-2, the shaded blocks representing those parts of the model which are changed from the existing NASA model.

The model, as described in this specification will be used to investigate design requirements for flight control systems and stability augmentation systems, criteria for structural design and operational characteristics of the airplane. For these investigations a number of the parameters of the model will require to be variables. These parameters are identified by a double asterisk (\*\*) and their nominal values are given in the text.

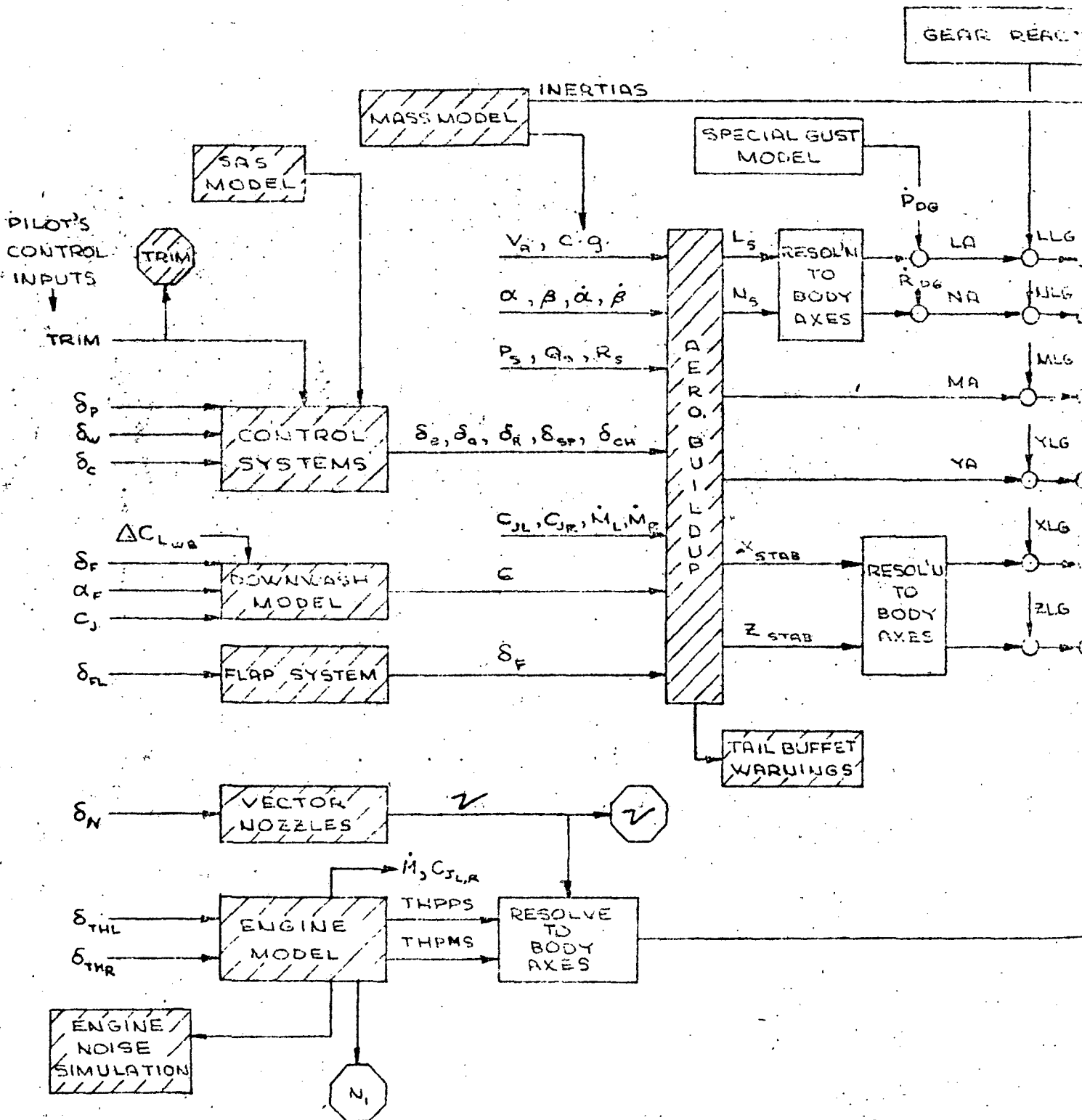
Appendix A to this specification is an outline of an expanded simulation of the hydraulic system and the control actuators. This simulation would be necessary to fully investigate the effects of rate limiting and hydraulic supply capability on airplane handling qualities.

Longitudinal control improvements are envisioned for the Modified C-8A.

Simulation models for powered longitudinal control, longitudinal stability augmentation and an automatic airspeed control system have been described in Boeing Document D6-26057.

- \*Reference 1. SAE Paper No. 710757, "A Progress Report on the Development of an Augmentor Wing Jet STOL Research Aircraft", H. C. Quigley, S. R. M. Sinclair, T. C. Nark, Jr., J. V. O'Keefe, September 1971.





|       |        |         |         |      |                                                                     |           |
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| CHECK |        |         |         |      |                                                                     | FIG1-2    |
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SHADED BLOCKS INDICATE CHANGES FROM PREVIOUS SIMULATION. BY NASA/DA.

## 2.0 PILOT'S CONTROLS AND CONTROL SYSTEM CHARACTERISTICS

### 2.1 Stick Control

Longitudinal control is provided by the fully-manual, spring-tab elevator system. Stick force is reacted by a combination of elevator and tab hinge moments, which vary with dynamic pressure. Since the stabilizer is fixed, hands-off trim is produced by moving the electrically-actuated trim tab to hold the elevator at its proper position.

The original NASA/DHC elevator control system simulation model (Reference 2) has been modified to include the reduction in stick force associated with reduced spring tab follow-up ratio and elevator torque stiffness. The revised simulation includes movement of the column due to trim inputs. Figure 2-1 presents the longitudinal control and trim simulation flow chart.

Basic equations and terms follow:

#### ● Elevator Angle

$$\delta e = \left( \frac{\delta e}{x'_s} \right) \cdot x'_s + \left( \frac{\delta e}{n_2 - 1} \right) \cdot (n_2 - 1) + \left( \frac{\delta e}{\dot{q}_B} \right) \cdot \dot{q}_B \quad \sim \text{DEG}$$

where

$$\text{a) } \left( \frac{\delta e}{x'_s} \right) = \left\{ \frac{-4.65}{1 + \left( \frac{1 + .039 \bar{Q}}{1 + 1.98 \bar{Q}} \right) (.495 \bar{Q})} \right\} \quad \sim \frac{\text{DEG}}{\text{INCH}}$$

b)

$$x'_s = x_s - .013 F_s \quad \sim \text{INCHES}$$

c)

$$\left( \frac{\delta e}{n_2 - 1} \right) = \left\{ \frac{-64.}{2.24 \bar{Q} + \left( \frac{4.5 + 8.9 \bar{Q}}{1 + .039 \bar{Q}} \right)} \right\} \quad \sim \frac{\text{DEG}}{g}$$

$$d) \left( \frac{\delta e}{\dot{q}_8} \right) = \left\{ \frac{+35.5}{2.24 \bar{Q} + \left( \frac{4.5 + 8.9 \bar{Q}}{1 + .039 \bar{Q}} \right)} \right\} \sim \frac{\text{DEG}}{\text{RAD/SEC}^2}$$

NOTE: The elevator angle is mechanically limited to  $-25^\circ < \delta_e < +15^\circ$

● Static Stick Force, Including Effect of Trim

$$F_s = \frac{F_s}{\delta_e} (\delta_e - \delta_{e_{\text{TRIM}}}) \sim \text{LBS}$$

where

$$a) \frac{F_s}{\delta_e} = -(.815 \bar{Q}) \left( \frac{1 + .195 \bar{Q}}{1 + 1.98 \bar{Q}} \right) \sim \text{LB/DEG}$$

$$b) \delta_{e_{\text{TRIM}}} = \left( \frac{\delta_e}{\delta_{tt}} \right) \cdot \delta_{tt} + \delta_{e_{\text{I.C. TRIM}}} \sim \text{DEG}$$

$$c) \left( \frac{\delta_e}{\delta_{tt}} \right) = -1.30 \text{ DEG/DEG}$$

$$d) \delta_{tt} = \text{TRIM TAB ANGLE} \sim \text{DEG}$$

$$= \left[ \frac{\dot{\delta}_{tt}}{S} + \frac{\delta_{tt}}{\delta_F} (\delta_F - \delta_{F_{\text{I.C. TRIM}}}) \right]$$

$$e) \dot{\delta}_{tt} = \pm 2.0 \text{ DEG/SEC (PILOT'S TRIM RATE)}$$

$$f) \frac{\delta_{tt}}{\delta_F} = -.0435 \text{ DEG/DEG (FLAP-TRIM INTERCONNECT PROGRAM)}$$

$$g) \delta_{e_{\text{I.C. TRIM}}} \neq \delta_{F_{\text{I.C. TRIM}}} = \text{INITIAL CONDITION TRIM VALUES} \sim \text{DEG}$$

NOTE: Optional trim tab authority limits (Figure 2-2) may be included in the simulation with minor programming change. Also, stick force gradient increases markedly above torque tube wind up limits ( $\theta_1$ ) where the pilot would react against direct elevator hinge moments. This wind up limit occurs at  $(\delta_e - \delta_{e_{\text{TRIM}}}) = -22$  degs at 80 kts decreasing to  $(\delta_e - \delta_{e_{\text{TRIM}}}) = -7.5$  deg at 180 kts. Such elevator deflections are beyond those needed for normal maneuvering, including pull ups

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to limit load factor ( $n_z = 2.25g$ ). Therefore, double stick force gradient is not simulated. A third point: The simulator cab feel system has an inherent, built-in stick force gradient which is cancelled by subtracting the same gradient in the computer.

### ④ Control Column Displacement

$$X_S = \frac{F_S \left\{ 1 + .013 \left( \frac{F_S}{\delta_e} \cdot \frac{\delta_e}{X_S} \right) \right\} + \left( \frac{F_S}{\delta_e} \right) \delta_{e_{TRIM}} + \frac{\delta_e}{(n_z - 1)} (n_z - 1) + \left( \frac{\delta_e}{\dot{q}_S} \right) \dot{q}_S}{\frac{I_c}{L} \left[ S^2 + \left( \frac{\text{VISCOUS DAMPING}}{L/I_c} \right) S + \frac{L}{I_c} \left( \frac{F_S}{\delta_e} \cdot \frac{\delta_e}{X_S} \right) \right]} \sim \text{INCHES}$$

where

a) TRIMMED ( $F_S = 0$ ) STICK POSITION,  $X_{S_{TRIM}} = \frac{\delta_{e_{TRIM}}}{(\delta_e/X_S)} \sim \text{INCHES}$

b) STEADY-STATE STICK POSITION AT UNIT LOAD FACTOR

$$X_S = \frac{F_S \left\{ 1 + .013 \left( \frac{F_S}{\delta_e} \cdot \frac{\delta_e}{X_S} \right) \right\} + \left( \frac{F_S}{\delta_e} \right) \delta_{e_{TRIM}}}{\left( \frac{F_S}{\delta_e} \cdot \frac{\delta_e}{X_S} \right)} \sim \text{INCHES}$$

c) Column dynamics should be set by adjusting the value of  $\frac{L}{I_c}$  and "Viscous Damping" to give a good representation of the Buffalo control system as judged by the pilots who have flown the Buffalo. Elevator dynamics have not been included in the simulation for two reasons: (1) elevator natural frequency is on the order of 10 RAD/SEC, which is "10 times" the short period and nearing the digital computation limits, and (2) cab column dynamics interact with simulated elevator dynamics to give "unrealistic" transients.

d) The neutral column position ( $\delta_e = 0^\circ$ ) is 2.8 inches ( $5^\circ \delta_c$ ) forward of the vertical. Total forward column travel is  $X_S = -4.3$  inches ( $-7.6^\circ$ ) ahead of this neutral position, and maximum aft travel is  $X_S = +7.5$  inches ( $+13.2^\circ$ ) from neutral. Column radius is 32.55 inches.

AD 18460

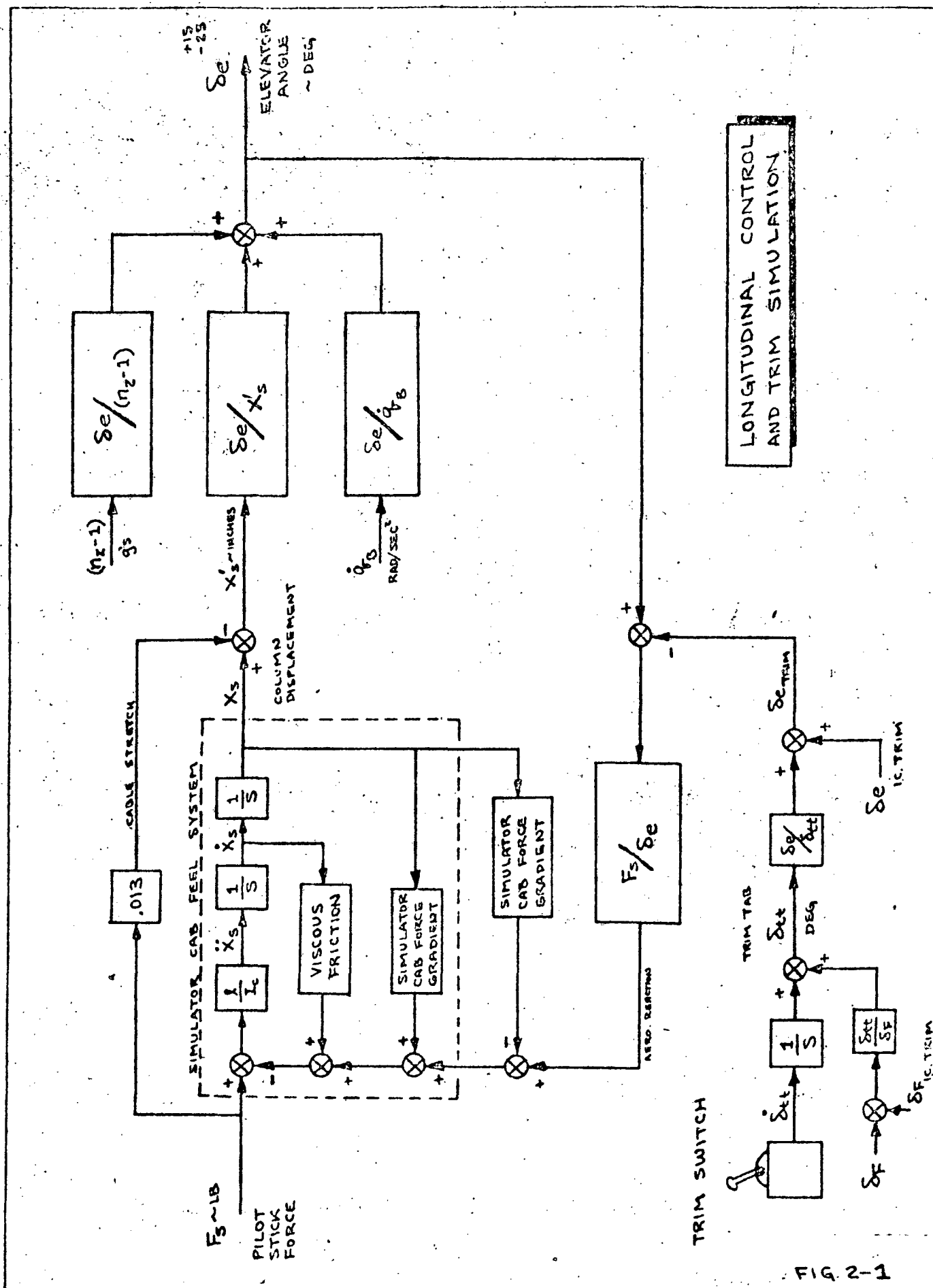
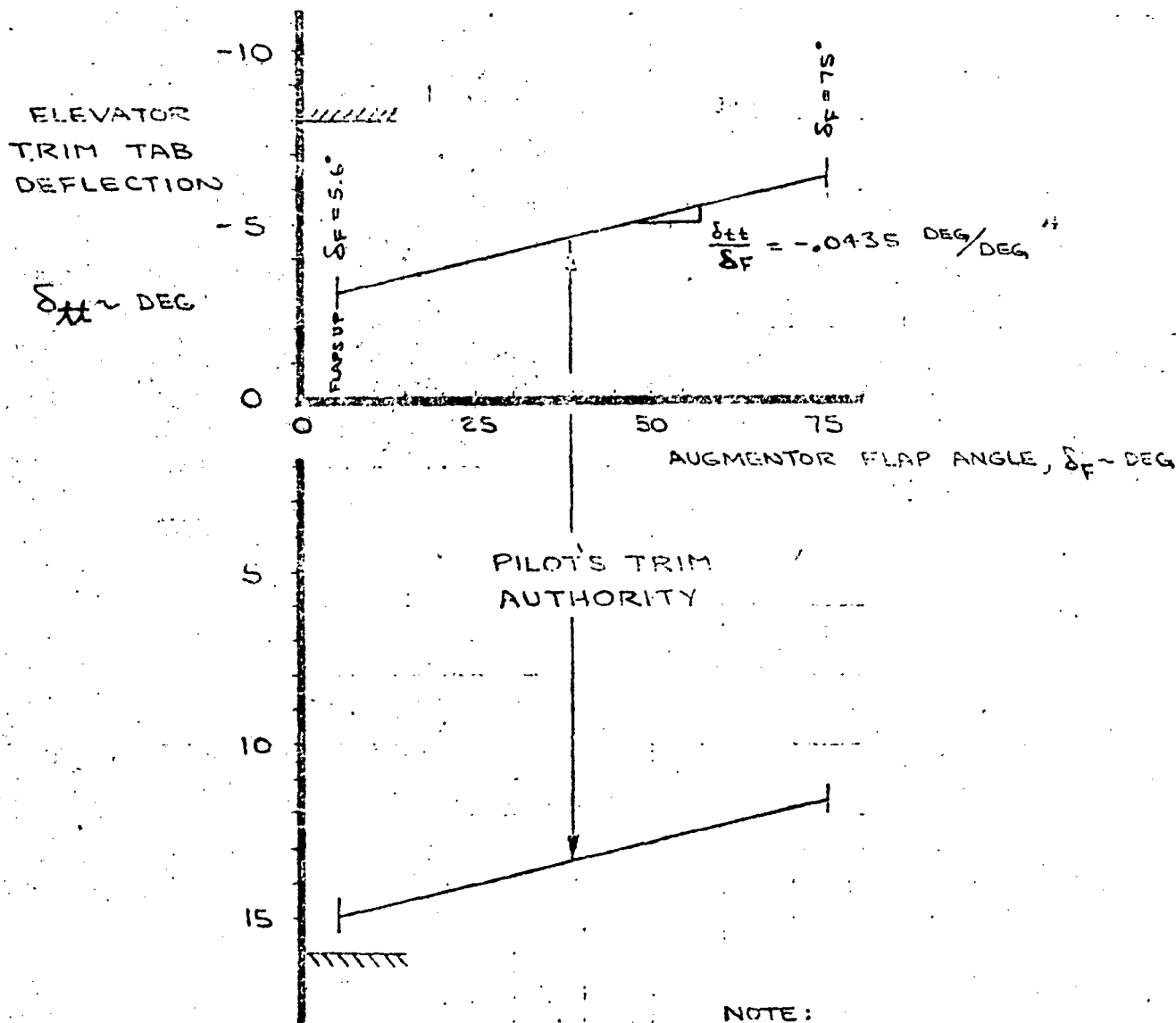


FIG. 2-1

ELEVATOR TRIM TABS  
AUTHORITY WITH  
FLAP INTERCONNECT

TAB T.E. UP  
ELEVATOR T.E. DOWN



NOTE:

- TAB RIGGED AT  $\delta_x = +1^\circ$   
AT NEUTRAL ELEVATOR
- TOLERANCE  $\delta_x = \pm 1^\circ$   
IN RIGGING AT AUTHORITY  
LIMITS

MOD C-BA

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FLAP-TRIM INTERCONNECT

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DB-2606STN

FIG 2-2

PAGE

9.

## 2.2 Wheel Control

The lateral axis flight control system can be operated in a fully powered mode or in a reversion mode giving direct manual control over the ailerons. There are therefore two sets of equations governing wheel forces,  $F_w$ , and wheel displacement,  $\delta w$ , as a function of aileron deflection,  $\delta a$ , spoiler deflection,  $\delta sp$ , and augmentor choking,  $\delta ch$ . The flow chart on Figure 2-3 outlines the two models and the logic for selecting which is to be used.

Power - represents the condition with at least one of the two hydraulic systems operating.

HYDA on, HYD B on are self explanatory. It is expected that some simulation work will be done with simulated single hydraulic failures.

Manual - represents the condition with all hydraulic power off

The lateral control system produces a linear rolling moment characteristic using electrical compensation via the lateral SAS circuitry. Figure 2-4 shows this network. The electrical path must be added to the mechanical path (Figure 2-3) to simulate the Modified C-8A lateral control characteristics.

AD 1546 D

1. With Power On the equations for the system are as follows:

In a static condition:

$$\text{Wheel Force, } F_{wtot} = \pm FRW + F_W \quad \text{lbs.}$$

\*\* where  $FRW = \pm 1.5$  lbs, friction force

\*\* and  $F_W$  is a function of  $\delta_W$  shown on Figure 2-5

(Note that positive wheel forces produce positive deflections of the wheel, the right wing spoilers and chokes, and the left wing aileron.)

Dynamically:

$$\text{Wheel deflection, } \delta_W = \frac{\delta_{WDEM}}{\left(\frac{S}{\omega_W}\right)^2 + 2 \frac{\delta_W}{\omega_W} S + 1}, \text{ degrees.}$$

where  $\omega_W$  and  $\delta_W$  are set to give rapid wheel centering with no overshoot.

The maximum wheel deflection is  $\pm 75^\circ$  over a radius of 7.5 inches. The demanded wheel deflection is the sum of the pilot's input and the trim, where

$$\delta_{WTRIM} = \frac{\delta_{WTRIM}}{S}, \text{ degrees}$$

\*\* The maximum trim authority is  $\pm 62^\circ$ , and the trim rate is

\*\*  $\pm 4.2^\circ/\text{sec.}$

The wheel signals the central actuator to move, and its deflection  $\delta_W$ , moves the aileron via the wing cable system, and the spoilers and chokes via their respective actuators. Thus

$$\delta_{W1} = \delta_W + \frac{\delta_W S_{AS}}{(1 + T_{\delta a S_{AS}} \cdot S)} \quad \text{degrees}$$

$$\delta_{W11} = \frac{\delta_{W1}}{(1 + T_{\delta W} \cdot S)} \quad \text{degrees}$$

AD 1546 D

where  $T_{\delta_{aSAS}} = 1/35$  secs  
 $T_{\delta_{aI}} = 1/30$  secs

\*\*  $\delta_{W_{SAS_{MAX}}} = +20$  degs.  
 \*\*  $\dot{\delta}_{W_{SAS_{MAX}}} = 48^\circ/\text{sec}$   
 \*\*  $\ddot{\delta}_{W_{MAX}} = 220^\circ/\text{sec}$

The wing cables stretch under the aileron hinge moments, thus

$$\delta_W''' = \delta_W - \left\{ \frac{\bar{q} \cdot S_a \bar{C}_a \cdot C_{H\delta_a} \cdot \left( \frac{\Delta \delta_{aI} - 4\delta_a}{2} \right)}{G_{\text{cable}}} \right\} \text{ degrees}$$

where  $C_{H\delta_a} = -.0047$  per degree

$S_a \bar{C}_a = 94 \text{ ft}^3$

and  $G_{\text{cable}} = 50 \text{ ft lb/degree}$

The differential aileron deflection about the droop angle is then given by

$$\Delta \delta_{aL,R} = G_{\delta_a} (\delta_W''') \text{ as shown on Figure 2-6}$$

\*\* The maximum differential deflection is limited to  $\pm 17.5^\circ$  for the nominal surface programming and  $\pm 20^\circ$  for the optional program.

\*\* The symmetrical droop deflection,  $\delta_{a\text{droop}}$  is tied to flap angle, as shown on Figure 2-7. Again two surface programs are possible.

The total aileron deflection is then

$$\delta_a = \delta_{a\text{droop}} + \Delta \delta_a \text{ degrees}$$

The spoiler deflection is given by

$$\delta_{SP_{L,R}} = \frac{G_{\delta SP} (\delta W'')}{(1 + T_{SP} \cdot S)} \quad \text{degrees}$$

Hyd A on

- \*\* where  $G_{\delta SP}$  is given on Fig. 2-8.  
and  $T_{SP} = 1/20$  secs.

- \*\* The spoiler rate limits are  $\pm 500^\circ/\text{sec}$  and deflections are limited to  $-0^\circ, +50^\circ$ .

The augmentor choking is given by

$$\delta_{CH_{L,R}} = \frac{G_{\delta CH} (\delta W'')}{(1 + T_{CH} \cdot S)} \quad \text{Hyd B on}$$

- \*\* where  $G_{\delta CH}$  is given on Figure 2-9.  
and  $T_{CH} = 1/15$  secs.

- \*\* The choke rate limits are  $\pm 5.0$  per sec, Maximum choke deflection  
\*\* (in percent of full closure) is also shown on Figure 2-9. Two lateral control options are available.

2. In manual reversion mode the equations for the system are as follows:

In a static condition the wheel forces are dependent on  $\delta_W$ ,  $\delta_W''$ , and  $\delta_a$ . For small initial movements of the wheel the only forces felt by the pilot are those due to system friction,  $F_{RW}$ , and the feel spring,  $K_W$ . The friction forces downstream of the actuator,  $F_{TA}$ , and the aerodynamic hinge moments,  $C_{H\delta a}$ , are separated from the pilot until the deadspace in the forward path (due to the inlet valve stroke on the control actuator) is overcome, see Figure 2-10. After this,

$$F_{Wtot} = \pm (F_{RW} + F_{TA}) + F_W + G_{\delta W} \cdot \bar{Q} \cdot (S a \bar{c}_a) \cdot C_{H\delta a} \cdot \frac{(\Delta \delta_{aR} - \Delta \delta_{aL})}{2}$$

where

\*\*  $F_{TA} = \pm 20 \text{ lbs. of friction}$

and

$$G_{\delta W} = .32 \text{ feet}^{-1}$$

Figure 2-10 indicates how wheel force and wheel deflection vary with further displacement. Note that during return to neutral the wheel picks up the dead space again when the aerodynamic hinge moments just balance the friction forces,  $F_{TA}$ , (or when total  $F_{Wtot} = F_W - F_{RW}$ ). Full and complete simulation of this cycle is not necessary; however, the essential features are the dead space in the forward path and the large increase in friction forces.

AD 1546 D

The aileron deflection is given by the equations

$$** \quad \Delta \delta_{a_{L,R}} = G_{\delta_a} (\delta_W''') \quad \text{Figure 2-6}$$

$$\text{where } \delta_W''' = \delta_W'' - \left\{ \frac{\bar{q} \cdot (S a \bar{c}_a) \cdot C_{ll} \delta_a \cdot \left( \frac{\Delta \delta_{a_R} - \Delta \delta_{a_L}}{2} \right)}{G_{\text{cable}_2}} \right\}$$

and

$$** \quad \delta_W'' = \delta_W - 9.0 \text{ degrees}$$

Due to the larger length of cable carrying the aileron loads

$$G_{\text{cable}_2} = 33.3 \text{ ft lb/degree}$$

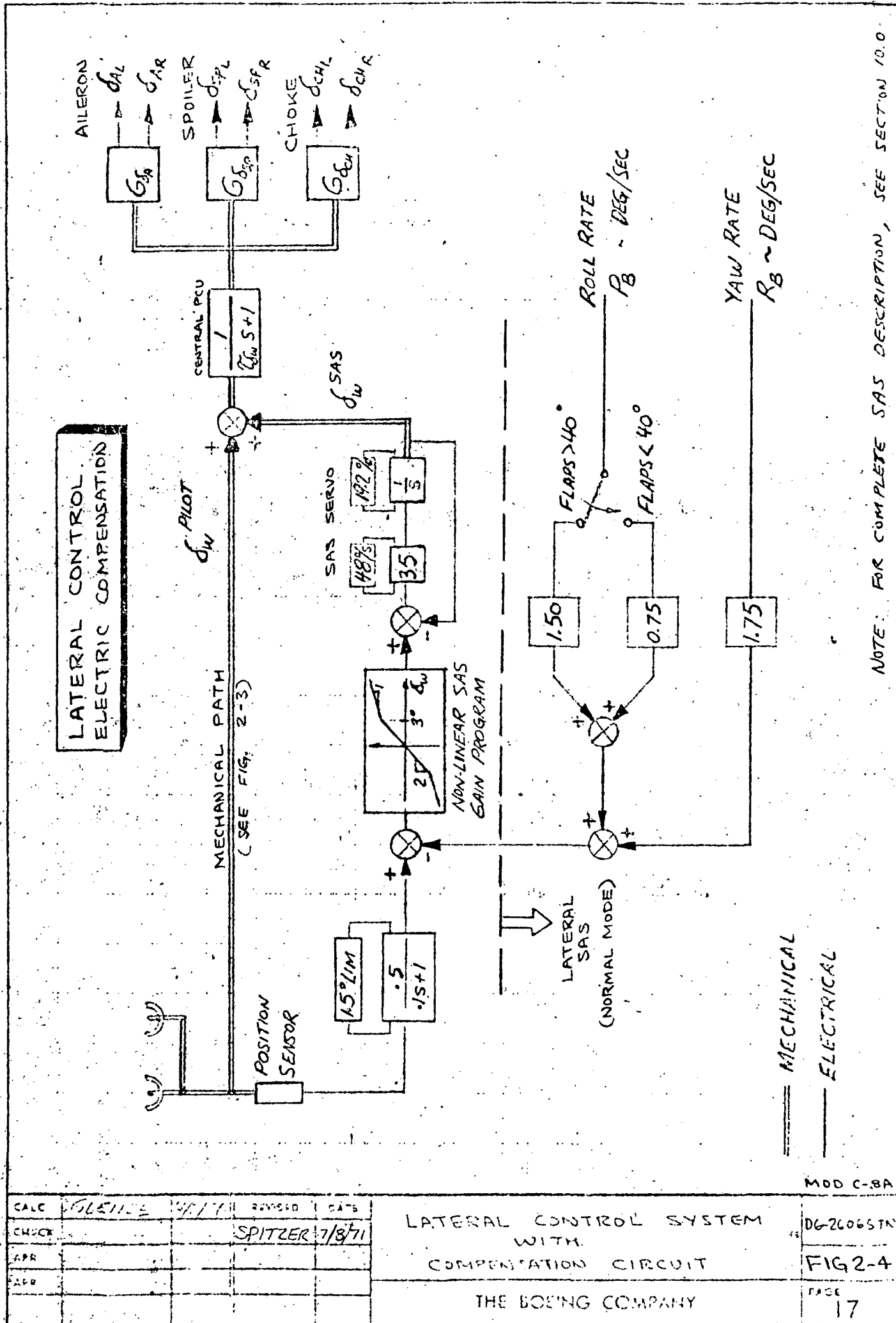
(NOTE that no spoiler or choke deflections occur with power off)

Dynamically, the viscous friction in the system is increased due to the idling actuator, and the total inertia felt at the wheel now includes the control surface inertia. For simplicity in the simulation, these effects can be simulated by adjusting the values of frequency and damping.

In manual it is assumed that the friction forces are so high that no wheel trim is available to the pilot, and no SAS inputs are felt at the aileron.

AD 1548D





NOTE: FOR COMPLETE SAS DESCRIPTION, SEE SECTION 10.0

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| CHKD |       |        | SPITZER 7/8/71 |      |
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LATERAL CONTROL SYSTEM WITH COMPENSATION CIRCUIT

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FIG 2-4  
PAGE 17

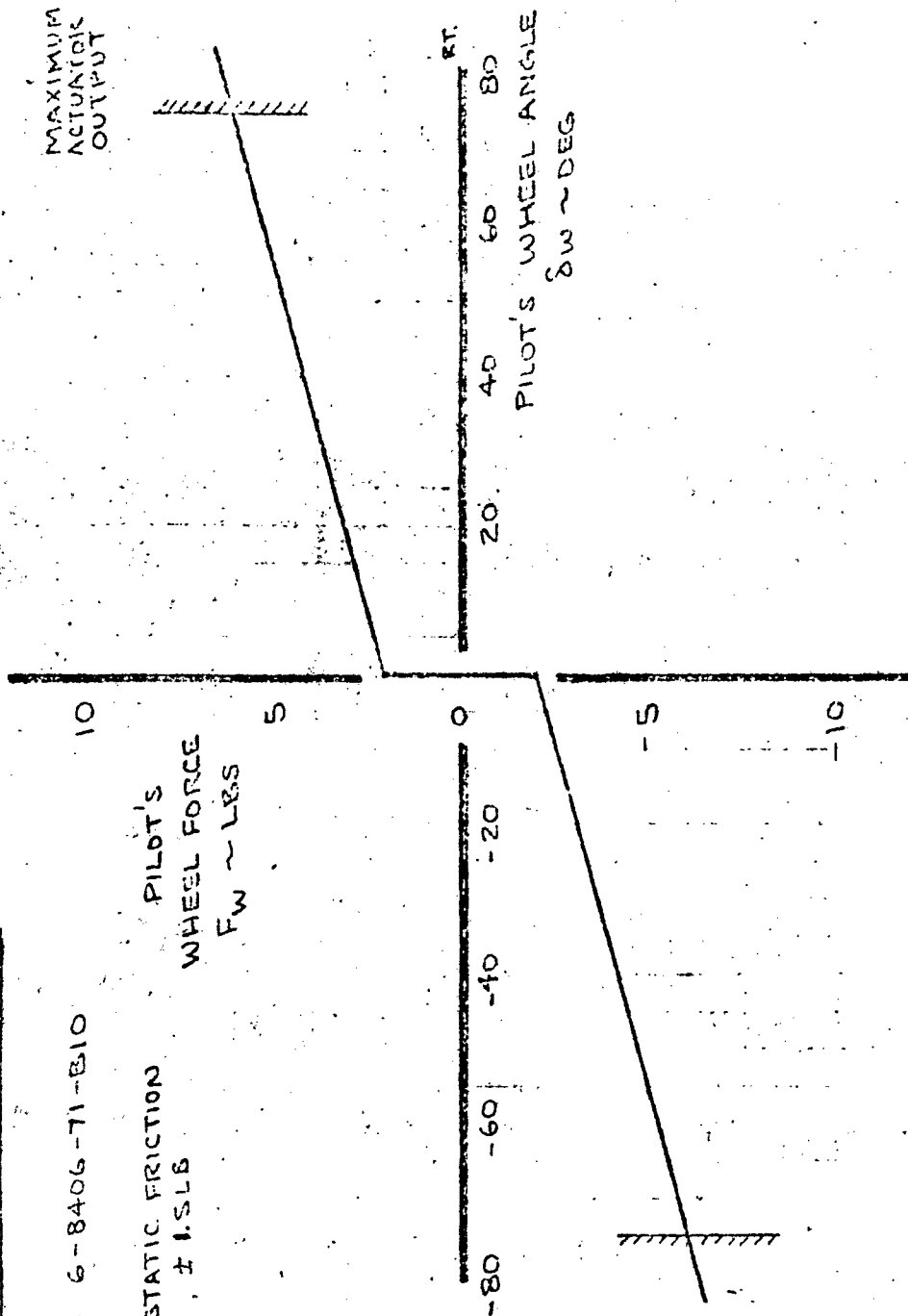
LATERAL CONTROL  
FEEL & CENTERING

REF: C/S 6-8406-71-B10

NOTE: STATIC FRICTION  
± 1.5 LB

PILOT'S  
WHEEL FORCE  
 $F_w \sim \text{LBS}$

MAXIMUM  
ACTUATOR  
OUTPUT



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BOOST-ON LATERAL  
FEEL & CENTERING FORCES

THE BOEING COMPANY

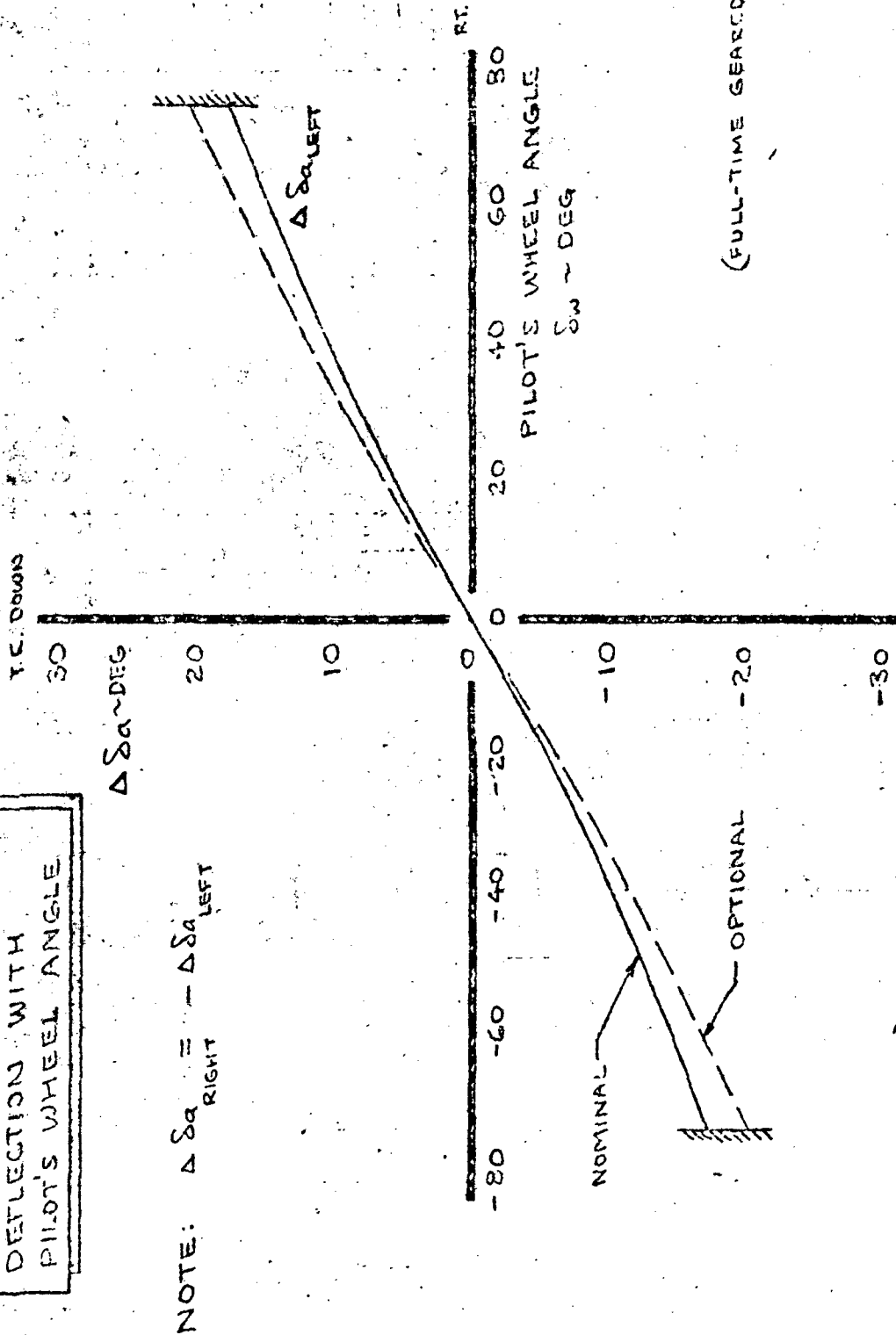
MOD C-8A

D6-26065TN

FIG 2-5

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DIFFERENTIAL AILERON  
DEFLECTION WITH  
PILOT'S WHEEL ANGLE



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AILERON PROGRAMMING  
WITH  
PILOT'S WHEEL

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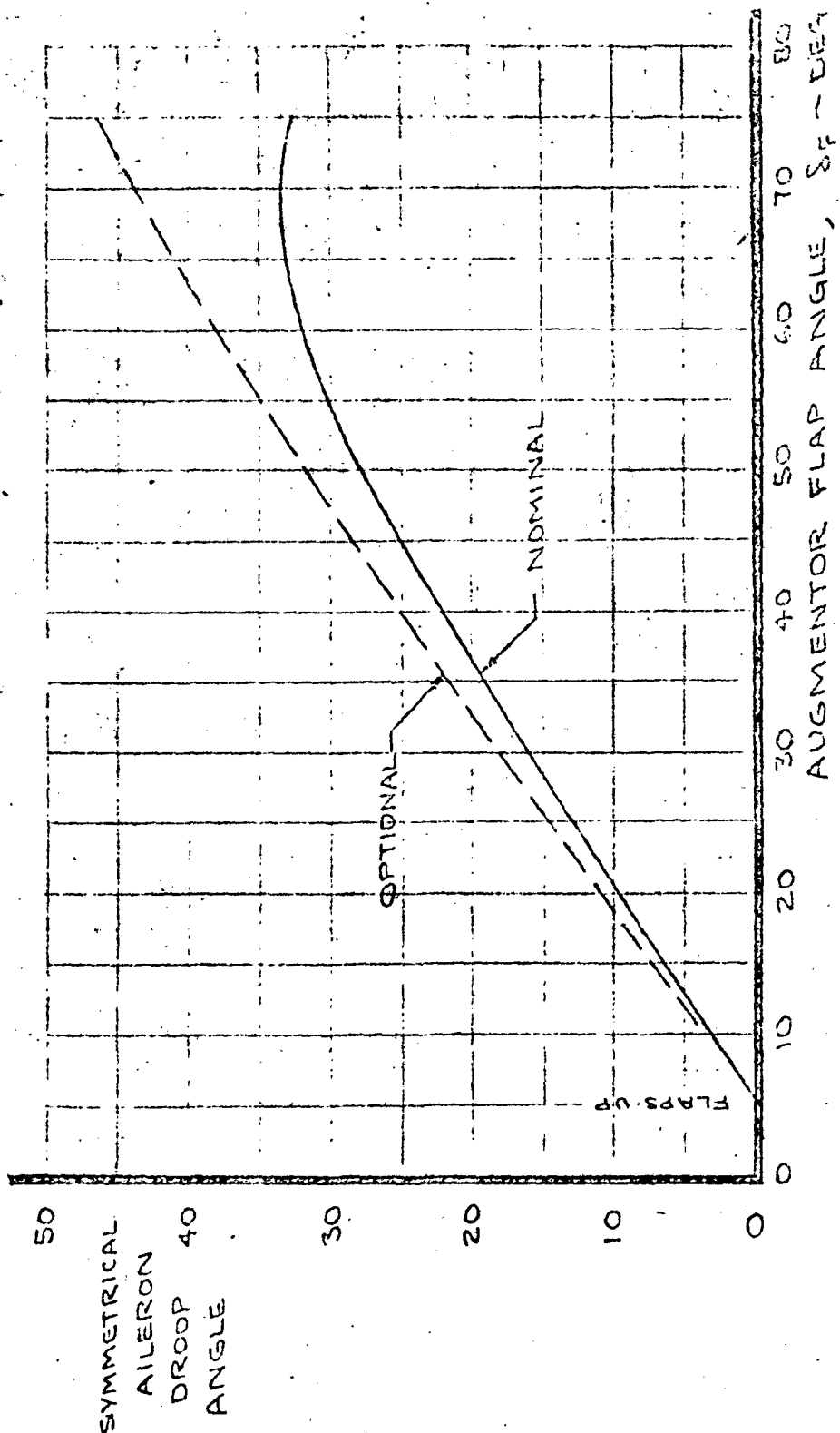
MOD C-8A

D6-26065TN

FIG 2-G

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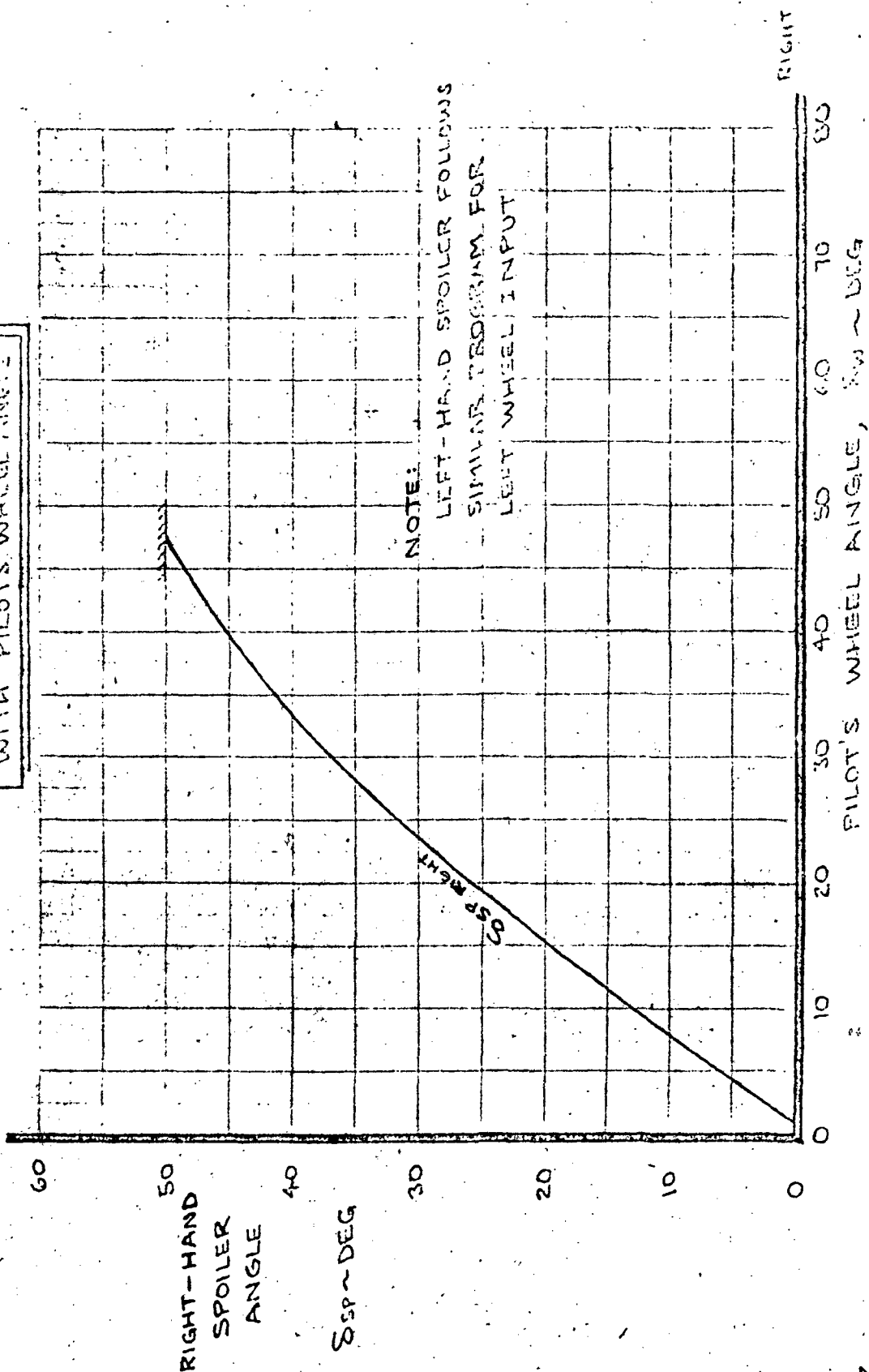
# AILERON DROOP PROGRAMMING



MOD C-BA

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| CHECK |         |         |         |      |                                                                        | FIG2-7     |
| APR   |         |         |         |      |                                                                        | PAGE       |
| APR   |         |         |         |      |                                                                        | 20         |

# SPOILER PROGRAMMING WITH PILOT'S WHEEL ANGLE



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SPOILER PROGRAMMING

THE BOEING COMPANY

MOD C-8A

06-26065TH

FIG 2-8

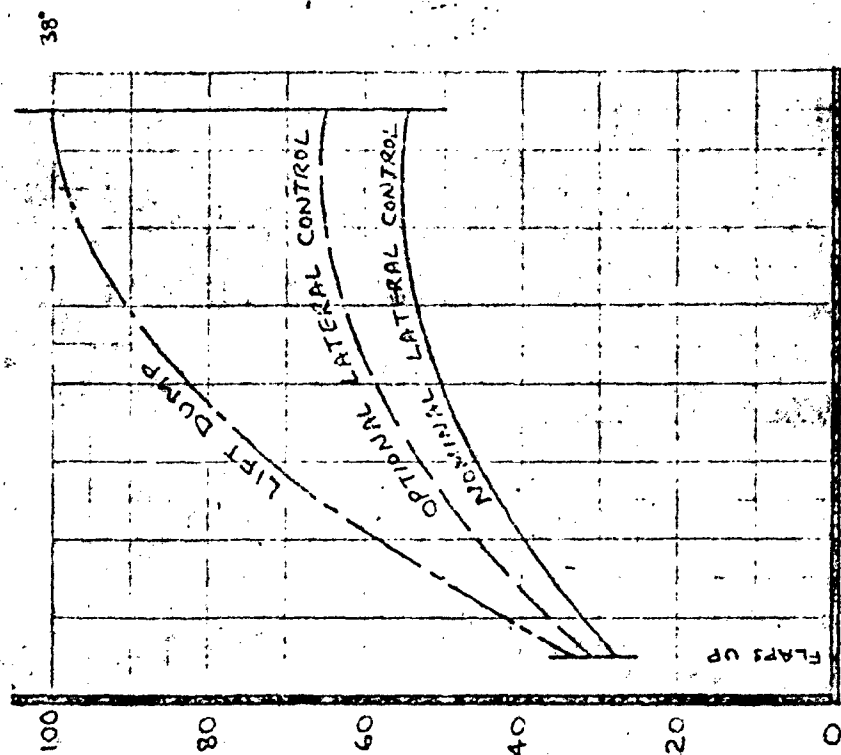
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# AUGMENTOR CHOKE PROGRAMMING

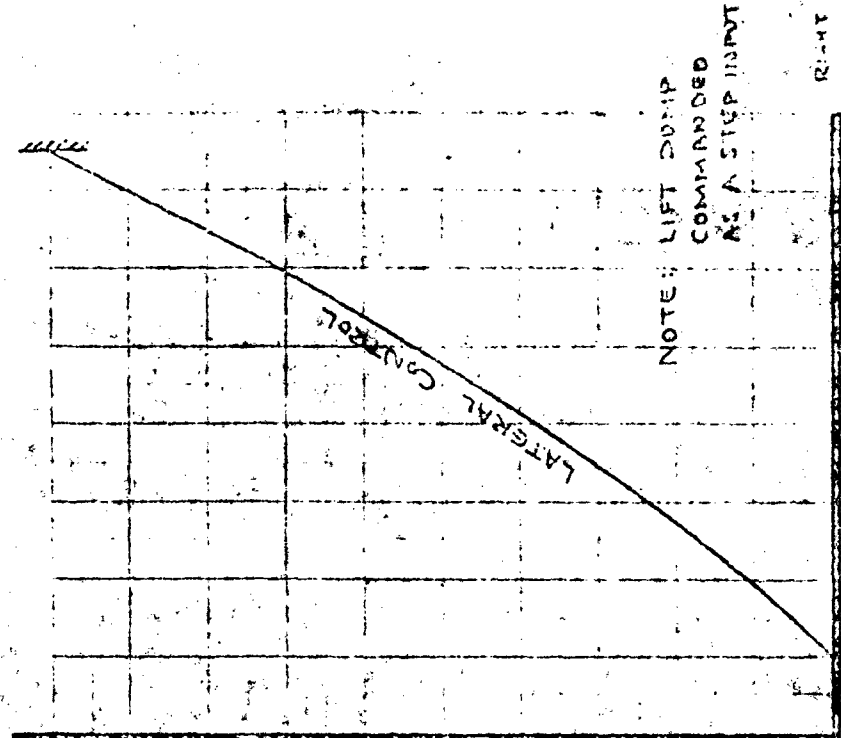
NOTE: RIGHT CHOKE DEFLECTS FOR RIGHT WHEEL  
LEFT CHOKE DEFLECTS FOR LEFT WHEEL

$$\delta_{CH} = G_{\delta_{CH}} \cdot \delta_{CH_{MAX}}$$

MAXIMUM AUGMENTOR CHOKE DEFLECTION,  $\delta_{CH} \sim \%$



CHOKE/WHEEL GEARING,  $G_{\delta_{CH}}$



## AUGMENTOR CHOKE PROGRAMMING

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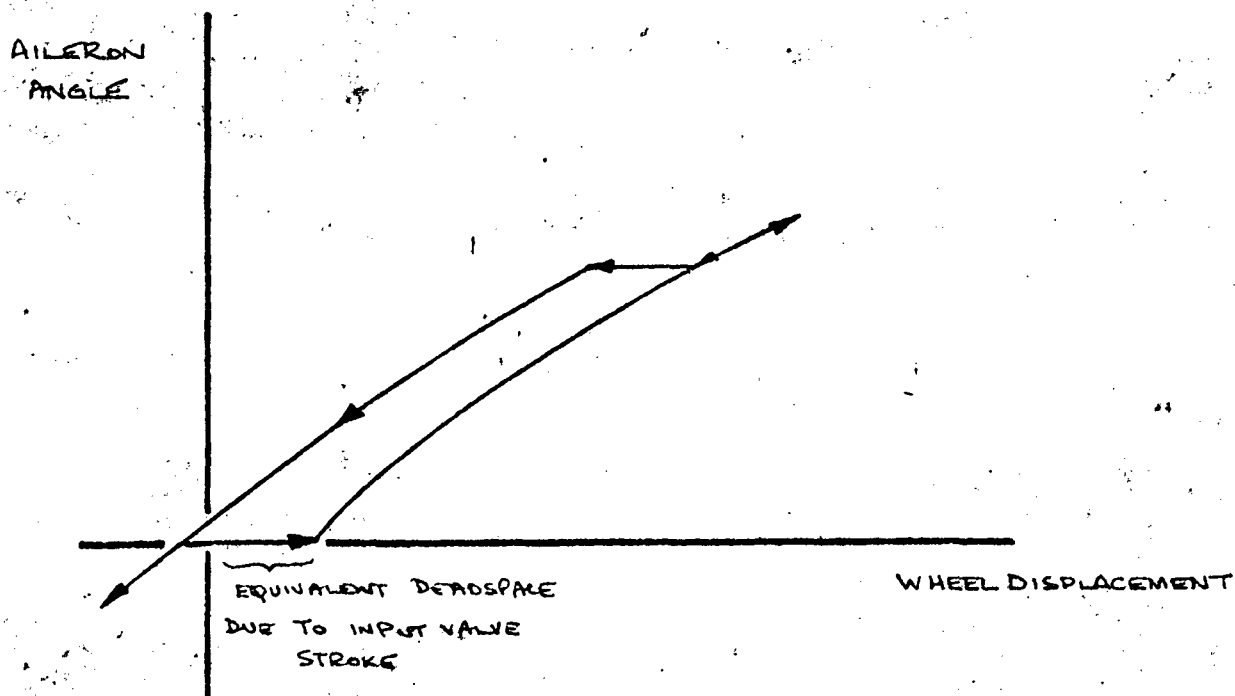
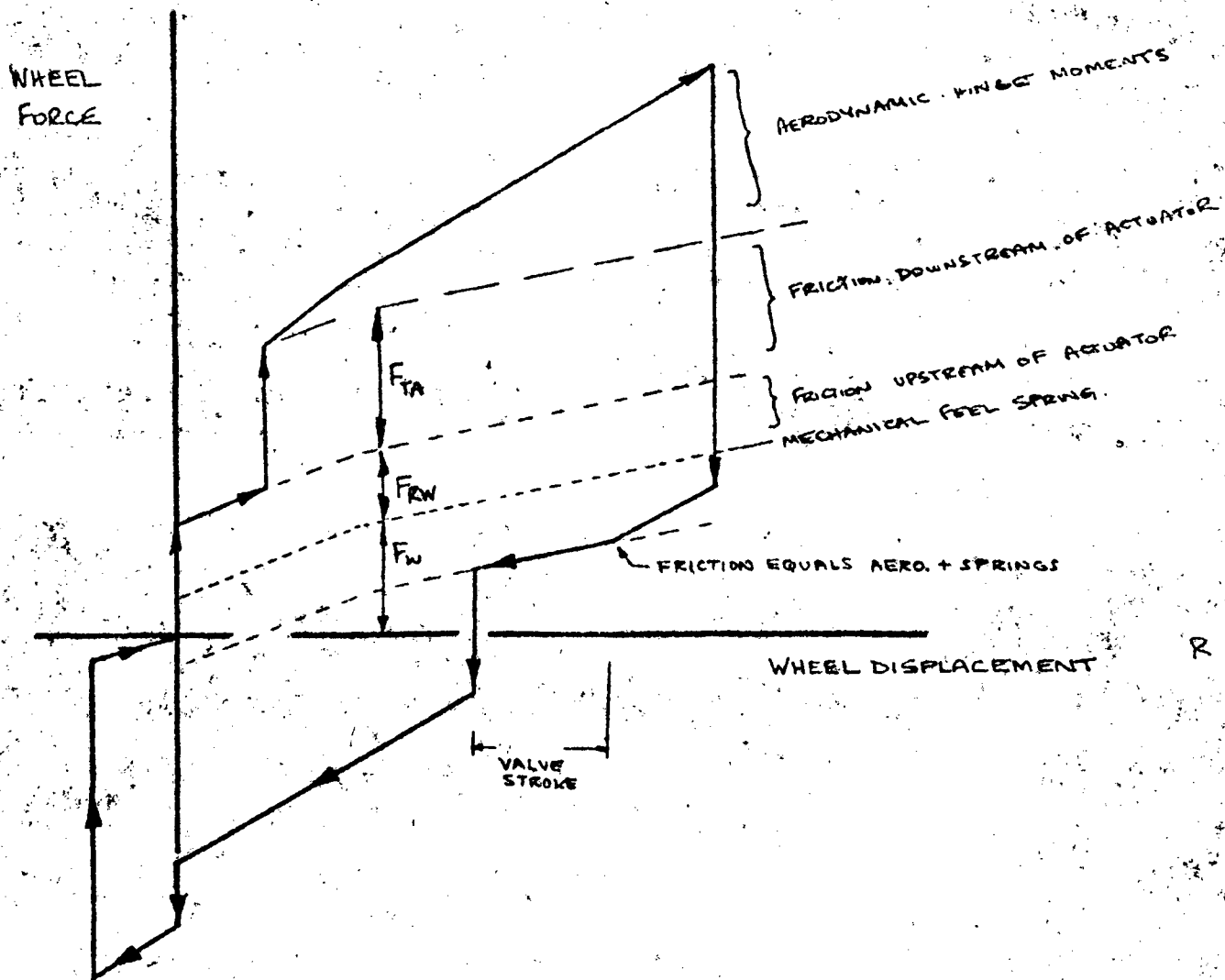
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FIG 2-9

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MANUAL REVERSION FORCE CHARACTERISTICS

D6-26065TN

FIG 2-10

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### 2.3 Pedal Control

The rudder system is a fully powered and irreversible system and as such there is assumed to be no control available over the rudder with both hydraulic systems failed. The flow chart for this system is shown on Figure 2-11 and the equations and constants follow:

Statically, the relation between pedal force,  $F_p$ , and pedal deflection,  $\delta_p$ , is  $F_p = \pm FR_p + K_p (\delta_p)$  lbs

where  $FR_p$  = friction = 6 lb

and  $K_p$  is shown on Figure 2-12 as a function of  $\delta_p$

NOTE: Positive rudder pedal forces produce positive rudder and pedal deflections.

Dynamically, the pedal deflection is given by the equation --

$$\delta_p = \frac{\delta_{p_{\text{demanded}}}}{\left(\frac{s}{\omega_{\text{ped}}}\right)^2 + 2 \cdot \frac{\zeta_{\text{ped}}}{\omega_{\text{ped}}} \cdot s + 1} \quad \text{inches}$$

where  $\omega_{\text{ped}}$  and  $\zeta_{\text{ped}}$  are set to values which give a good representation of the Buffalo directional control system as judged by the opinion of pilots who have flown the Buffalo.

$$\delta_{p_{\text{max}}} = \pm 4.0 \text{ inches (over a radius of 10.35")}$$

$$\text{Rudder deflection, } \delta_R = \frac{\delta_{R_{\text{DEM}}}}{(1 + T\delta_R \cdot s)} \quad \text{degrees}$$

where  $\delta_R$  is measured as the deflection of the forward rudder panel relative to the chord line of the vertical tail.

AD 1546D



$$T\delta_R = .138 \text{ secs.}$$

$$\delta_{R\text{DEN}} = G_R \cdot \delta_P + \frac{\delta_{RSAS}}{(1 + T\delta_{RSAS} \cdot S)}$$

up to a maximum of  
 $\pm 25^\circ$

where  $G_R = 6.25$  degrees/inch

$$T\delta_{RSAS} = 1/35 \text{ secs}$$

\*\*

$$\dot{\delta}_{RSAS \text{ MAX}} = \pm 25^\circ/\text{sec}$$

$$\delta_{RSAS \text{ MAX}} = \pm 10^\circ$$

The blowdown limited rudder deflections are given by the identities:

$$\begin{aligned} \pm \delta_{R\text{MAX}} &= \frac{5700 \cos \delta_R}{6.163 \times Q} \text{ degrees, with both} \\ & \text{hydraulic systems on.} \\ &= \frac{2850 \cos \delta_R}{6.163 \times Q} \text{ degrees, with only one} \\ & \text{hydraulic system.} \end{aligned}$$

The maximum rudder rate is limited to  $\pm 35^\circ/\text{sec}$

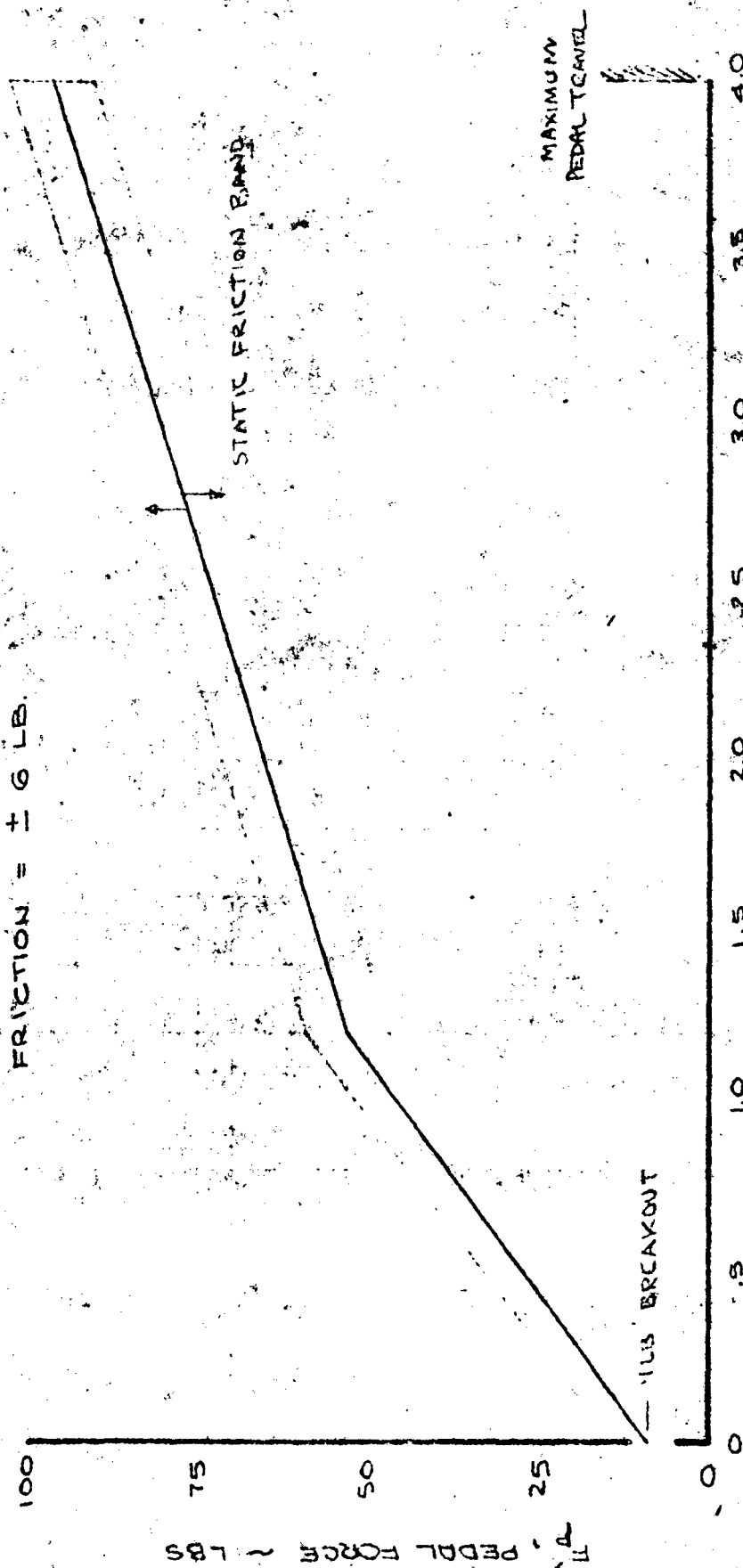
Rudder trim is accomplished by re-indexing the rudder feel system. This may be simulated by an additional input into the pedal force computation as shown in the flow diagram Fig. 2-11. The rudder trim rate is  $\pm 8^\circ/\text{sec}$ , and trim authority is  $\pm 5.2^\circ$  rudder deflection.



RUDDER PEDAL FORCE

$$K_P = F_P(\delta_P)$$

FRICITION =  $\pm 6$  LB.



$\delta_P$ , PEDAL TRAVEL, INS.

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PEDAL FORCES  
 $K_P(\delta_P)$

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MD0 C-8A

FIG.2-12

DL6-2606STN

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## 2.4 Throttle Controls

The throttle controls are mounted on an overhead console in the same relative position to pilot's eye as in the Buffalo. A variable friction nut is provided. The maximum lever throw will be  $38^\circ$ , the same as for the basic Buffalo. Figures 2-13 and 2-14 show the geometry. The throttle position,  $\delta_{TH}$ , feeds directly into the engine simulation module dealt with in Section 4.

## 2.5 Nozzle Vector Controls

The vector handles are mounted on an overhead console in the same relative position to the pilot's eye as the existing Buffalo propeller pitch controls, see Figures 2-13 and 2-14. Several handle and knob designs may be evaluated by the simulation pilots, and the simulator hardware should be capable of easy modification from one set to another. A variable friction nut should be provided. In the simulator, space should be provided for a total throw of the vector handles of  $90^\circ$ , a forward position ( $\delta_N = 0^\circ$ ) indicating that the hot thrust is being vectored aft ( $\nu = 6^\circ$ ), and an aft position ( $\delta_N = 39^\circ$ ), indicating that the thrust is vectored down and forwards ( $\nu = 104^\circ$ ).

Movement of the vector handles will generally cause the nozzles to move at their maximum rate to the new demanded position.

$$\text{Statically, } \nu_{\text{demand}} = G_\nu (\delta_N - \text{deadspace})$$

where  $G_\nu$  is linear giving full nozzle throw for full lever throw, and the deadspace is nominally zero.



\*\* Dynamically,  $\nu = \frac{\dot{\nu}}{S}$

\*\* where  $\dot{\nu}_{\text{MAX}} = \pm 60^\circ/\text{sec}$  for  $\nu \neq \nu_{\text{demand}}$

and  $\dot{\nu} = 0$  for  $\nu = \nu_{\text{demand}}$

## 2.6 Flap Deflection Controls

The flap selector will be mounted on an overhead console in the same relationship to the pilot's eye position as the flap selector in the Buffalo, see Figures 2-13 and 2-14. It will have the same throw as the Buffalo control, approximately  $\delta_{\text{FL}} = 65^\circ$ , with little or no friction, and gates set for FLAPS UP,  $30^\circ$ ,  $50^\circ$ ,  $65^\circ$  and  $75^\circ$  travel. Flap handle deflection,  $\delta_{\text{FL}}$ , causes the flaps to move at a given deflection rate until the demanded position is achieved. The flap deflection rate is fixed at  $\pm 4$  deg/sec.

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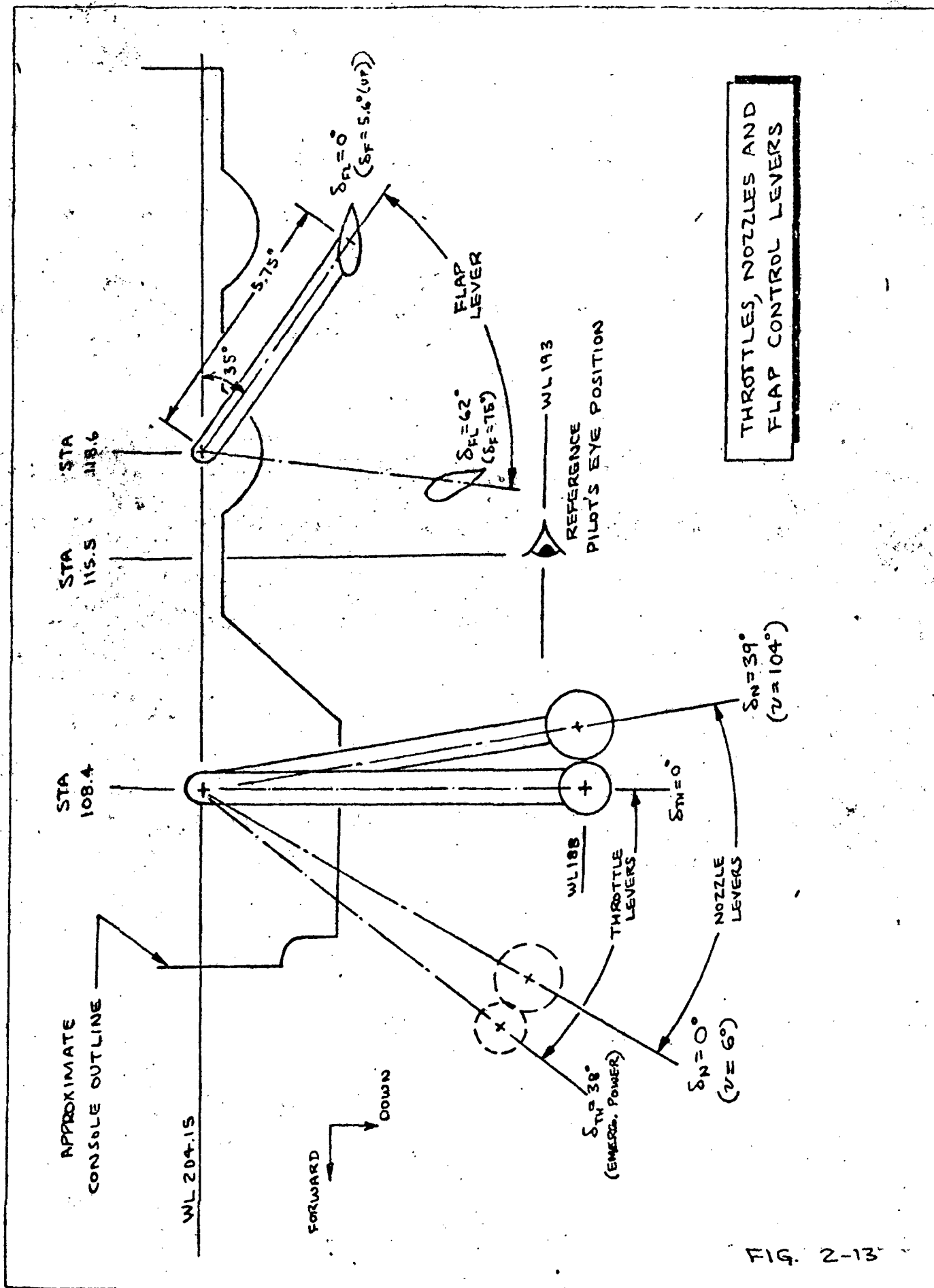
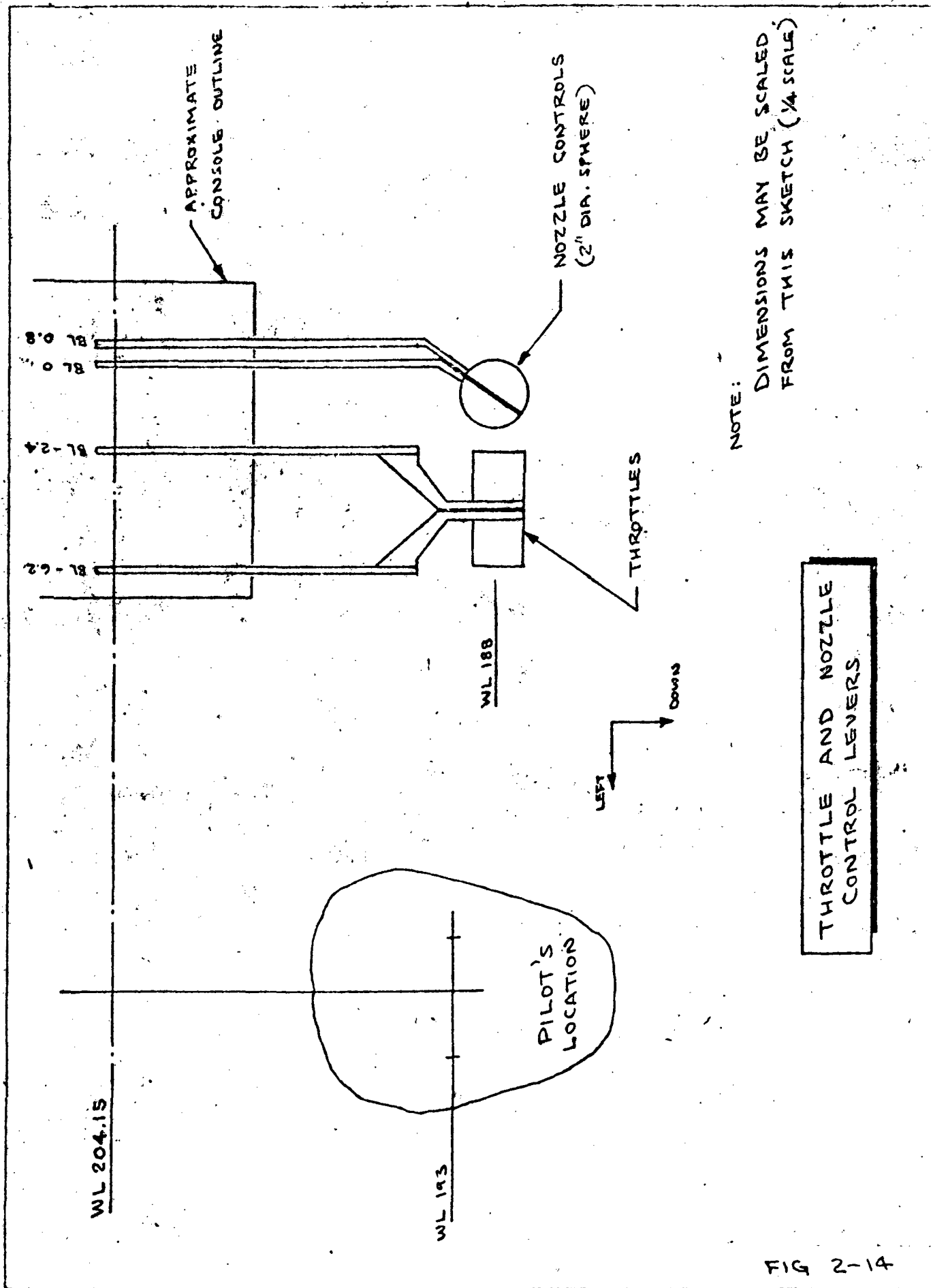


FIG. 2-13



NOTE:  
DIMENSIONS MAY BE SCALED  
FROM THIS SKETCH (1/4 SCALE)

THROTTLE AND NOZZLE  
CONTROL LEVERS

FIG 2-14

### 3.0 Aerodynamic Forces and Moments

The equations for the aerodynamic force and moment build-up in stability axes are shown on Tables 3-1 and 3-2, and the flow diagrams, Figures 3-1 to 3-6. This build-up differs from that used in previous studies of the augmentor wing FTV and utilizes directly data gained in the Phase IV and V large scale wind tunnel testing.

The ram drag effects have been subtracted out from the tunnel data, and are added back into the simulation as ram drag,  $D_{ram}$ , and ram drag pitching moment,  $M_{ram}$ , varying with flight speed.

$$D_{ram} = (\dot{M}_R + \dot{M}_L) \cdot V_A / 32.2 \sim \text{lbs}$$

$$M_{ram} = -D_{ram} (z_{EI} \cos \alpha_F - x_{EI} \sin \alpha_F) \sim \text{ft.lb.}$$

where  $\dot{M}_L$  and  $\dot{M}_R$  (mass flows into left and right engines) are defined in Section 4.

$V_A$  = total airplane velocity, ft/sec (true airspeed)

$\alpha_F$  = fuselage angle of attack =  $\alpha_W - 2.5^\circ$

$z_{EI}$  = distance from c.g. to engine  $\oplus$ , measured positive downwards in body axes =  $(\frac{WLC.G - 161.67}{12})$  ft.

$x_{EI}$  = distance from c.g. to engine inlet face, measured positive forwards in body axes =  $(\frac{BSC.G - 184}{12})$  ft.

Note that with one engine failed the ram drag on the operating engine causes a yawing moment as described in the equation for NS.

AD 1546 D

The tail angle of attack (table 3-1) includes the lag in downwash (in  $\epsilon^*$ ), and the tail incidence due to pitch rate,  $\Delta\alpha_Q$ . The aerodynamic effects of tail stall will not be included in the simulation, but it is essential to know if any maneuver brings the tail to the predicted buffet boundary or outside the range of linear aerodynamics. For this reason, when the horizontal or vertical tails exceed the incidences shown on Figure 3-7 a warning should be given to the simulation engineers - either in the form of a flashing light, or preferably by triggering an event marker on the data recorders. The downwash model is that presented in Table 3-3.

The rotary derivatives  $Ci_{\dot{\beta}}$ ,  $Ci_{\dot{p}}$ ,  $Ci_{\dot{r}}$  ( $i = n, l, y$ ) are computed using the polynomial expressions included here on Table 3-4.

The equations for the static sideslip derivatives (Table 3-1) have been changed to give a better fit to the latest available wind tunnel data. However, the derivative  $C_{l\beta}$  is still very poorly defined in this data, and various constant values will therefore be used in the simulator studies.

The rolling and yawing moment equations include a calculation of the out-of-balance moments caused by asymmetric blowing of the wings. This would be caused by asymmetric power from the engines since each wing derives its blowing air in unequal

proportions from each engine. To calculate these moments, use has been made of the existing  $\Delta C_{LWB}$  and  $\Delta C_{DWB}$  data. These tables are entered at an equivalent left and right  $C_J$  value for the flaps; and the differences in lift and drag between the wings are used to calculate rolling and yawing moments through the use of an effective moment arm,  $y_{c.p.}$ .

For the nominal 64/36 split of blowing air from each engine:

$$C_{JL} = 2 \left[ \frac{.38 T_{CL} + .43 T_{CR}}{\bar{Q} \cdot SW} \right]$$

$$C_{JR} = 2 \left[ \frac{.38 T_{CR} + .43 T_{CL}}{\bar{Q} \cdot SW} \right]$$

Since the  $\Delta C_{LWB}$  and  $\Delta C_{DWB}$  data are functions of  $C_{JTOT}$ , then the effective  $C_J$  values for each wing are doubled in order to read off the data in the correct  $C_J$  range. This ensures that the correct  $\frac{\partial C_L}{\partial C_J}$  and  $\frac{\partial C_D}{\partial C_J}$  are used to calculate the asymmetric forces. However, the forces so derived are double the actual values for each wing and this is taken into account in the equations as follows:

$$C_{L \text{ asymmetric blowing}} = 1/2 \left[ \Delta C_{LWB} \Big|_{C_J=C_{JL}} - \Delta C_{LWB} \Big|_{C_J=C_{JR}} \right] \times \frac{y_{c.p.}}{BW}$$

$$C_{N \text{ asymmetric blowing}} = 1/2 \left[ \Delta C_{DWB} \Big|_{C_J=C_{JR}} - \Delta C_{DWB} \Big|_{C_J=C_{JL}} \right] \times \frac{y_{c.p.}}{BW}$$

The individual components of each of the equations of Tables 3-1 and 3-2 are plotted on Figures 3-8 to 3-41. Table 3-5 is an index to these figures. Table 3-6 includes all the geometric constants used in the equations.

AD 1546 D



Since the major aim of the FTV is to demonstrate approach and landings at very high lift coefficients, the simulation should remain valid for speeds down to 30 knots.

AD 1546 D

# AERODYNAMIC FORCES IN STABILITY AXES

$$X_{STAB} = -\bar{Q} \cdot S_W \cdot \left\{ \Delta C_{D_{WB}}(\alpha_W, \delta_F, G_{TOT}) \cdot \left[ \frac{C_{LGE}}{C_{LFA}} \left( \frac{W}{CW} \right) + \Delta \frac{C_{LGE}}{C_{LFA}}(\alpha_W, \frac{h}{CW}) \right] \right. \\ \left. + \Delta C_{D_{\delta a}}(\delta_a, G_{SA})_{L+R} + \Delta C_{D_{\delta a}}(\delta_a, \alpha_W)_{L+R} + \Delta C_{D_{\delta sp}}(\delta_{sp}, \delta_a) \cdot K_{SP}(G_{SA})_{L+R} \right. \\ \left. + \left[ (\Delta C_{D_{\delta_{CH=50}}}(\alpha_W, G_{FLAP}) - .02) \cdot K_2(\delta_F) + .02 \right] \cdot K_{\delta CH}(\delta_{CH})_{L+R} \right\} - D_{RAM}$$

$$Y_{STAB} = \bar{Q} \cdot S_W \cdot \left\{ C_{Y\beta} \cdot \beta + \frac{S_V}{S_W} \cdot \Delta C_{L_V}(\delta_R, \beta) + C_{Y\delta a} \cdot (\delta_{aL} - \delta_{aR}) \right. \\ \left. + \Delta C_{Y_{\delta sp}}(\delta_{sp}) \cdot K_{SP}(G_{SA})_{R-L} + \Delta C_{Y_{\delta_{CH=50}}}(\alpha_W, G_{FLAP}) \cdot K_{\delta CH}(\delta_{CH}) \cdot K_1(\delta_F)_{R-L} \right. \\ \left. + \frac{BW}{2 \cdot V_A} \cdot [C_{Y\dot{\beta}} \cdot \dot{\beta} + C_{Y\dot{P}} \cdot P_S + C_{Y\dot{R}} \cdot R_S] \right\}$$

$$Z_{STAB} = -\bar{Q} \cdot S_W \cdot \left\{ \Delta C_{L_{WB}}(\alpha_W, \delta_F, G_{TOT}) \cdot \left[ \frac{C_{LGE}}{C_{LFA}} \left( \frac{W}{CW} \right) + \Delta \frac{C_{LGE}}{C_{LFA}}(\alpha_W, \frac{h}{CW}) \right] \right. \\ \left. + \Delta C_{L_{\delta a}}(\delta_a, G_{SA})_{L+R} + \Delta C_{L_{\delta sp}}(\delta_{sp}, \delta_a) \cdot K_{SP}(G_{SA})_{L+R} \right. \\ \left. + \Delta C_{L_{\delta_{CH=50}}}(\alpha_W, G_{FLAP}) \cdot K_{\delta CH}(\delta_{CH}) \cdot K_1(\delta_F)_{L+R} \right. \\ \left. + C_{L_{TAIL}} + \frac{CW}{2 \cdot V_A} [C_{L\dot{\alpha}}_{WING} \cdot \dot{\alpha} + C_{L\dot{Q}}_{WING} \cdot \dot{Q}_S] \right\}$$

$$C_{L_{TAIL}} = C_{L_{\delta e}} \frac{S_T}{S_W} \cdot \alpha_{TAIL} + C_{L_{\delta e}} \frac{S_T}{S_W} \cdot \delta_e$$

$$\alpha_{TAIL} = \alpha_F - \epsilon^* + i_T + \Delta \alpha_Q - \alpha_{OT} \quad \text{degrees}$$

$$\alpha_F = \alpha_W - 2.5^\circ \quad \Delta \alpha_Q = (l_T \cos \alpha_F + z_T \sin \alpha_F) \cdot \frac{QS}{V_A} \times 57.29$$

$$\alpha_{OT} = +0.8^\circ \quad i_T = \text{stabiliser incidence.} \\ = +1^\circ (\text{NOMINAL})$$

$$C_{Y\beta} = -.75 - .00504 \delta_F (\text{deg}) - .0344 \alpha_W (\text{deg}) \quad \text{— per radian.}$$

$$C_{Y\delta a} = -.0005/\text{deg} \quad C_{L\dot{\alpha}}_{WING} = 0$$

$$C_{L\dot{Q}}_{WING} = 0$$

$$C_{L_{\delta e}} = .061 / \text{degree} \quad C_{L_{\delta a}} = .035 / \text{degree}$$

For  $C_{Y\dot{\beta}}$ ,  $C_{Y\dot{P}}$ ,  $C_{Y\dot{R}}$  — see Table 3-4

For  $\epsilon^*$  — see Table 3-3

For GEOMETRIC CONSTANTS — see Table 3-6.

|            |               |           |
|------------|---------------|-----------|
| DC-26065TW |               |           |
| CALC       | RUNSY 9/20/70 | TABLE 3-1 |
|            |               | PAGE      |
|            |               | 36        |

# AERODYNAMIC MOMENTS IN STABILITY AXES

$$LS = \bar{Q} \cdot SW \cdot BW \cdot \left\{ C_{l\beta} \cdot \beta + \Delta C_{l\delta R}(\delta_R, \beta) \cdot \left( \frac{z_v}{BW} \cos \delta_F - \frac{l_v}{BW} \sin \delta_F \right) + \Delta C_{l\delta a}(\delta_a, \dot{\delta}_a)_{L-R} \right. \\ + \Delta C_{l\delta sp=50}(\alpha_w, \delta_a) \cdot \frac{C_l}{C_{l\delta sp=50}}(\delta_{sp}) \cdot K_{sp}(G_A)_{R-L} + \left[ \Delta C_{l\delta ch=50}(\alpha_w, G_{FMAP}) \cdot K_{\delta ch}(\delta_a) \cdot K_1(\delta_F) \right]_{R-L} \\ \left. + \frac{1}{2} \left[ \Delta C_{lwb} \right]_{G=CSL} \cdot \frac{y_{c.p.}}{BW} + \frac{BW}{2 \cdot V_A} \cdot \left[ C_{l\beta} \cdot P_S + C_{l\beta} \cdot R_S + C_{l\beta} \cdot \dot{\beta} \right] \right\}$$

$$MA = \bar{Q} \cdot SW \cdot CW \cdot \left\{ \Delta C_{mwb}(\alpha, \delta_F, G_{TOT}) \cdot \left[ \frac{C_{mcf}}{C_{mFA}}(\alpha_w, G_{TOT}, \frac{h}{CW}) + \frac{\Delta C_{mcf}}{C_{mFA}}(\alpha_w, \frac{h}{CW}) \right] \right. \\ + \Delta C_{m\delta a}(\delta_a, G_A)_{L-R} + \Delta C_{m\delta sp}(\delta_{sp}, \delta_a) \cdot K_{sp}(G_A)_{L-R} + \Delta C_{m\delta ch=50}(\alpha_w, G_{FMAP}) \cdot K_{\delta ch}(\delta_a) \cdot K_1(\delta_F)_{L-R} \\ + \Delta C_{mwb} \cdot \left( \frac{l_{cg} \cos \delta_F + \frac{z_{cg}}{CW} \sin \delta_F \right) - \Delta C_{mwb} \cdot \frac{z_{cg} \cos \delta_F}{CW} - \frac{C_l}{I_{AR}} \cdot \left( \frac{l_T}{CW} \cos \delta_F + \frac{z_b}{CW} \sin \delta_F \right) \\ \left. + \frac{CW}{2 \cdot V_A} \left[ C_{m\dot{\alpha}} \cdot \dot{\alpha} + C_{m\dot{q}} \cdot \dot{q}_S \right] \right\} + M_{RAM}$$

$$NS = \bar{Q} \cdot SW \cdot BW \cdot \left\{ C_{N\beta} \cdot \beta - \Delta C_{N\delta R} \cdot \left( \frac{z_v}{BW} \sin \delta_F + \frac{l_v}{BW} \cos \delta_F \right) + \Delta C_{N\delta a}(\delta_a, \dot{\delta}_a)_{L-R} \right. \\ + \Delta C_{N\delta a}(\delta_a, \alpha_w)_{L-R} + \Delta C_{N\delta sp=50}(\alpha_w, \delta_a) \cdot \frac{C_N}{C_{N\delta sp=50}}(\delta_{sp}, \delta_a) \cdot K_{sp}(G_A)_{R-L} \\ + \left[ \left( \Delta C_{N\delta ch=50}(\alpha_w, G_{FMAP}) - .005 \right) \cdot K_2(\delta_F) + .005 \right] \cdot K_{\delta ch}(\delta_a)_{R-L} \\ \left. + \frac{1}{2} \left[ \Delta C_{Nwb} \right]_{G=CSL} \cdot \frac{y_{c.p.}}{BW} + \frac{BW}{2 \cdot V_A} \left[ C_{N\beta} \cdot P_S + C_{N\beta} \cdot R_S + C_{N\beta} \cdot \dot{\beta} \right] \right\} \\ + (\dot{M}_R - \dot{M}_L) \cdot \frac{V_A}{g} \cdot y_E + Y_{STAB} \cdot l_{cg} \cdot \cos \delta_F$$

$$C_{N\beta} = .149 + .00154 \delta_F^\circ + .000766 \alpha_w^\circ - .0397 G_{TOT} \quad \text{— per radian.}$$

$$C_{l\beta} \approx 0 \text{ @ } \delta_F = 75^\circ \text{ (MAX. VALUE MAY GO TO } C_{l\beta} = -.004/\text{DEG)}$$

$$l_v = \frac{856.08 - BSCG}{12} \text{ ft.}$$

$$z_v = \frac{306.61 - WLCA}{12} \text{ ft.}$$

$$l_{cg} = \frac{BSCG - 335.2}{12} \text{ ft.}$$

$$z_{cg} = \frac{WLCA - 168.88}{12} \text{ ft.}$$

$$l_T = \frac{888.6 - BSCG}{12} \text{ ft.}$$

$$z_T = \frac{403.84 - WLCA}{12} \text{ ft.}$$

$$C_{m\dot{\alpha}}_{wing} = 0$$

$$C_{m\dot{q}}_{wing} = 0$$

For  $C_{l\beta}$ ,  $C_{l\dot{\alpha}}$ ,  $C_{l\dot{q}}$ ,  $C_{N\beta}$ ,  $C_{N\dot{\alpha}}$ ,  $C_{N\dot{q}}$  — see Table 3-4.  
For geometry constants and arms — see Table 3-6.

|            |                |           |
|------------|----------------|-----------|
| DG-26065TN |                |           |
| CALC       | RUNSET 9/20/70 | TABLE 3-2 |
|            |                | PAGE 37   |

DOWNWASH MODEL

The downwash model used was identical to a previous NASA Ames simulation for a configuration very similar to the Modified C-8A. No changes were made to reflect the actual F.T.V. geometry except that the lag in downwash was computed using the latest values of tail length  $l_T$  and  $z_T$ , see Table 3-2. The model has been used to generate the reference downwash data for various flap settings shown in Figures 3-13 to 3-17.

DOWNWASH MODEL

Table 3-3



# ROTARY DERIVATIVE EQUATIONS

The existing (Reference 2) equations for the rotary derivatives of a configuration similar to the Modified C-8A have been used with no modifications to reflect the actual geometry. These equations analytically define data curves generated by deHavilland. The values of  $C_{LWB}$  used in these equations is replaced by the output of the table look up routine in the new aerodynamic build-up. For reference purposes the rotary derivative equations are as follows:

$$C_{l\beta} = (.0975 \delta_F + .222 \alpha_w) \delta_F - .19 C_J' - .54$$

RADIANS      RADIANS      RADIANS

$$C_{l\beta} = 1.04 C_{LWB} (.23 - .0656 \sqrt{C_J'}) P_f' + .045 + 1.99 \times 10^{-4} Z_{Vsa} Z_{Vca}$$

where  $P_f' = 1 + P' \left( \frac{\alpha_w}{\delta_F} \right)$  ,  $P' = \begin{cases} .30 & \alpha_w < 0 \\ .15 & \alpha_w \geq 0 \end{cases}$  ,  $C_J' = 1.2 C_J$   
RADIANS  
 and

$$Z_{Vsa} = Z_V \sin \alpha_F + l_V \cos \alpha_F , \quad Z_{Vca} = Z_V \cos \alpha_F - l_V \sin \alpha_F$$

$$C_{l\beta} = -1.712 \times 10^{-5} Z_{Vca} Z_{Vsa}$$

$$C_{n\beta} = (.0138 CG - .00484) C_{LWB} + .0234 CG - .00578$$

$$- [.066 + (.0108 \delta_{F_{RAD}} + .0267 \alpha_{w_{RAD}}) \delta_{F_{RAD}} - .02 C_J'] C_{LWB}$$

$$+ Z_{Vsa} (1.99 \times 10^{-4} Z_{Vca} - .00216)$$

where CG ~ MAC fraction

$$C_{n\beta} = .1 C_{LWB}^2 (.0118 - .0076 C_J'^2) Q_F' - 1.99 \times 10^{-4} Z_{Vsa}^2$$

$$- .25 [.08 + .0004 \delta_{F_{RAD}} + R' C_{JA} - (1.27 - .00692 \delta_{F_{RAD}}) C_J']$$

$$C_{n\beta} = 1.712 \times 10^{-5} Z_{Vsa}^2$$

where  $R' = 2 - .0166 \delta_{F_{RAD}}$  ,  $Q_F' = 1 + Q' \left( \frac{\alpha_w}{\delta_F} \right)$  ,  $Q' = \begin{cases} .36 & \alpha_w < 0 \\ .18 & \alpha_w \geq 0 \end{cases}$   
RAD      RAD

$$C_{Y\beta} = .216 C_{l\beta} - .15 C_{LWB} - .0172 Z_{Vca} + .186$$

$$C_{Y\beta} = -.0112 C_{LWB}^2 + .0172 Z_{Vsa}$$

$$C_{Y\beta} = -1.48 \times 10^{-3} Z_{Vsa}$$

|       |         |         |         |      |                                                 |            |
|-------|---------|---------|---------|------|-------------------------------------------------|------------|
| CALC  | SPITZER | 7/12/71 | REVISED | DATE | ROTARY DERIVATIVE<br>EQUATIONS                  | TABLE 3-A  |
| CHECK |         |         |         |      |                                                 |            |
| APPD  |         |         |         |      |                                                 | D6-26065TN |
| APPD  |         |         |         |      | THE <b>BOEING</b> COMPANY<br>RENTON, WASHINGTON | PAGE<br>39 |

# INDEX TO AERODYNAMIC DATA

| DESCRIPTION                           | FIGURE NO.                           | PLOTTED DATA                                                                                                                                                                                                                                                                                                                       |
|---------------------------------------|--------------------------------------|------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| BASIC WING-BODY AERODYNAMICS          | 3-8<br>3-9<br>3-10<br>3-11<br>3-12   | $\delta F = UP$<br>30°<br>50°<br>65°<br>75°<br>$\Delta C_{LWB}, \Delta C_{DWB}, \Delta C_{mWB} \sim f(\alpha, C_T)$                                                                                                                                                                                                                |
| COMPUTED DOWNWASH<br>(Reference only) | 3-13<br>3-14<br>3-15<br>3-16<br>3-17 | $\delta F = UP$<br>30°<br>50°<br>65°<br>75°<br>$E \sim f(\alpha, C_T)$                                                                                                                                                                                                                                                             |
| FLAPS 75                              | 3-18                                 | LIFT $\sim f(h/cw)$ NO HOT THRUST                                                                                                                                                                                                                                                                                                  |
| GROUND EFFECT FACTORS                 | 3-19<br>3-20<br>3-21<br>3-22<br>3-23 | $\sim f(\alpha_w, h/cw)$ HOT THRUST INTERFERENCE<br>DRAG $\sim$ NO HOT THRUST<br>$\sim$ HOT THRUST INTERFERENCE<br>PITCHING MOMENT $\sim$ NO HOT THRUST<br>$\sim$ HOT THRUST INTERFERENCE                                                                                                                                          |
| AILERON EFFECTIVENESS                 | 3-24<br>3-25<br>3-26                 | $\Delta C_{Lsa}, \Delta C_h, \Delta C_y \sim f(\delta_a, C_{JA})$<br>$\Delta C_{Lsa}, \Delta C_{Dsa}, \Delta C_{msa} \sim f(\delta_a, C_{JA})$<br>$\Delta C_{hsa}, \Delta C_{Dsa} \sim f(\alpha_w, \delta_a)$                                                                                                                      |
| SPOILER EFFECTIVENESS                 | 3-27<br>3-28<br>3-29<br>3-30         | EFFECTIVENESS WITH DEFLECTION TO $\delta_{sp} = 50^\circ$<br>$\Delta C_L, \Delta C_h @ \delta_{sp} = 50^\circ \sim f(\delta_a, \alpha_w)$<br>$\Delta C_y, \Delta C_L, \Delta C_D, \Delta C_m \sim f(\delta_a, \delta_{sp})$<br>EFFECT OF $C_{JA}$ ON EFFECTIVENESS                                                                 |
| CHOKE EFFECTIVENESS                   | 3-31<br>3-32<br>3-33<br>3-34<br>3-35 | $\Delta C_L, \Delta C_h @ \delta_{ch} = .5 \sim f(\alpha_w, C_{JFLAP})$<br>$\Delta C_y, \Delta C_L @ \delta_{ch} = .5 \sim f(\alpha_w, C_{JFLAP})$<br>$\Delta C_D, \Delta C_m @ \delta_{ch} = .5 \sim f(\alpha_w, C_{JFLAP})$<br>EFFECTIVENESS WITH DEFLECTION $\sim \delta_{ch}$<br>EFFECTIVENESS WITH FLAP ANGLE $\sim \delta_F$ |
| RUDDER EFFECTIVENESS                  | 3-36                                 | $\Delta C_{Lv} \sim f(\delta_R)$                                                                                                                                                                                                                                                                                                   |

|       |         |         |         |      |                                                   |            |
|-------|---------|---------|---------|------|---------------------------------------------------|------------|
| CALC  | SPITZER | 7/12/71 | REVISED | DATE | INDEX TO AERODYNAMIC DATA                         | TABLE 3-5  |
| CHECK |         |         |         |      |                                                   |            |
| APPD  |         |         |         |      |                                                   | D6-26065TA |
| APPD  |         |         |         |      | THE <b>BRENNING</b> COMPANY<br>RENTON, WASHINGTON | PAGE 40    |

## LIST OF GEOMETRIC CONSTANTS

• AREAS:  $S_T = 232 \text{ ft}^2$   
 $S_W = 865 \text{ ft}^2$   
 $S_V = 152 \text{ ft}^2$

• WING GEOMETRY:

$CW = 12.4 \text{ ft}$   
 $BW = 78.75 \text{ ft}$   
 Aspect Ratio,  $AR = 7.2$

• ASYMMETRIC BLOWING CENTER OF PRESSURE:

$$Y_{CP} = \frac{200}{12} \text{ ft.} = 16.66 \text{ ft.}$$

• ENGINE MOMENT ARMS:

Hot thrust yawing moment arm,  $Y_E = \frac{183}{12} \text{ ft} = 15.25 \text{ ft}$   
 & ram drag yawing arm.

Hot thrust pitching moment arms:—

Nozzle center is at BS 324  
 WL 145

∴ moment arm about c.g. is

$$\left\{ \frac{(324 - BSCG)}{12} \sin V - \frac{(WL_{CG} - 145.00)}{12} \cos V \right\}$$

where  $V$  = angle between fuselage reference line and line of action of hot jet.

Note that This gives a nose down pitching moment using this expression for the moment arm.

• GEAR POSITION:

At touchdown (extended gear)

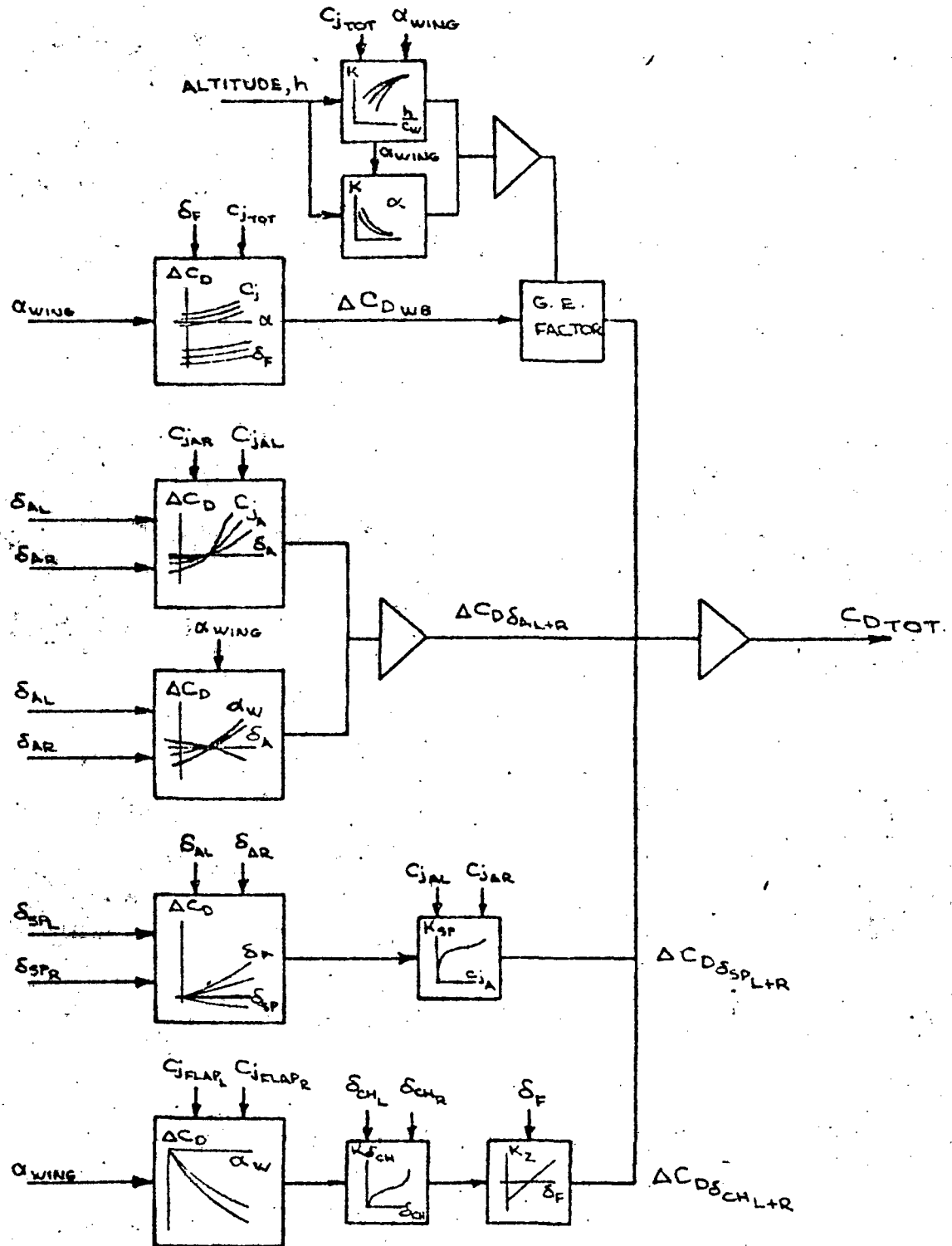
Nose gear is at WL 63, BS 50 and on the center-line.

Main gear is at WL 46, BS 384 and BL 183

At max gross weight, compressed oleo, the main gear is at WL 63

|       |        |         |         |      |                                         |            |
|-------|--------|---------|---------|------|-----------------------------------------|------------|
| ENGR. | RUNSEY | 10/2/70 | REV. ED | DATE | GEOMETRIC CONSTANTS                     | TABLE 3-6  |
| CHECK |        |         |         |      |                                         | DG-2606STN |
| APR   |        |         |         |      |                                         | PAGE       |
| APR   |        |         |         |      |                                         | 41         |
|       |        |         |         |      | THE BOEING COMPANY<br>RENTON WASHINGTON |            |

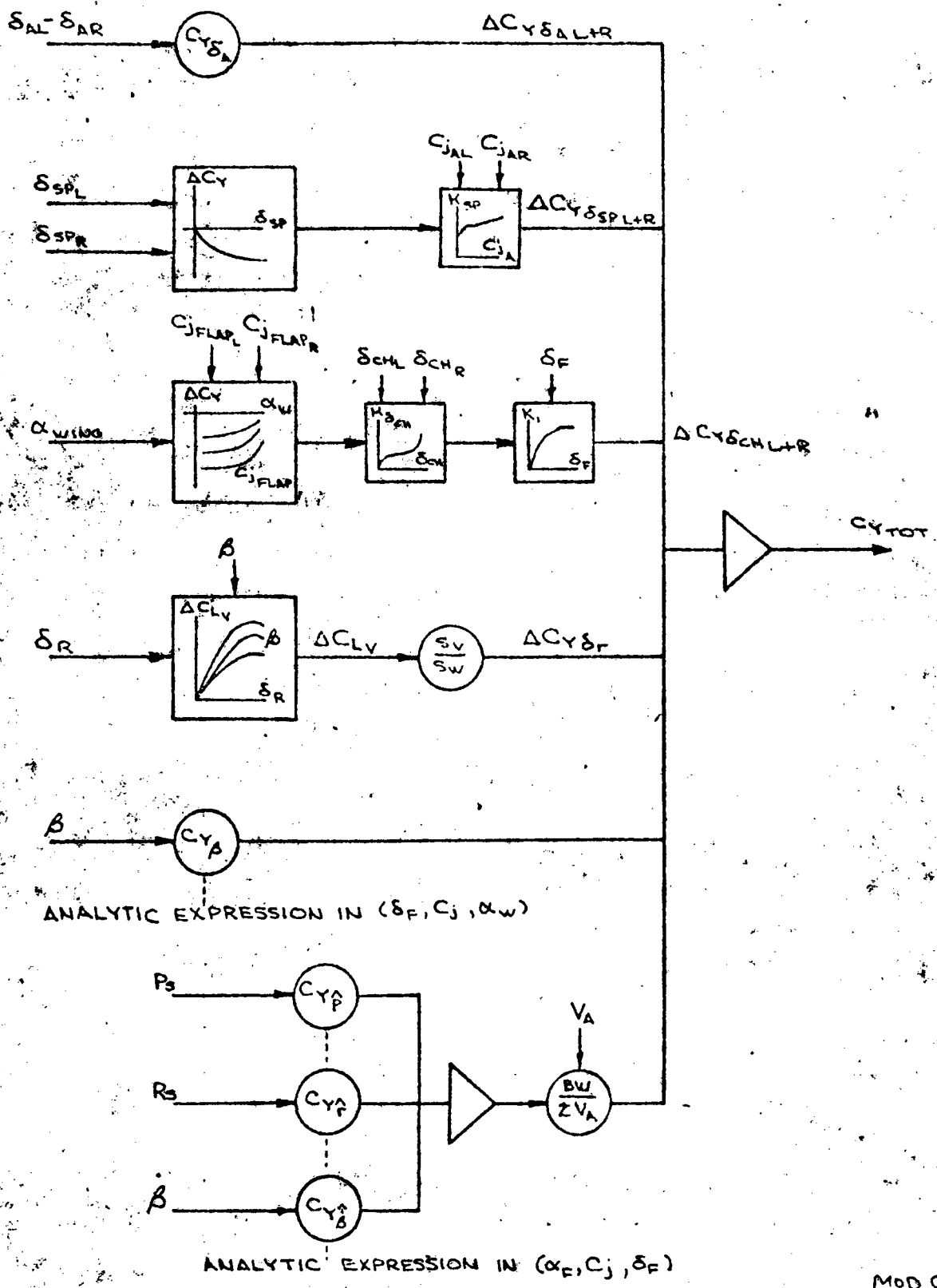
$$X_{STAB} = -\bar{Q} \cdot S_W \cdot C_{DTOT} - DRAM$$



$$DRAM = (\dot{M}_L + \dot{M}_R) \cdot V_A / 32.2$$

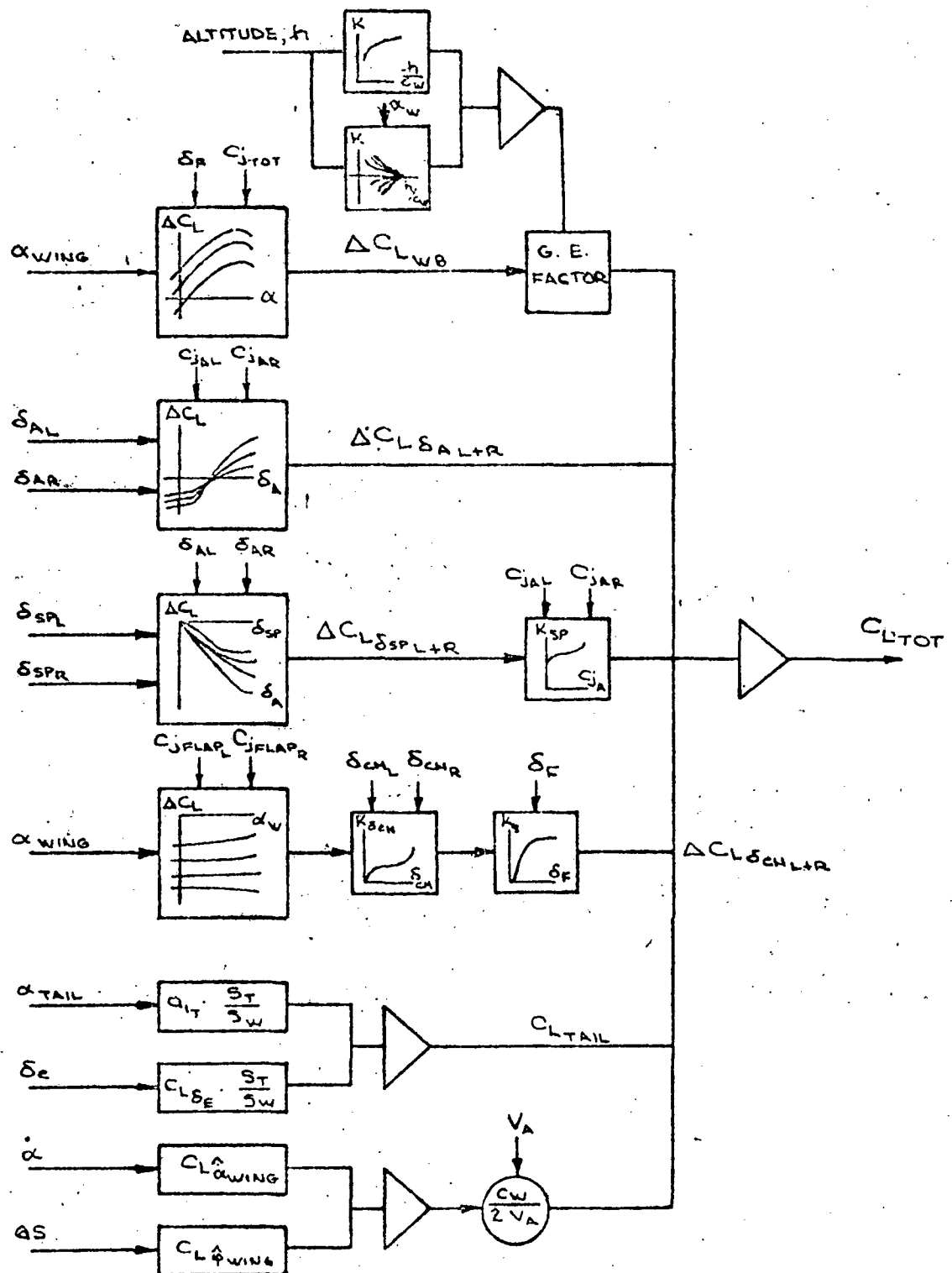
|       |        |          |         |      |                           |            |
|-------|--------|----------|---------|------|---------------------------|------------|
| CAL   | RUMSEY | 11/20/70 | REVISED | DATE | AERODYNAMIC DATA BUILD UP | MOD C-8A   |
| CHECK |        |          |         |      | X FORCE EQN.              | PG 3-1     |
| APPD  |        |          |         |      |                           | DG-26065TN |
| APPD  |        |          |         |      | BOEING AIRPLANE COMPANY   | PAGE 42    |

$$Y_{STAT} = \bar{Q} \cdot S_{VI} \cdot C_{YTOT}$$



|                                                                                                       |  |  |  |                                                                          |  |                       |
|-------------------------------------------------------------------------------------------------------|--|--|--|--------------------------------------------------------------------------|--|-----------------------|
| <div> <div>CALC</div> <div>RUMSEY</div> <div>11/20/70</div> <div>REVISED</div> <div>DATE</div> </div> |  |  |  | <div> <div>AERODYNAMIC DATA BUILD UP</div> <div>Y FORCE EQN</div> </div> |  | <div>MOD C-8A</div>   |
| <div> <div>CHECK</div> <div></div> <div></div> <div></div> </div>                                     |  |  |  | <div> <div>BOEING AIRPLANE COMPANY</div> </div>                          |  | <div>FIG 3-2</div>    |
| <div> <div>APPD</div> <div></div> <div></div> <div></div> </div>                                      |  |  |  | <div> <div>43</div> </div>                                               |  | <div>06-26065TN</div> |

$$Z_{STAB} = -\bar{Q} \cdot S_w \cdot C_{LTOT}$$



|       |        |          |         |      |
|-------|--------|----------|---------|------|
| CALC  | RUMSEY | 11/20/70 | REVISED | DATE |
| CHECK |        |          |         |      |
| APPD  |        |          |         |      |
| APPD  |        |          |         |      |

AERODYNAMIC DATA BUILD OP  
Z FORCE EQN.

BCEING AIRPLANE COMPANY

MOD C-8A

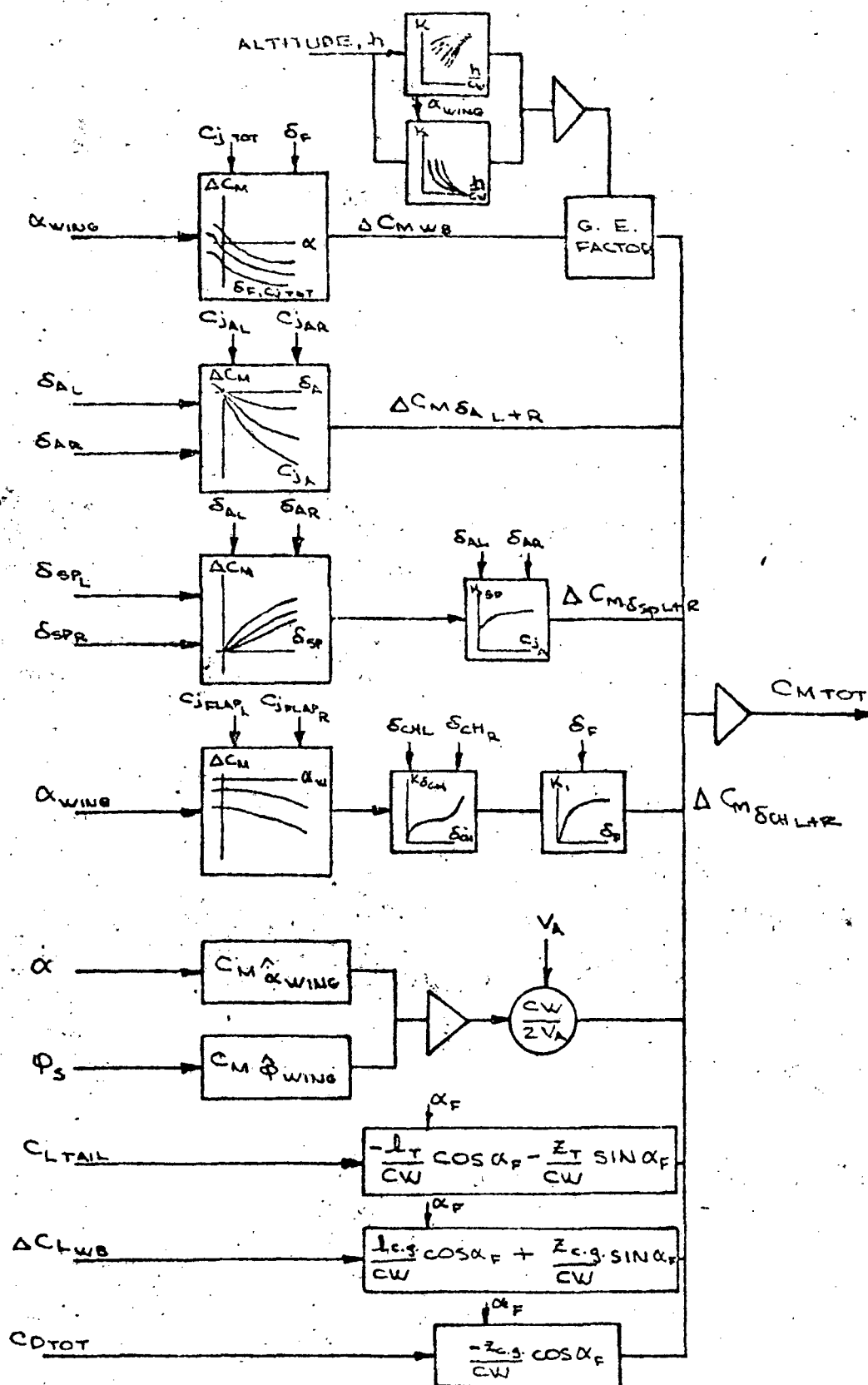
FIG 3-3

DL-26065TN

PAGE 44

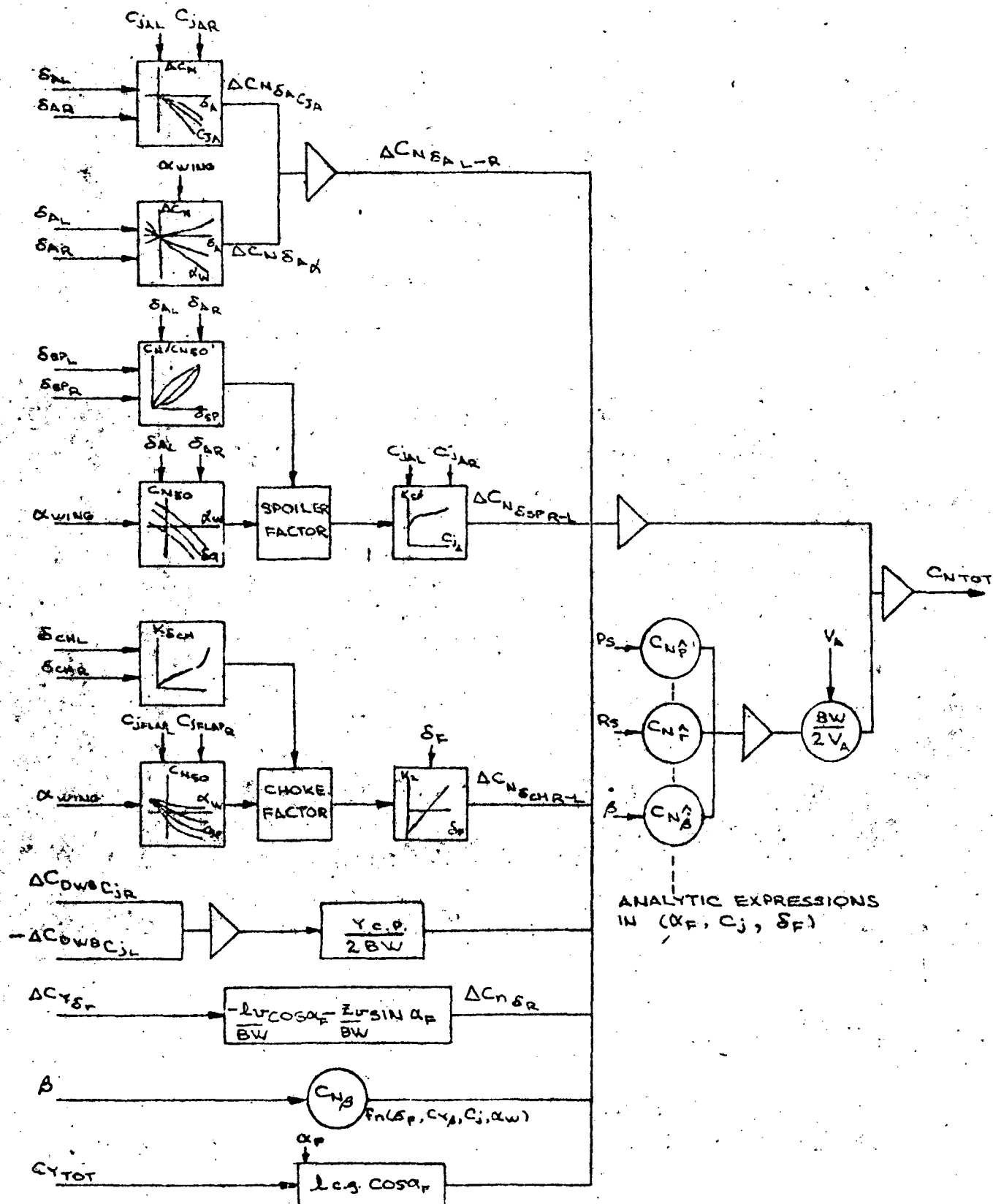


$$MS = \bar{Q} \cdot Sw \cdot CW \cdot CMTOT - DRAM (Z_{EI} \cos \alpha_F - X_{EI} \sin \alpha_F)$$



|       |        |          |         |      |                           |            |
|-------|--------|----------|---------|------|---------------------------|------------|
| CALC  | RUMSEY | 11/20/70 | REVISED | DATE | AERODYNAMIC DATA BUILD UP | FIG 3-5    |
| CHECK |        |          |         |      | PITCHING MOMENT EQUATION  | D6-26065TN |
| APPO  |        |          |         |      |                           |            |
| APPO  |        |          |         |      | BOEING AIRPLANE COMPANY   | PAGE 46    |

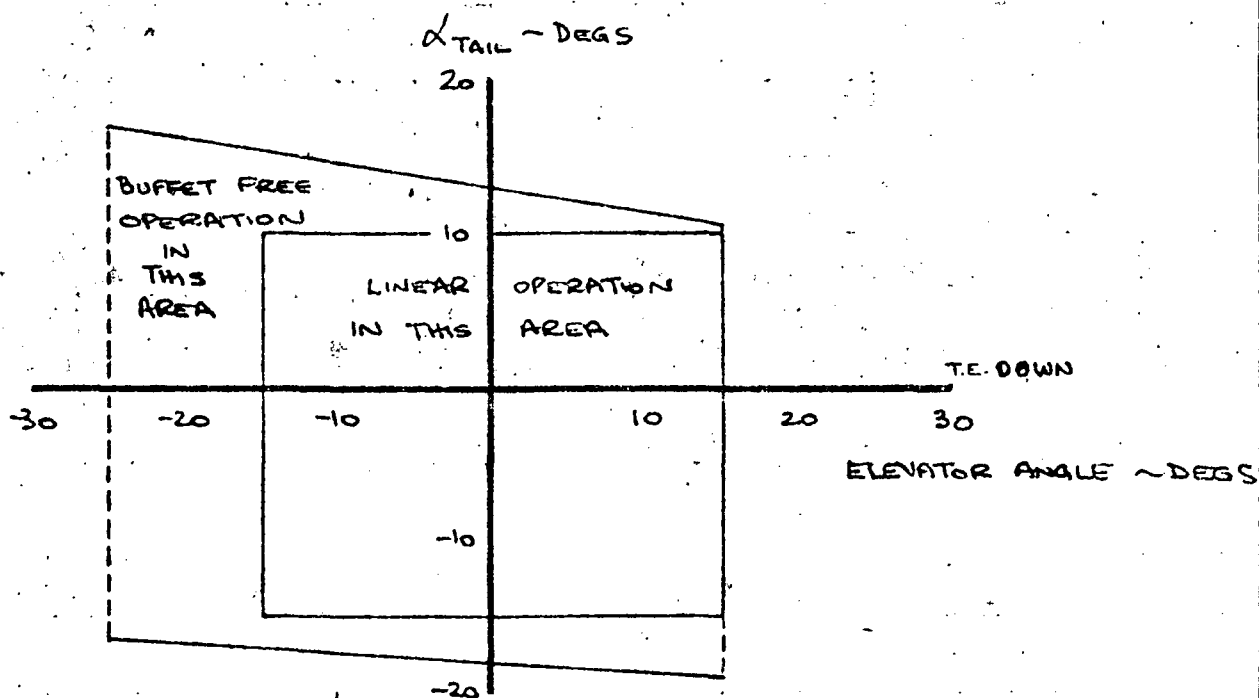
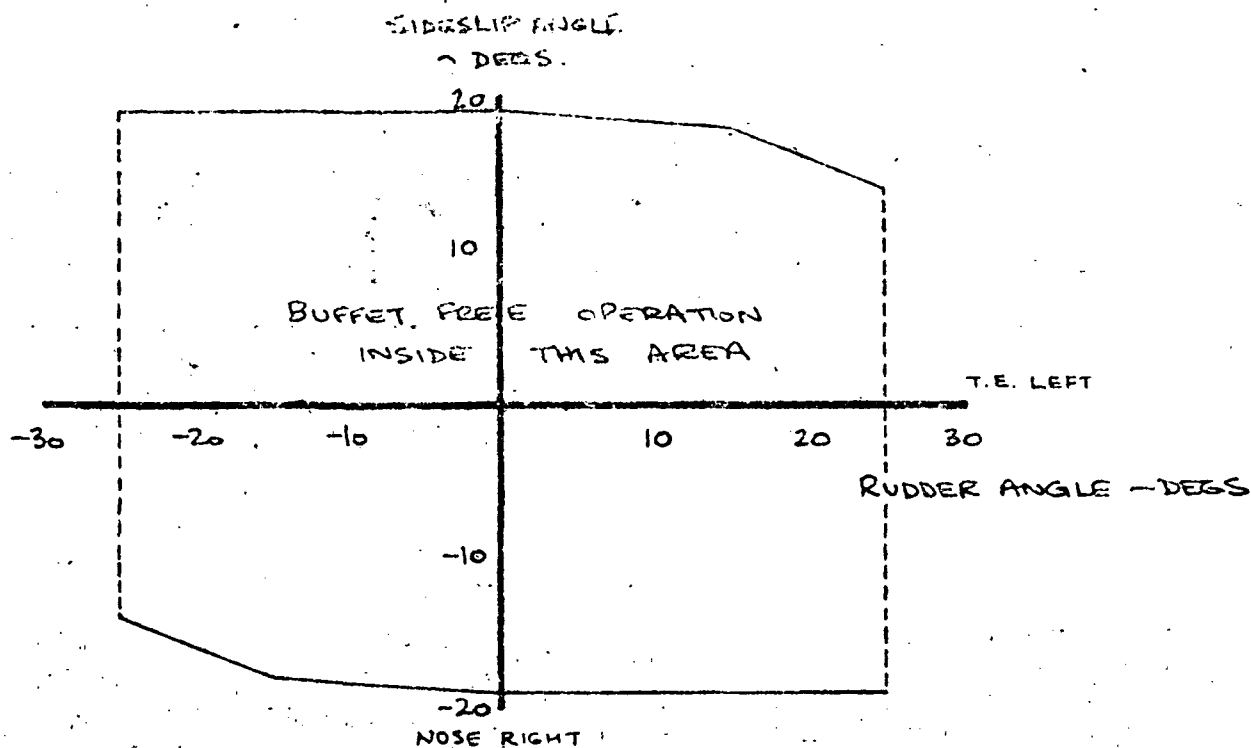
$$NS = \bar{Q} \cdot S \cdot W \cdot BW \cdot CNTOT + Y_E \cdot (\bar{M}_R - \bar{M}_L) \cdot V_A / 32.2$$



ANALYTIC EXPRESSIONS  
IN  $(\alpha_F, C_J, \delta_F)$

MOD C-BA

|       |        |          |         |      |                            |             |
|-------|--------|----------|---------|------|----------------------------|-------------|
| CALC  | RUMSEY | 11/20/70 | REVISED | DATE | APPRODYNAMIC DATA BUILD UP | FIG 3-6     |
| CHECK |        |          |         |      | YAWING MOMENT EQUATION     | DG-27,065TR |
| APPD  |        |          |         |      | BOEING AIRPLANE COMPANY    | PAGE 47     |
| APPD  |        |          |         |      |                            |             |

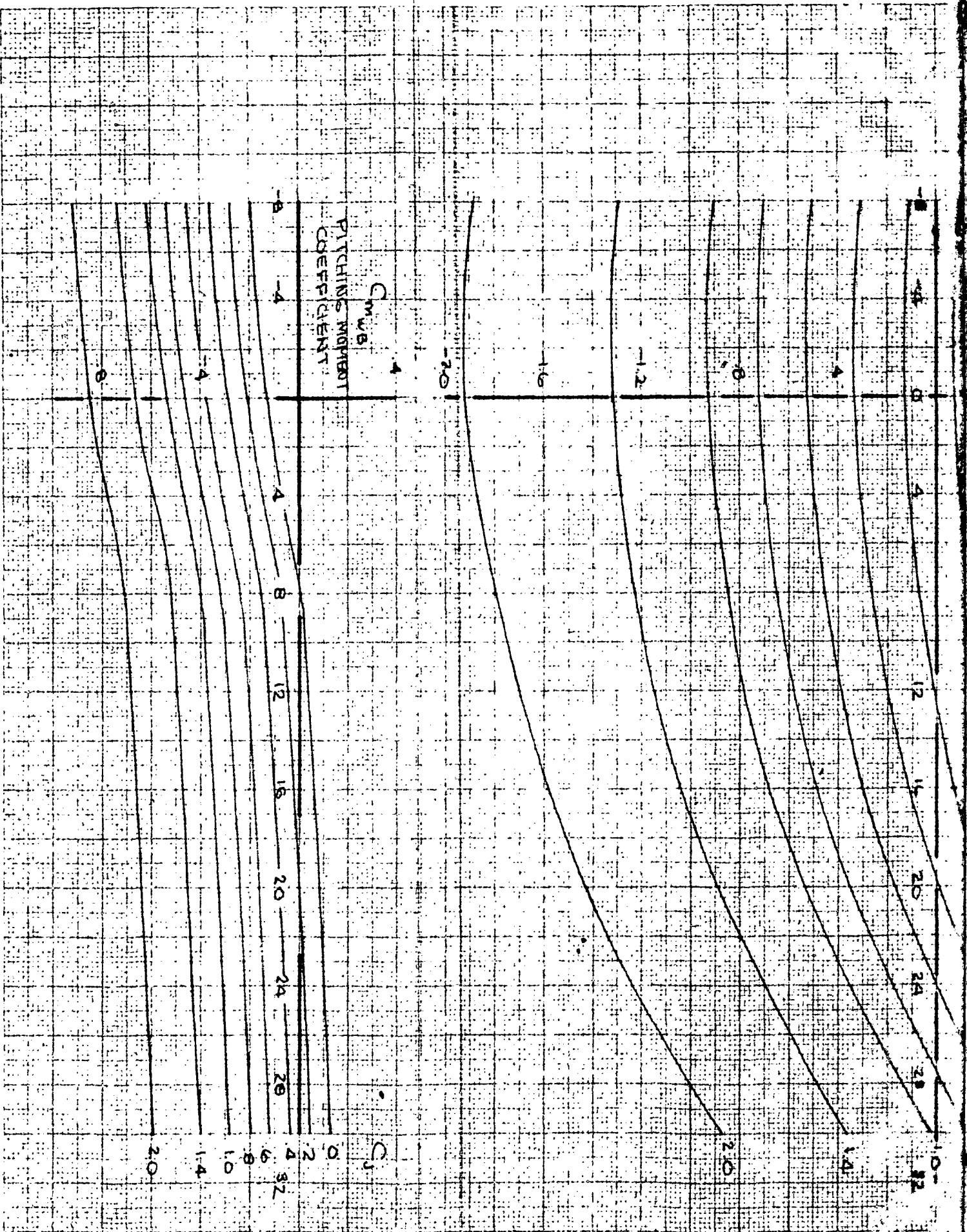
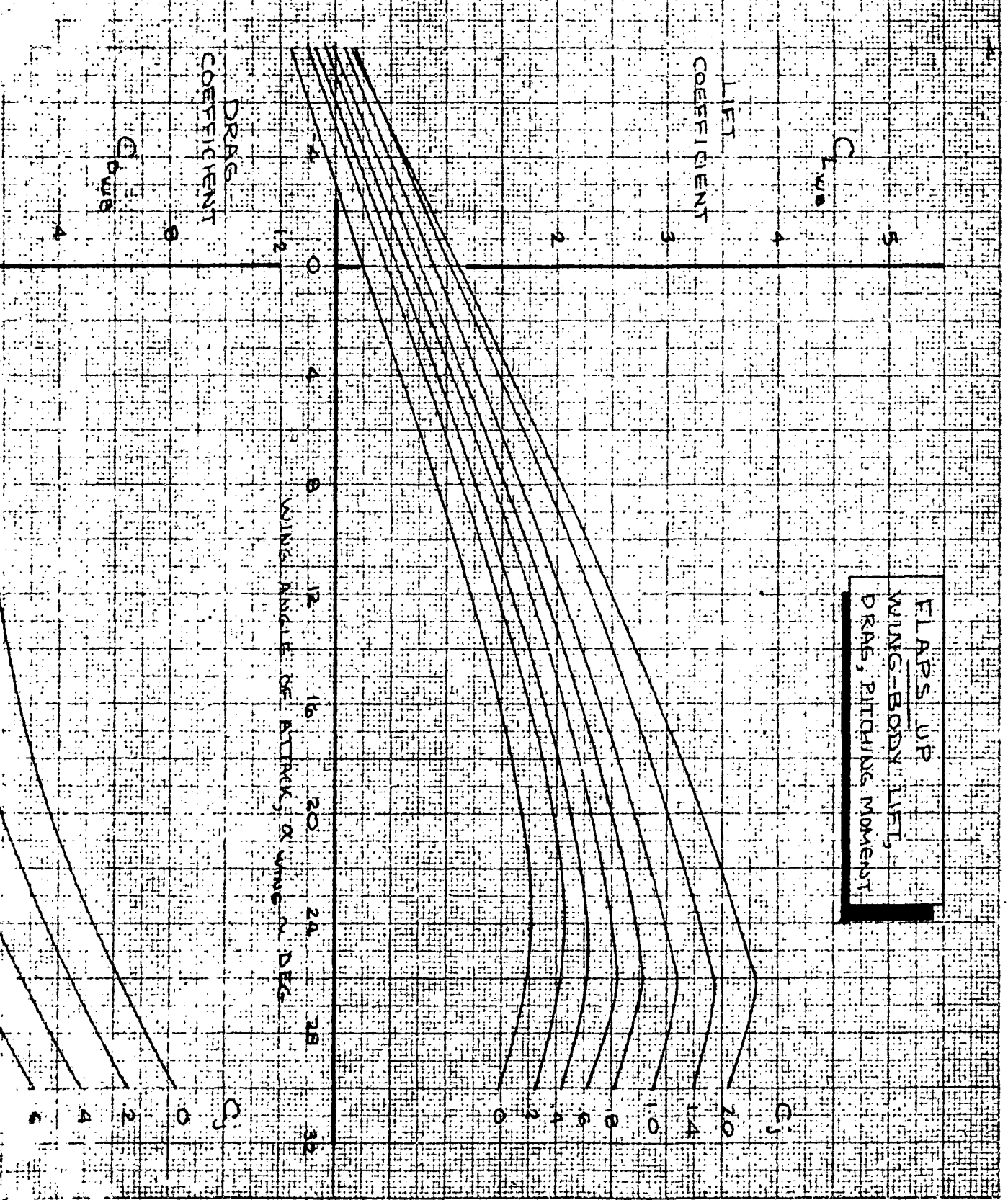


MOD C-8A

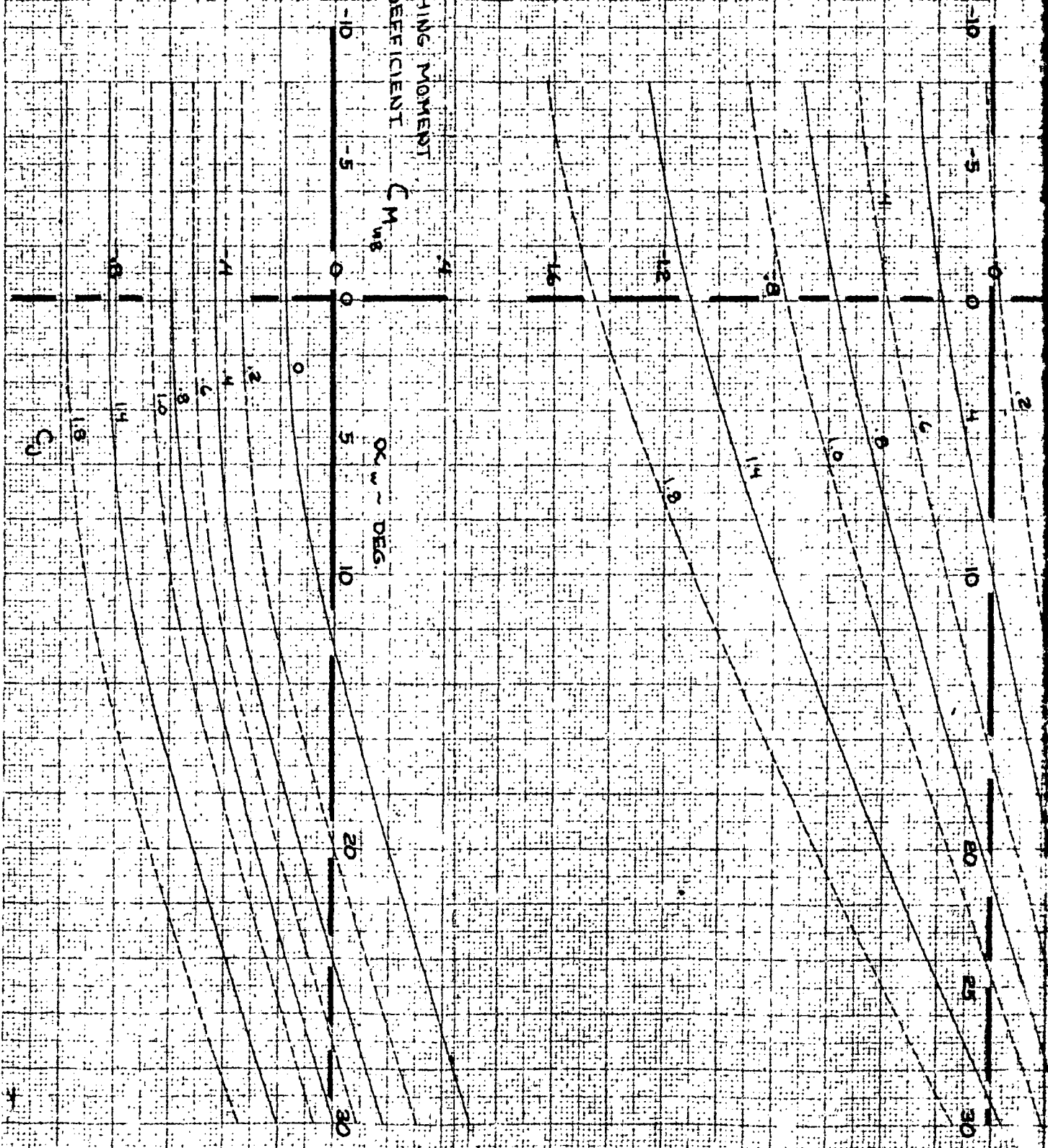
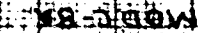
|       |        |         |         |      |                                                                                                      |            |
|-------|--------|---------|---------|------|------------------------------------------------------------------------------------------------------|------------|
| CALC  | RUMSEY | 9/20/70 | REVISED | DATE | WARNING BOUNDARIES<br>HORIZONTAL AND VERTICAL TAIL<br>- SIMULATOR USE ONLY<br><br>THE BOEING COMPANY | FIG 3-7    |
| CHECK |        |         |         |      |                                                                                                      | D6-2606STN |
| APR   |        |         |         |      |                                                                                                      | PAGE       |
| APR   |        |         |         |      |                                                                                                      | 48         |

|                                             |         |         |         |      |
|---------------------------------------------|---------|---------|---------|------|
| CALC                                        | Revised | 6/11/70 | Revised | 0.7E |
| CHECK                                       |         |         |         |      |
| APPD                                        |         |         |         |      |
| APPD                                        |         |         |         |      |
| FLAPS UP (S.G.)                             |         |         |         |      |
| FREE AIR $\delta_{\text{DROOP}} = 30^\circ$ |         |         |         |      |
| THE BOEING COMPANY                          |         |         |         |      |
| FIG 3-8                                     |         |         |         |      |
| 40                                          |         |         |         |      |

TAIL OFF DATA

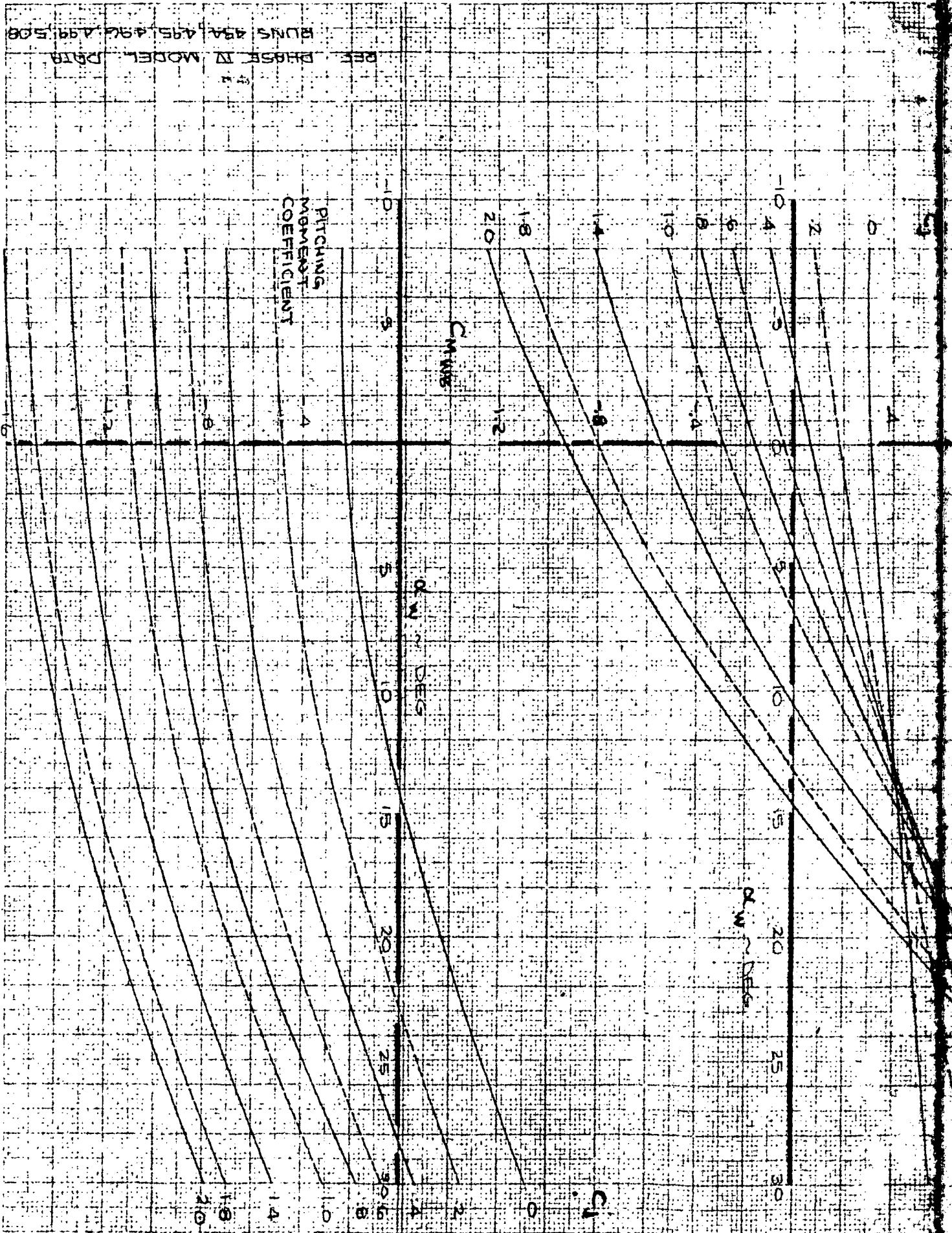
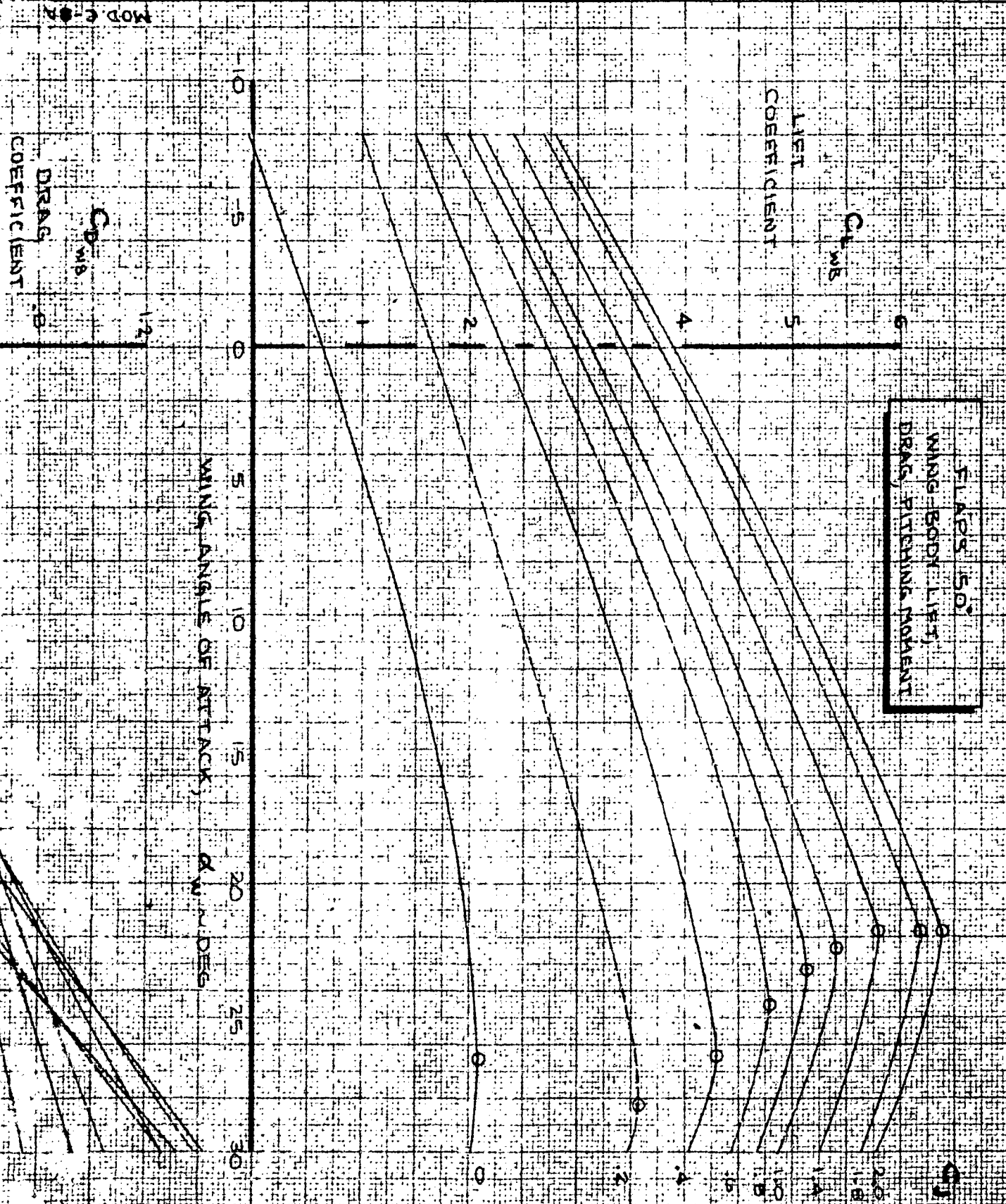


|           |          |            |                    |                    |         |
|-----------|----------|------------|--------------------|--------------------|---------|
| FLAPS 30° | TAIL OFF | CL, CD, CM | FREE AIR           | C <sub>T</sub> = 0 | α = 30° |
| DB-26057M | FIG 3-9  | PAGE 50    | THE BOEING COMPANY |                    |         |



|       |           |        |         |      |
|-------|-----------|--------|---------|------|
| CALC  | ARMSTRONG | 9/2/70 | REVISIO | DATE |
| CHECK | V. FASE   |        |         |      |
| APPD  | V. FASE   |        |         |      |
| APPD  |           |        |         |      |

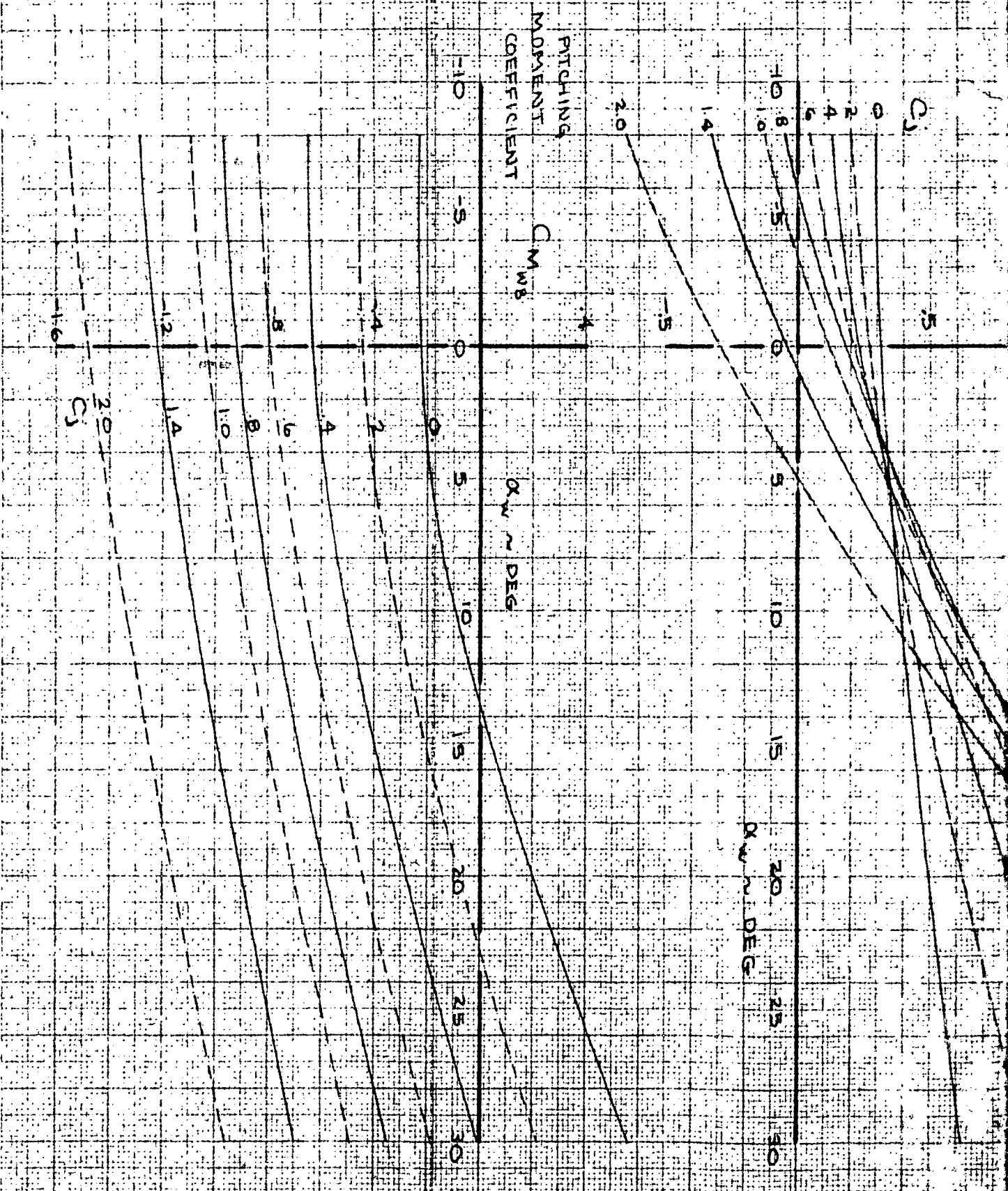
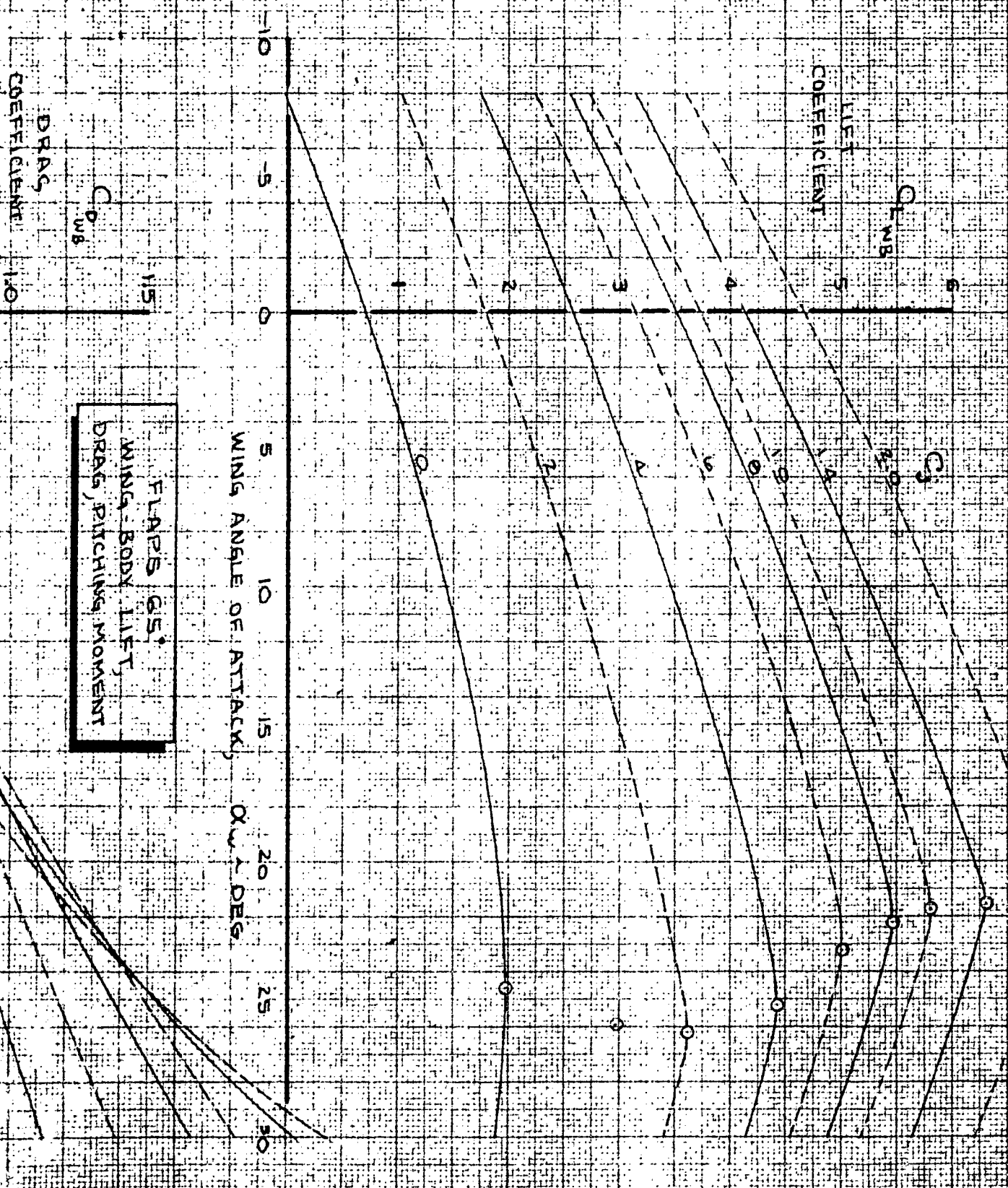
|            |                                |                                                      |                    |
|------------|--------------------------------|------------------------------------------------------|--------------------|
| FLAPS 50°  | TAIL OFF $C_L$ , $C_D$ , $C_M$ | FREE AIR, $C_T = 0$ , $\delta a = 30^\circ$ FIG 3-10 | THE BOEING COMPANY |
| MOD. C-102 | DB-26057N                      | PAGE 1                                               | 51                 |



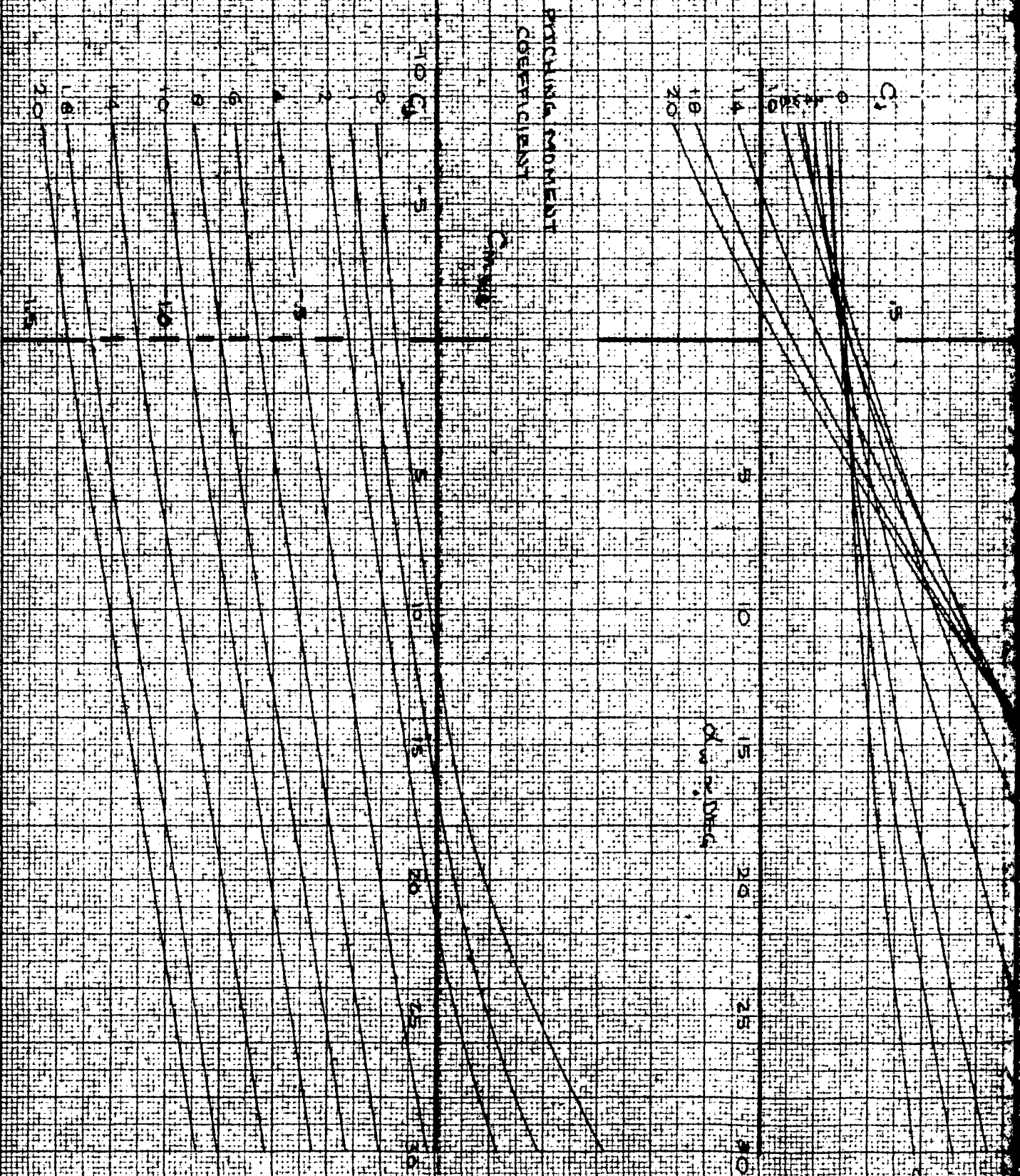
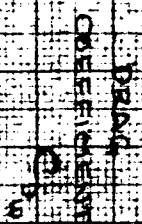
REF. PHASE IV MODEL DATA  
BURNS 45A, 49A, 49B, 49C, 49D, 49E, 49F, 49G, 49H, 49I, 49J, 49K, 49L, 49M, 49N, 49O, 49P, 49Q, 49R, 49S, 49T, 49U, 49V, 49W, 49X, 49Y, 49Z, 49AA, 49AB, 49AC, 49AD, 49AE, 49AF, 49AG, 49AH, 49AI, 49AJ, 49AK, 49AL, 49AM, 49AN, 49AO, 49AP, 49AQ, 49AR, 49AS, 49AT, 49AU, 49AV, 49AW, 49AX, 49AY, 49AZ, 49BA, 49BB, 49BC, 49BD, 49BE, 49BF, 49BG, 49BH, 49BI, 49BJ, 49BK, 49BL, 49BM, 49BN, 49BO, 49BP, 49BQ, 49BR, 49BS, 49BT, 49BU, 49BV, 49BW, 49BX, 49BY, 49BZ, 49CA, 49CB, 49CC, 49CD, 49CE, 49CF, 49CG, 49CH, 49CI, 49CJ, 49CK, 49CL, 49CM, 49CN, 49CO, 49CP, 49CQ, 49CR, 49CS, 49CT, 49CU, 49CV, 49CW, 49CX, 49CY, 49CZ, 49DA, 49DB, 49DC, 49DD, 49DE, 49DF, 49DG, 49DH, 49DI, 49DJ, 49DK, 49DL, 49DM, 49DN, 49DO, 49DP, 49DQ, 49DR, 49DS, 49DT, 49DU, 49DV, 49DW, 49DX, 49DY, 49DZ, 49EA, 49EB, 49EC, 49ED, 49EE, 49EF, 49EG, 49EH, 49EI, 49EJ, 49EK, 49EL, 49EM, 49EN, 49EO, 49EP, 49EQ, 49ER, 49ES, 49ET, 49EU, 49EV, 49EW, 49EX, 49EY, 49EZ, 49FA, 49FB, 49FC, 49FD, 49FE, 49FF, 49FG, 49FH, 49FI, 49FJ, 49FK, 49FL, 49FM, 49FN, 49FO, 49FP, 49FQ, 49FR, 49FS, 49FT, 49FU, 49FV, 49FW, 49FX, 49FY, 49FZ, 49GA, 49GB, 49GC, 49GD, 49GE, 49GF, 49GG, 49GH, 49GI, 49GJ, 49GK, 49GL, 49GM, 49GN, 49GO, 49GP, 49GQ, 49GR, 49GS, 49GT, 49GU, 49GV, 49GW, 49GX, 49GY, 49GZ, 49HA, 49HB, 49HC, 49HD, 49HE, 49HF, 49HG, 49HH, 49HI, 49HJ, 49HK, 49HL, 49HM, 49HN, 49HO, 49HP, 49HQ, 49HR, 49HS, 49HT, 49HU, 49HV, 49HW, 49HX, 49HY, 49HZ, 49IA, 49IB, 49IC, 49ID, 49IE, 49IF, 49IG, 49IH, 49II, 49IJ, 49IK, 49IL, 49IM, 49IN, 49IO, 49IP, 49IQ, 49IR, 49IS, 49IT, 49IU, 49IV, 49IW, 49IX, 49IY, 49IZ, 49JA, 49JB, 49JC, 49JD, 49JE, 49JF, 49JG, 49JH, 49JI, 49JJ, 49JK, 49JL, 49JM, 49JN, 49JO, 49JP, 49JQ, 49JR, 49JS, 49JT, 49JU, 49JV, 49JW, 49JX, 49JY, 49JZ, 49KA, 49KB, 49KC, 49KD, 49KE, 49KF, 49KG, 49KH, 49KI, 49KJ, 49KK, 49KL, 49KM, 49KN, 49KO, 49KP, 49KQ, 49KR, 49KS, 49KT, 49KU, 49KV, 49KW, 49KX, 49KY, 49KZ, 49LA, 49LB, 49LC, 49LD, 49LE, 49LF, 49LG, 49LH, 49LI, 49LJ, 49LK, 49LL, 49LM, 49LN, 49LO, 49LP, 49LQ, 49LR, 49LS, 49LT, 49LU, 49LV, 49LW, 49LX, 49LY, 49LZ, 49MA, 49MB, 49MC, 49MD, 49ME, 49MF, 49MG, 49MH, 49MI, 49MJ, 49MK, 49ML, 49MM, 49MN, 49MO, 49MP, 49MQ, 49MR, 49MS, 49MT, 49MU, 49MV, 49MW, 49MX, 49MY, 49MZ, 49NA, 49NB, 49NC, 49ND, 49NE, 49NF, 49NG, 49NH, 49NI, 49NJ, 49NK, 49NL, 49NM, 49NO, 49NP, 49NQ, 49NR, 49NS, 49NT, 49NU, 49NV, 49NW, 49NX, 49NY, 49NZ, 49OA, 49OB, 49OC, 49OD, 49OE, 49OF, 49OG, 49OH, 49OI, 49OJ, 49OK, 49OL, 49OM, 49ON, 49OO, 49OP, 49OQ, 49OR, 49OS, 49OT, 49OU, 49OV, 49OW, 49OX, 49OY, 49OZ, 49PA, 49PB, 49PC, 49PD, 49PE, 49PF, 49PG, 49PH, 49PI, 49PJ, 49PK, 49PL, 49PM, 49PN, 49PO, 49PP, 49PQ, 49PR, 49PS, 49PT, 49PU, 49PV, 49PW, 49PX, 49PY, 49PZ, 49QA, 49QB, 49QC, 49QD, 49QE, 49QF, 49QG, 49QH, 49QI, 49QJ, 49QK, 49QL, 49QM, 49QN, 49QO, 49QP, 49QQ, 49QR, 49QS, 49QT, 49QU, 49QV, 49QW, 49QX, 49QY, 49QZ, 49RA, 49RB, 49RC, 49RD, 49RE, 49RF, 49RG, 49RH, 49RI, 49RJ, 49RK, 49RL, 49RM, 49RN, 49RO, 49RP, 49RQ, 49RR, 49RS, 49RT, 49RU, 49RV, 49RW, 49RX, 49RY, 49RZ, 49SA, 49SB, 49SC, 49SD, 49SE, 49SF, 49SG, 49SH, 49SI, 49SJ, 49SK, 49SL, 49SM, 49SN, 49SO, 49SP, 49SQ, 49SR, 49SS, 49ST, 49SU, 49SV, 49SW, 49SX, 49SY, 49SZ, 49TA, 49TB, 49TC, 49TD, 49TE, 49TF, 49TG, 49TH, 49TI, 49TJ, 49TK, 49TL, 49TM, 49TN, 49TO, 49TP, 49TQ, 49TR, 49TS, 49TT, 49TU, 49TV, 49TW, 49TX, 49TY, 49TZ, 49UA, 49UB, 49UC, 49UD, 49UE, 49UF, 49UG, 49UH, 49UI, 49UJ, 49UK, 49UL, 49UM, 49UN, 49UO, 49UP, 49UQ, 49UR, 49US, 49UT, 49UU, 49UV, 49UW, 49UX, 49UY, 49UZ, 49VA, 49VB, 49VC, 49VD, 49VE, 49VF, 49VG, 49VH, 49VI, 49VJ, 49VK, 49VL, 49VM, 49VN, 49VO, 49VP, 49VQ, 49VR, 49VS, 49VT, 49VU, 49VV, 49VW, 49VX, 49VY, 49VZ, 49WA, 49WB, 49WC, 49WD, 49WE, 49WF, 49WG, 49WH, 49WI, 49WJ, 49WK, 49WL, 49WM, 49WN, 49WO, 49WP, 49WQ, 49WR, 49WS, 49WT, 49WU, 49WV, 49WW, 49WX, 49WY, 49WZ, 49XA, 49XB, 49XC, 49XD, 49XE, 49XF, 49XG, 49XH, 49XI, 49XJ, 49XK, 49XL, 49XM, 49XN, 49XO, 49XP, 49XQ, 49XR, 49XS, 49XT, 49XU, 49XV, 49XW, 49XX, 49XY, 49XZ, 49YA, 49YB, 49YC, 49YD, 49YE, 49YF, 49YG, 49YH, 49YI, 49YJ, 49YK, 49YL, 49YM, 49YN, 49YO, 49YP, 49YQ, 49YR, 49YS, 49YT, 49YU, 49YV, 49YW, 49YX, 49YY, 49YZ, 49ZA, 49ZB, 49ZC, 49ZD, 49ZE, 49ZF, 49ZG, 49ZH, 49ZI, 49ZJ, 49ZK, 49ZL, 49ZM, 49ZN, 49ZO, 49ZP, 49ZQ, 49ZR, 49ZS, 49ZT, 49ZU, 49ZV, 49ZW, 49ZX, 49ZY, 49ZZ

|                                                                                   |        |          |         |      |
|-----------------------------------------------------------------------------------|--------|----------|---------|------|
| CALC                                                                              | V PAGE | 11/1/70  | REVISED | DATE |
| CHECK                                                                             |        |          |         |      |
| APPD                                                                              |        |          |         |      |
| APPD                                                                              |        |          |         |      |
| FLAPS 65°<br>TAIL OFF $C_L, C_D, C_M$<br>FREE AIR $C_T = 0$<br>THE BOEING COMPANY |        |          |         |      |
| PAGE 52                                                                           |        | FIG 3-11 |         |      |
| DL-26065M                                                                         |        |          |         |      |

MOD C-82



FL8375  
Tail off 1/8 in  
Free Air 170.5 = 34  
THE BOBIN COMPANY

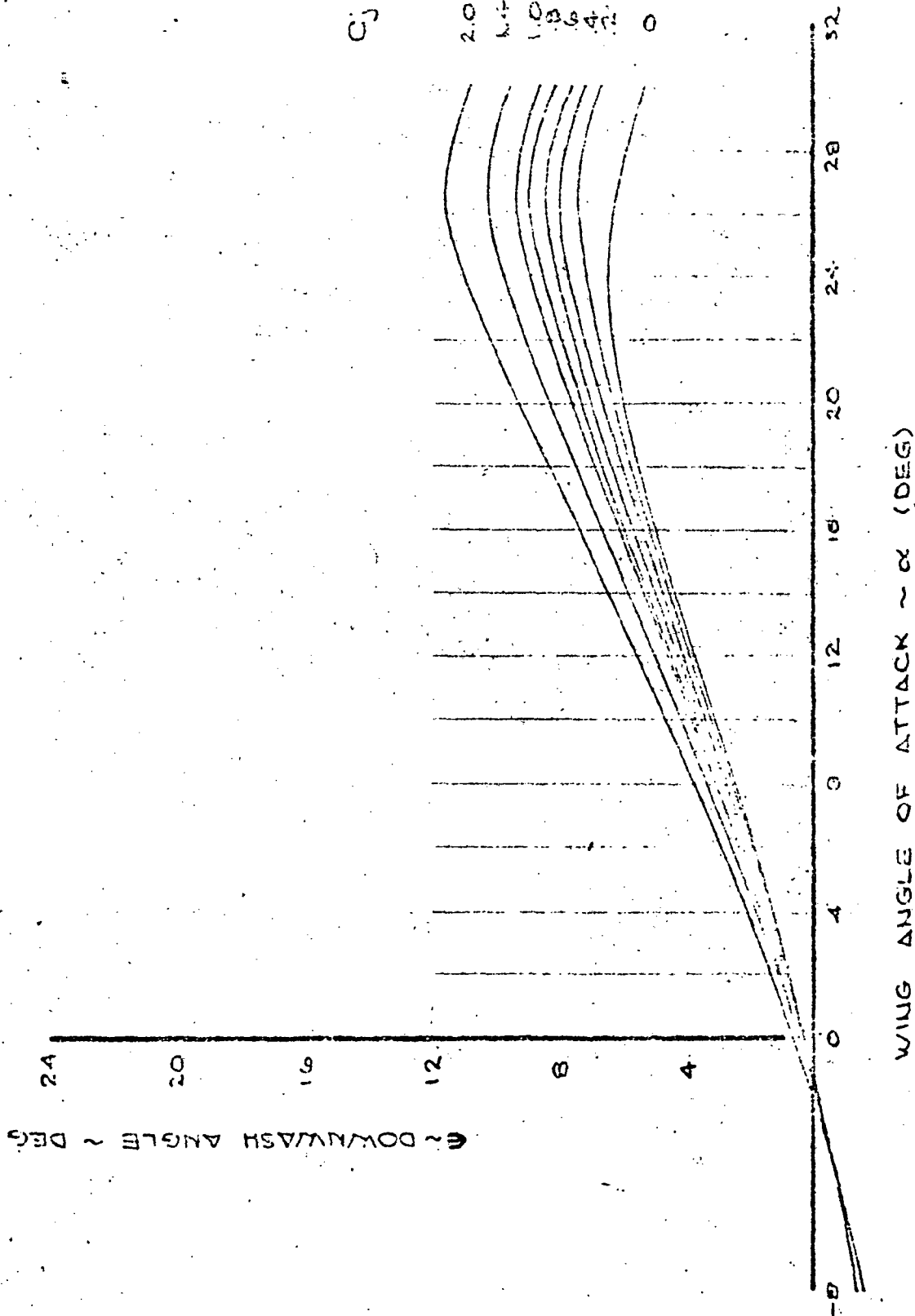


REF: 11/12/70

FREE AIR

FLAPS UP

- 1) SIMULATOR (11/10) DOWNWASH MODEL
- 2) VORTEX POSITION PER TABLE



MOD C-8A

|       |     |         |         |      |                                                   |            |
|-------|-----|---------|---------|------|---------------------------------------------------|------------|
| CALC  | ADD | 11/3/70 | REVISED | DATE | COMPUTED DOWNWASH<br>FLAPS UP<br>(REFERENCE ONLY) | D6-2606STN |
| CHECK |     |         |         |      |                                                   |            |
| APR   |     |         |         |      |                                                   | FIG 3-13   |
| APR   |     |         |         |      |                                                   | PAGE 54    |
|       |     |         |         |      | THE BOEING COMPANY                                |            |

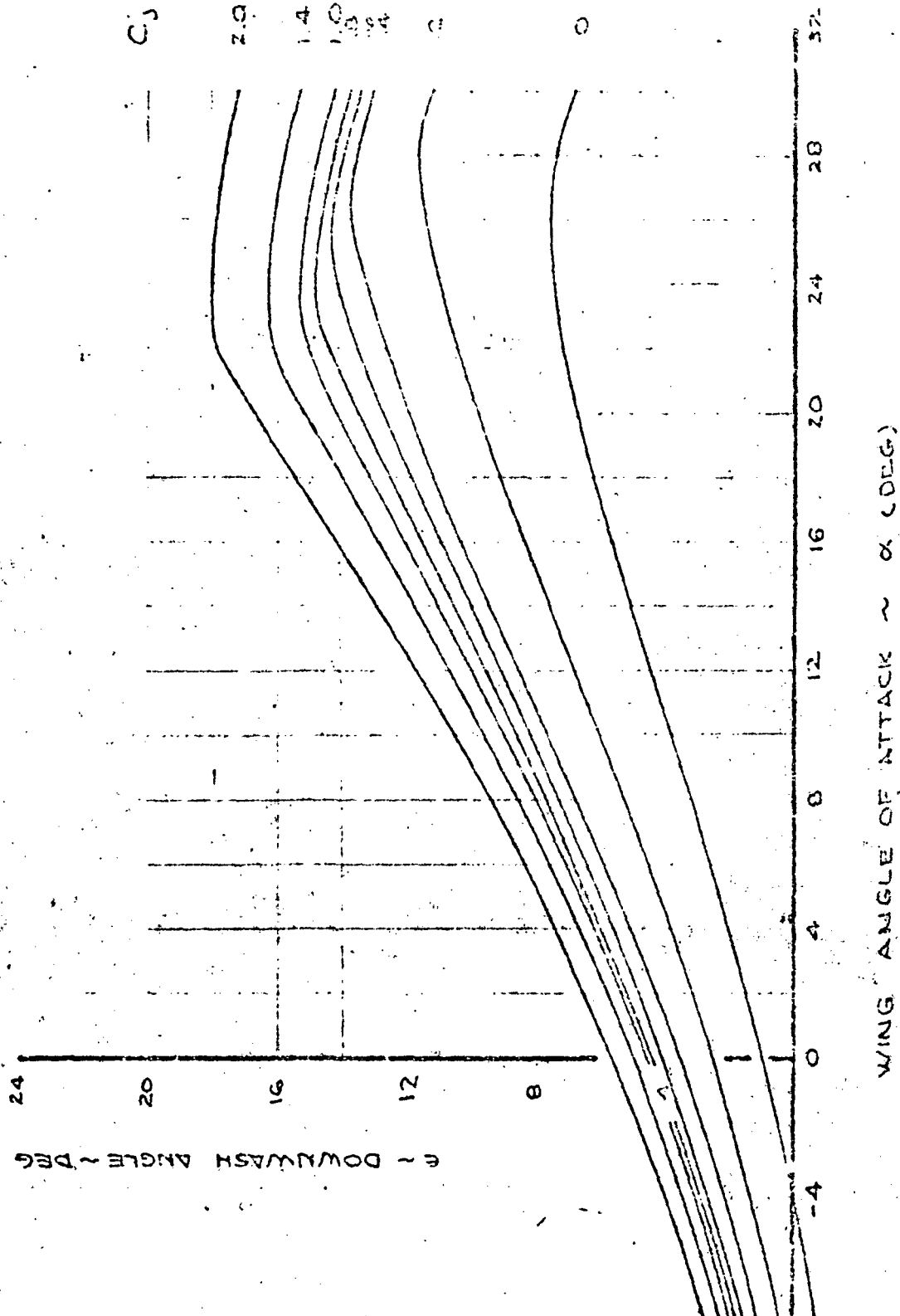
K-E ALPHAS 11/12/70

REF: 11/12/70

FREE AIR

FLAPS 30°

- 1) SIMULATOR (1/170) DOWNWASH MODEL
- 2) VORTEX POSITION PER TABLE



MOD C-8A

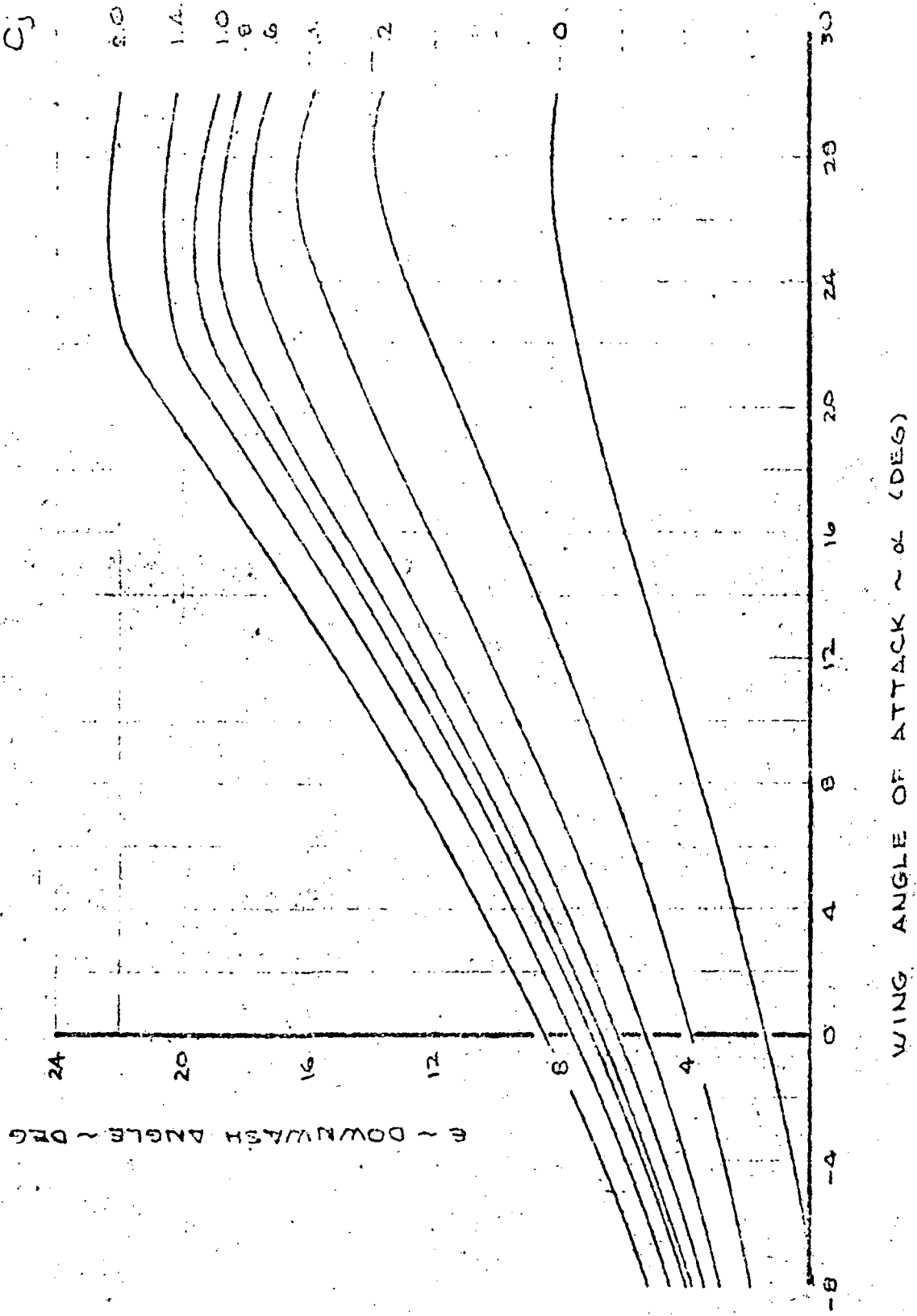
|       |     |          |         |      |                                                    |            |
|-------|-----|----------|---------|------|----------------------------------------------------|------------|
| CALC  | LDO | 11/13/70 | REVISED | DATE | COMPUTED DOWNWASH<br>FLAPS 30°<br>(REFERENCE ONLY) | DB-26065TN |
| CHECK |     |          |         |      |                                                    | FIG 3-1A   |
| APP   |     |          |         |      | THE BOEING COMPANY                                 | PAGE       |
| APP   |     |          |         |      |                                                    | 55         |

REP: 11/12/70

FREE AIR

FLAPS 50°

- 1) SIMULATOR (W/16) DOWNWASH MODEL
- 2) VORTEX POSITION PER TABLE



MOD C-8A

|       |     |          |         |      |
|-------|-----|----------|---------|------|
| CALC  | LDC | 11/12/70 | REVISED | DATE |
| CHECK |     |          |         |      |
| APR   |     |          |         |      |
| APR   |     |          |         |      |

COMPUTED DOWNWASH  
FLAPS 50°  
(REFERENCE ONLY)

THE BOEING COMPANY

06-26065TN  
FIG 3-15  
PAGE 56

FREE AIR

FLAPS 65°

α ~ DOWNWASH ANGLE ~ DEG

0.2 0.4 0.6 0.8 1.0 1.2 1.4 1.6 1.8 2.0

24 20 16 12 8 4

WING ANGLE OF ATTACK ~ α (DEG)

REF: 11/12/70

MOD C-8A

|       |  |         |           |      |
|-------|--|---------|-----------|------|
| CALC  |  | 1-13-70 | PERFORMED | DATE |
| CHECK |  |         |           |      |
| APR   |  |         |           |      |
| APR   |  |         |           |      |

COMPUTED DOWNWASH  
FLAPS 65°  
(REFERENCE ONLY)

THE BOEING COMPANY

DB-26065T12

FIG 3-16

PAGE

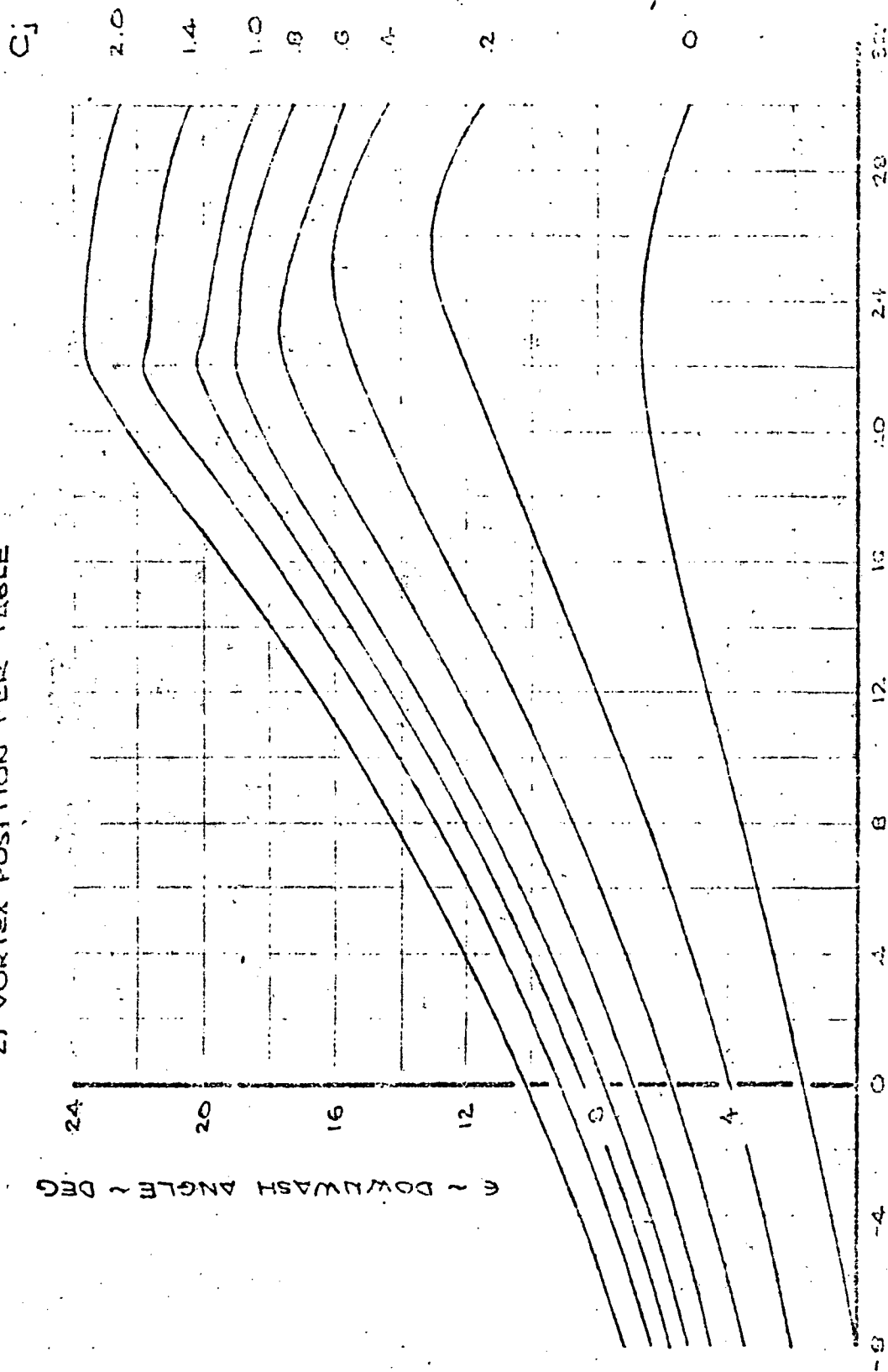
57

REF: 11/12/70

FREE AIR

FLAPS 75°

- 1) SIMULATOR (11/10) DOWNWASH MODEL
- 2) VORTEX POSITION PER TABLE



MOD C-8A

|       |     |          |         |      |
|-------|-----|----------|---------|------|
| CALC  | LDO | 11/13/70 | REVISED | DATE |
| CHECK |     |          |         |      |
| APR   |     |          |         |      |
| APR   |     |          |         |      |
|       |     |          |         |      |

COMPUTED DOWNWASH  
FLAPS 75°  
(REFERENCE ONLY)

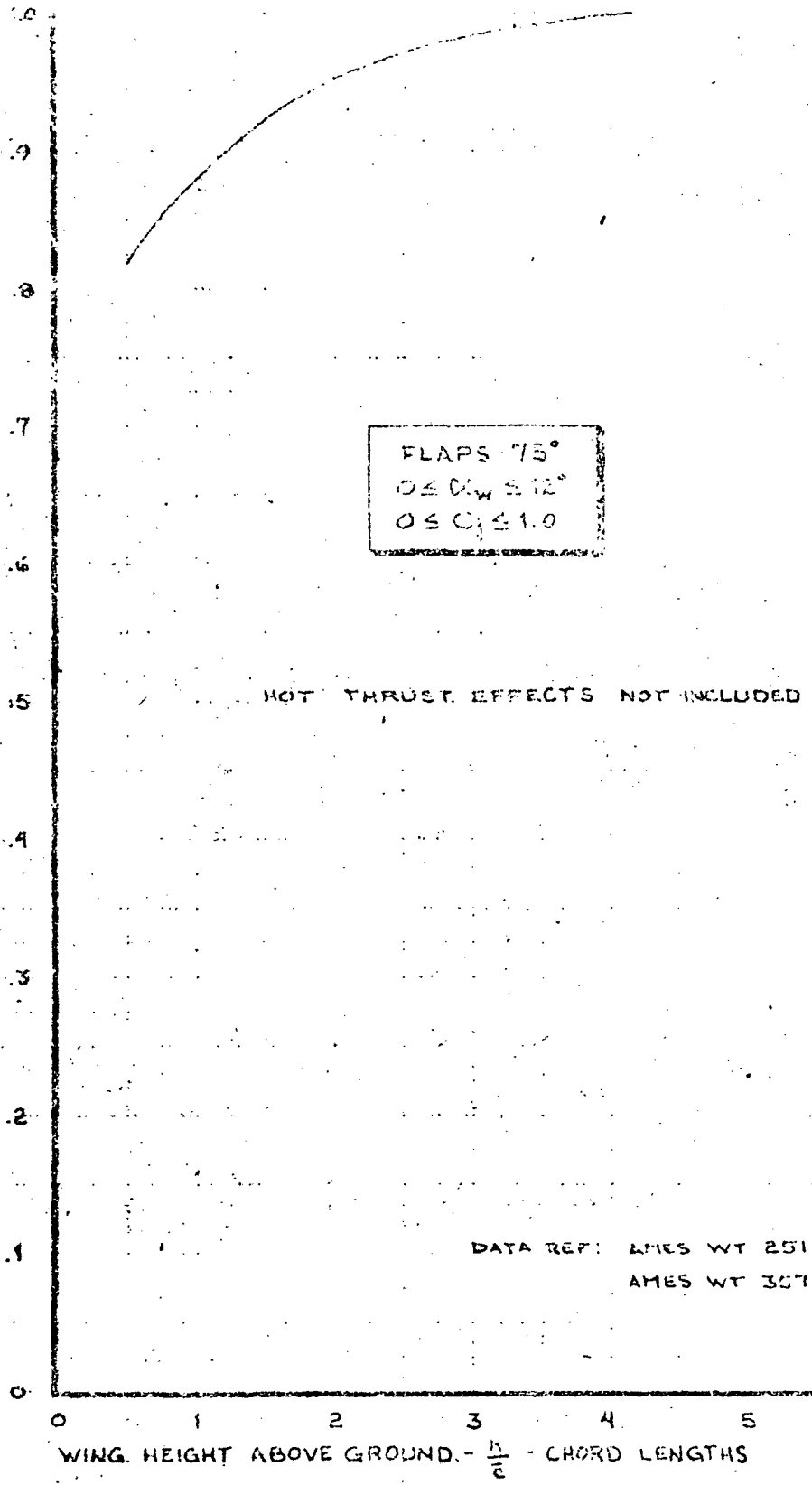
THE BOEING COMPANY

D6-2606STN

FIG 3-17

PAGE  
58

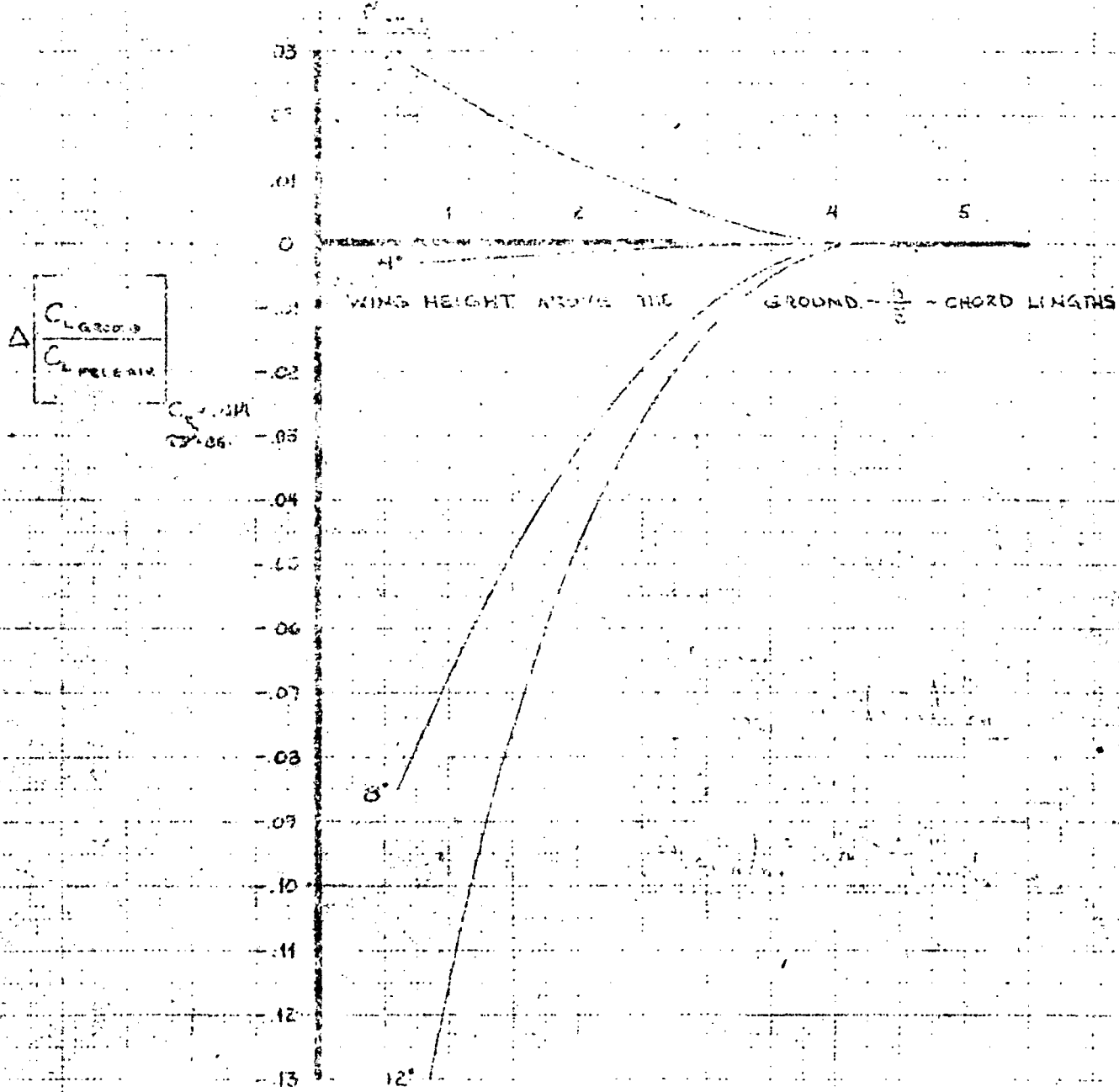
$$\frac{C_{L \text{ GROUND}}}{C_{L \text{ FREE AIR}}} = C_{r, \text{EO}}$$



MOD C-8A

|       |          |         |         |      |                                                                                        |                                           |
|-------|----------|---------|---------|------|----------------------------------------------------------------------------------------|-------------------------------------------|
| CALC  | VERIFIED | 2/21/70 | REVISED | DATE | GROUND EFFECT ON WING-BODY LIFT COEFFICIENT<br><br>FLAPS 75°<br><br>THE BOEING COMPANY | 06-2666STN<br><br>FIG 3-18<br><br>PAGE 59 |
| CHECK |          |         |         |      |                                                                                        |                                           |
| APR   |          |         |         |      |                                                                                        |                                           |
| APR   |          |         |         |      |                                                                                        |                                           |
|       |          |         |         |      |                                                                                        |                                           |

FIG 3-17



MOD C-8A

|       |      |         |      |                                   |            |
|-------|------|---------|------|-----------------------------------|------------|
| CALC  | DATE | REVISED | DATE | VECTOR OF HOT THROST INTERFERENCE | D6-2606ST. |
| CHECK |      |         |      | ON WING-BODY LIFT IN              |            |
| APP   |      |         |      | GROUND EFFECT                     | FIG 3-17   |
| APP   |      |         |      | FLAPS 75°                         |            |
|       |      |         |      | THE BOEING COMPANY                | 60         |

$C_{D, \text{GROUND}}$   
 $C_{D, \text{FREE AIR}}$

$C_{D, 0}$

FLAPS 75°  
 $0 \leq \alpha \leq 10^\circ$

$C_{D, 0.27}$

HOT THRUST EFFECTS NOT INCLUDED

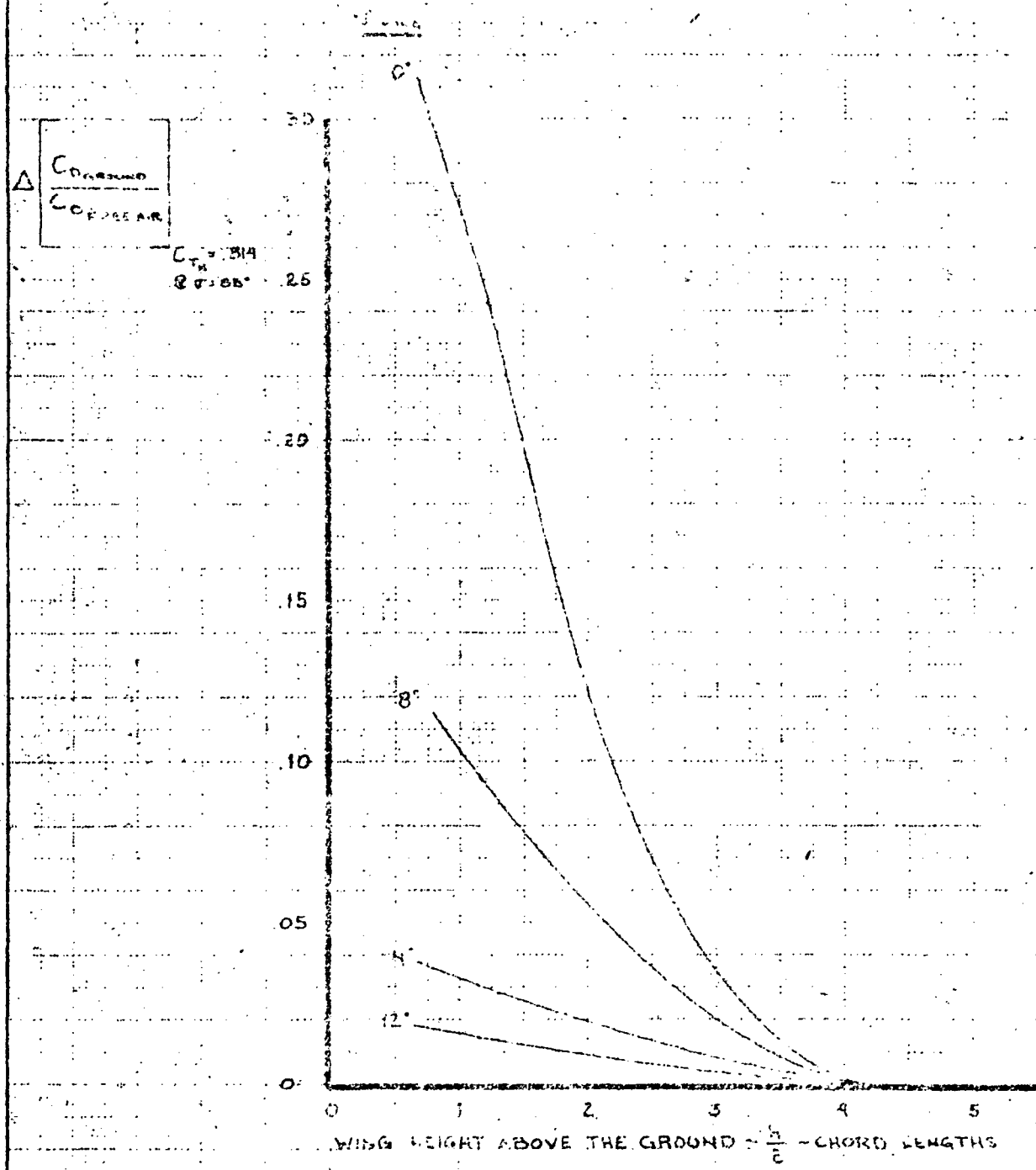
DATA IN FLAMES 251

AMES 357

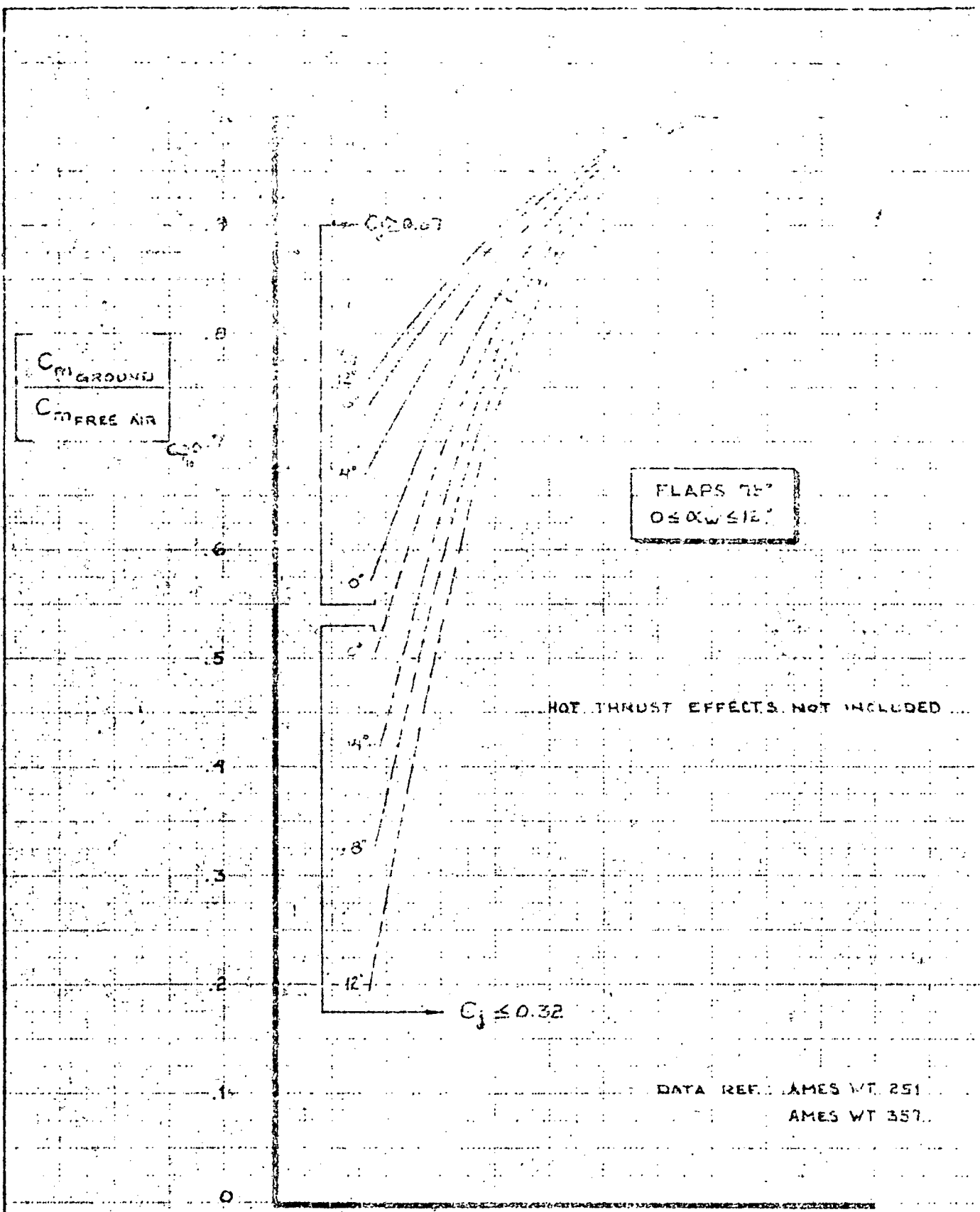
WING HEIGHT ABOVE GROUND -  $\frac{h}{c}$  - CHORD LENGTHS

MOD C-8A

|       |          |      |         |      |                                                |            |
|-------|----------|------|---------|------|------------------------------------------------|------------|
| CALC  | DESIGNED | DATE | REVISED | DATE | GROUND EFFECT ON WING-BODY<br>DRAG COEFFICIENT | DE-26065TN |
| CHECK |          |      |         |      |                                                |            |
| APR   |          |      |         |      |                                                |            |
| APR   |          |      |         |      |                                                |            |
|       |          |      |         |      | FLAPS 75°                                      | FIG 3-20   |
|       |          |      |         |      | THE ROEING COMPANY                             | 61         |



|       |      |         |      |                                                                           |            |
|-------|------|---------|------|---------------------------------------------------------------------------|------------|
|       |      |         |      | MOD C-8A                                                                  |            |
| CALC  | DATE | REVISED | DATE | VECTORED NOT THRUST INTERFERENCE<br>OR WING-BODY DRAG IN GROUND<br>EFFECT | DC-2606ST. |
| CHECK |      |         |      |                                                                           |            |
| APR   |      |         |      |                                                                           | FIG 3-21   |
| APR   |      |         |      |                                                                           | PA 62      |
|       |      |         |      | FLAPS 75°                                                                 |            |
|       |      |         |      | THE BOEING COMPANY                                                        |            |



WING HEIGHT ABOVE GROUND -  $\frac{h}{c}$  - CHORD LENGTHS

DATA REF. AMES WT. 251  
AMES WT. 357.

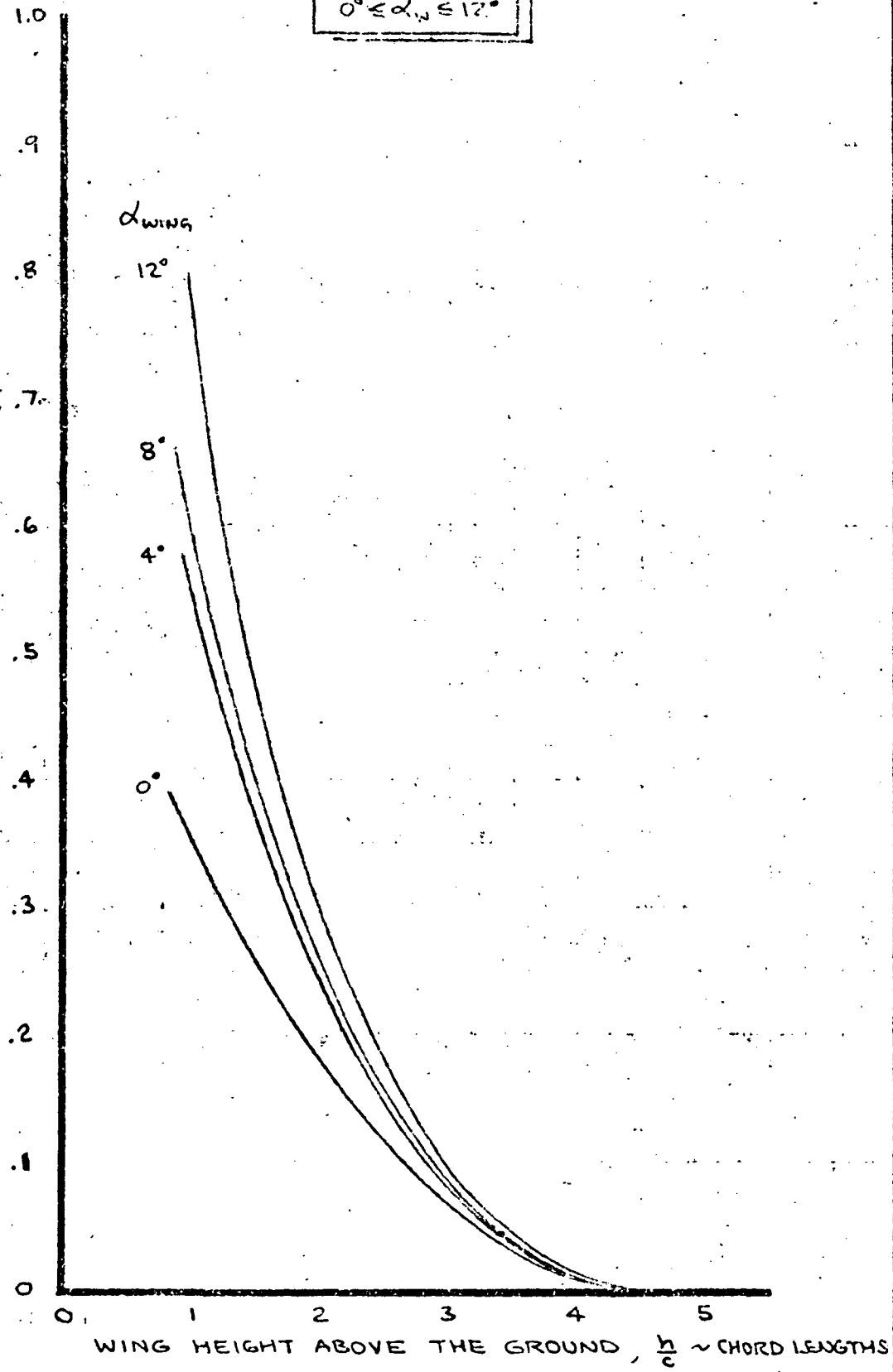
Mod C-8A

|       |          |          |      |                                                        |            |
|-------|----------|----------|------|--------------------------------------------------------|------------|
| CALC  | DESIGNED | REVIEWED | DATE | GROUND EFFECT ON WING-BODY PITCHING MOMENT COEFFICIENT | D6-24065TN |
| CHECK |          |          |      | FLAPS 75°                                              | FIG 3-22   |
| APR   |          |          |      | THE BOEING COMPANY                                     | 63         |
| APR   |          |          |      |                                                        |            |

FLAPS 75°  
 $0^\circ \leq \alpha_w \leq 12^\circ$

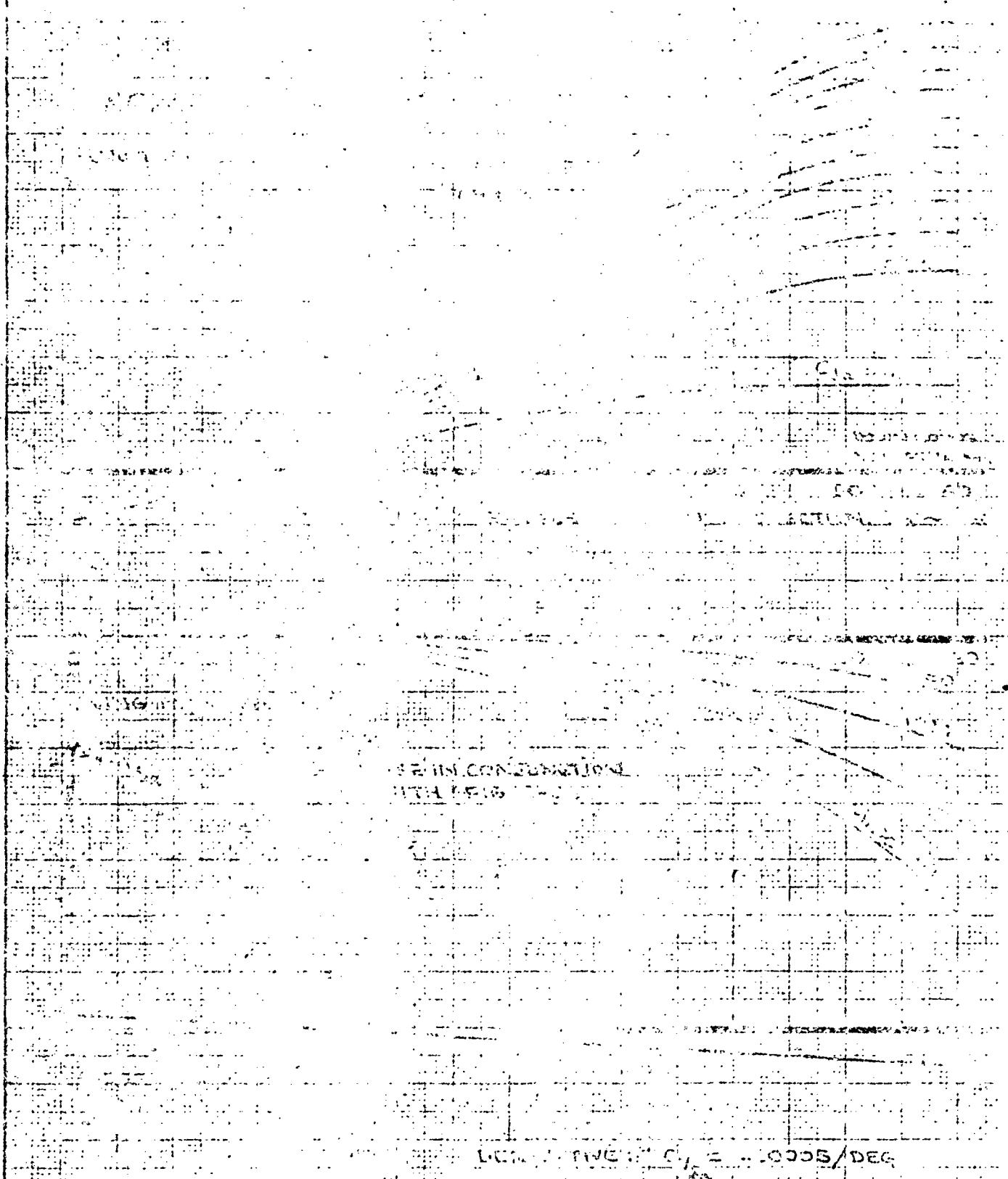
$$\Delta \left[ \frac{C_{M \text{ GROUND}}}{C_{M \text{ FREE AIR}}} \right]$$

$C_T = 0.14$   
 $\nu = 85^\circ$



MOD C-8A

|                    |            |         |         |      |                                                                                                     |            |
|--------------------|------------|---------|---------|------|-----------------------------------------------------------------------------------------------------|------------|
| CALC               | VEDER 0-11 | 1-21-70 | REVISED | DATE | VECTORED HOT THRUST INTERFERENCE<br>ON WING-BODY PITCHING MOMENT<br>IN GROUND EFFECT<br>~ FLAPS 75° | 06-260657N |
| CHECK              |            |         |         |      |                                                                                                     |            |
| APR                |            |         |         |      |                                                                                                     | FIG 3-23   |
| APR                |            |         |         |      |                                                                                                     | PAGE 64    |
| THE BOEING COMPANY |            |         |         |      |                                                                                                     |            |



CL vs alpha

CL vs alpha

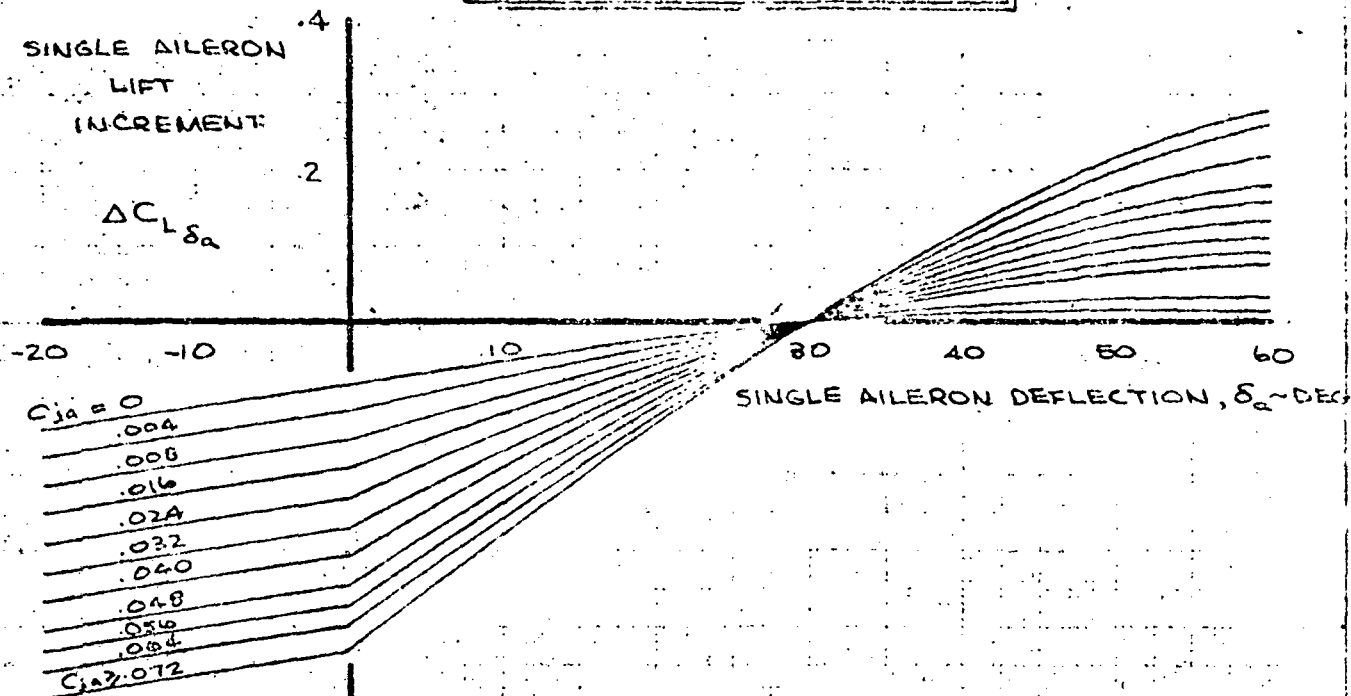
|       |       |      |      |                           |            |
|-------|-------|------|------|---------------------------|------------|
| CALC  | ENTER | DATE | TIME | ALLGON EFFECTIVENESS      | 86-26045TN |
| CHECK |       |      |      | $\alpha_{WING} = 0^\circ$ | FIG 3-2    |
| APR   |       |      |      | THE BOEING COMPANY        | PAGE       |
| APR   |       |      |      |                           | 65         |

# AILERON INTERACTIONS

$\alpha_{wing} = 0^\circ$

SINGLE AILERON  
LIFT  
INCREMENT

$\Delta C_{L\delta_a}$

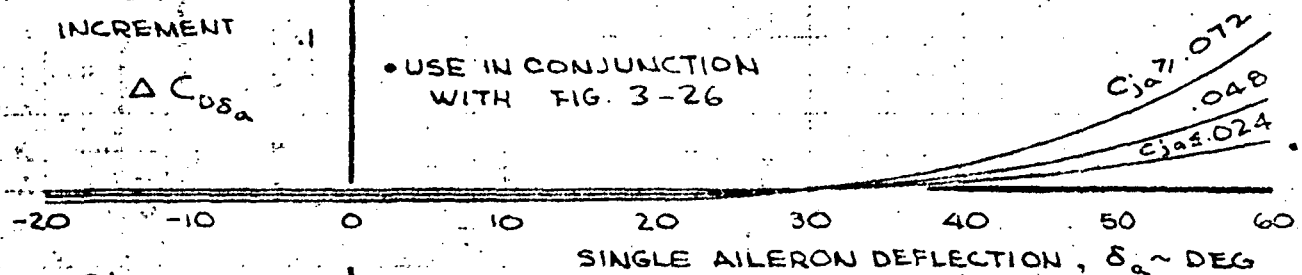


- NOTE:
- USE FOR ALL FLAP SETTINGS
  - $C_{ja} = \frac{m V^2}{q S_{wing}}$  FOR ONE AILERON
  - USE FOR ALL SIDESLIP ANGLES

SINGLE AILERON  
DRAG  
INCREMENT

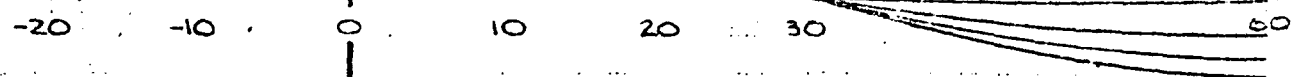
$\Delta C_{D\delta_a}$

• USE IN CONJUNCTION  
WITH FIG. 3-26



SINGLE AILERON  
PITCHING MOMENT  
INCREMENT

$\Delta C_{m\delta_a}$



MOD C-3A

|                      |  |  |  |
|----------------------|--|--|--|
| W.C. SPITZER 9-14-70 |  |  |  |
| CH-10A               |  |  |  |
| 113                  |  |  |  |
| APD                  |  |  |  |

AILERON EFFECTIVENESS

$\alpha_{wing} = 0^\circ$

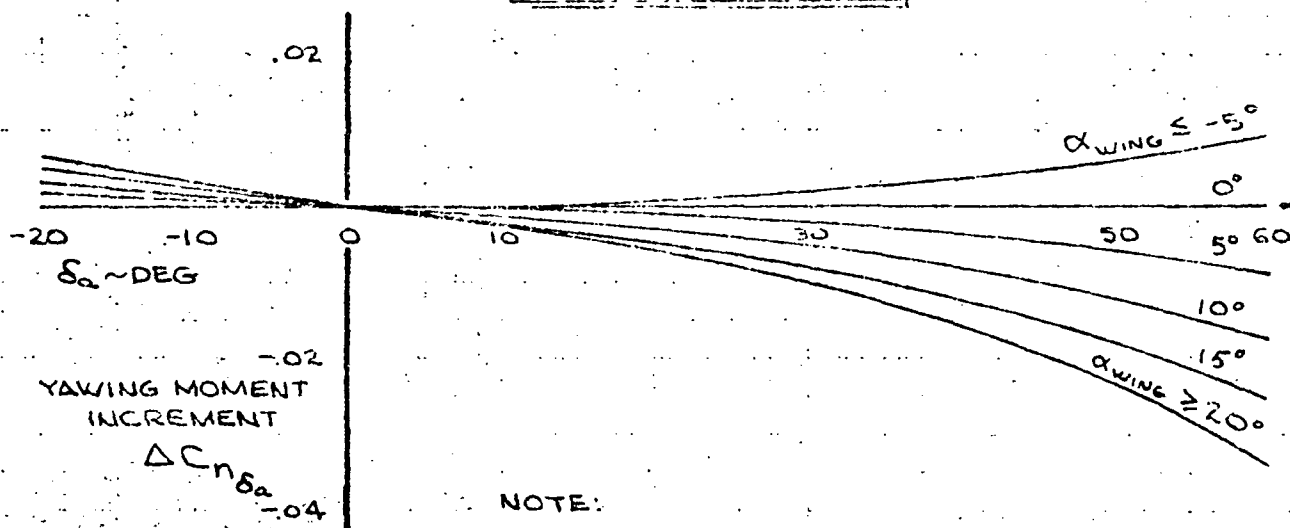
THE FORDING COMPANY

D6-2606STH

FIG 3-23

66

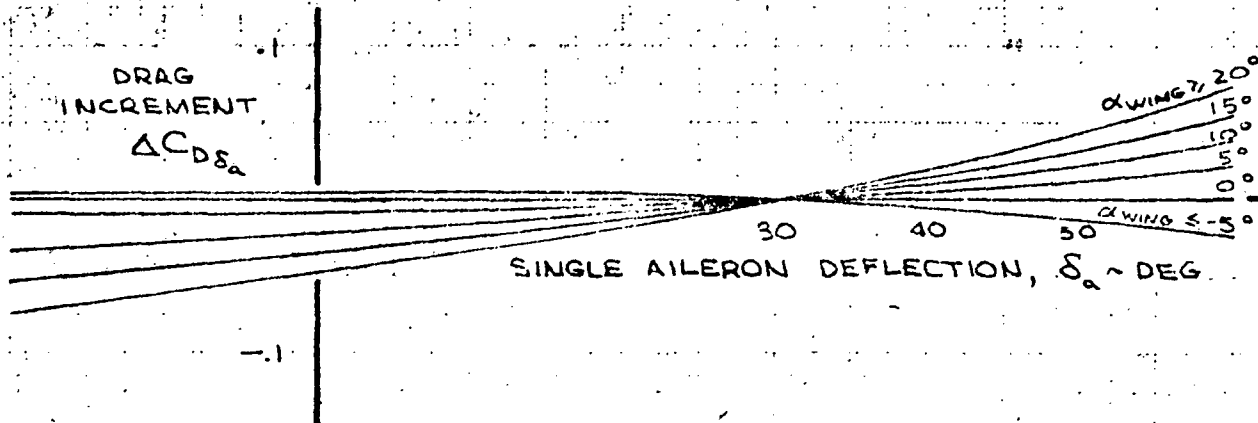
ANGLE OF ATTACK  
EFFECT ON THE  
AILERON



NOTE:

- USE FOR ALL FLAP SETTINGS
- USE FOR ALL  $C_{\mu}$  BLOWING LEVELS

$C_L$ ,  $C_Y$ ,  $C_L$  AND  $C_m$  ARE  
INDEPENDENT OF ANGLE OF ATTACK



Mod C-8A

|      |         |         |      |  |
|------|---------|---------|------|--|
| NAME | SPITZER | 9-14-70 | DATE |  |
| REV  |         |         |      |  |
| APP  |         |         |      |  |

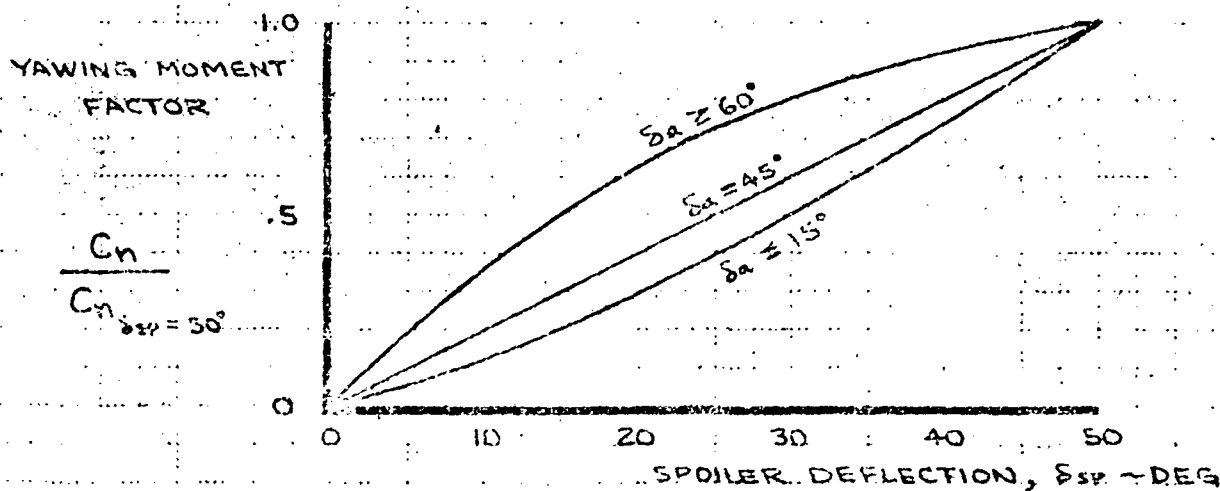
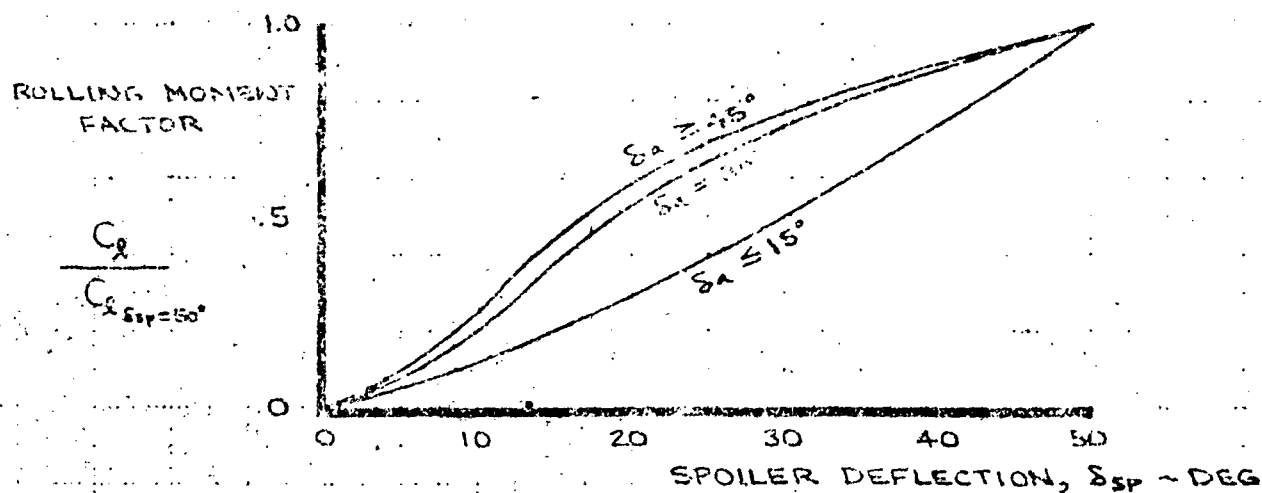
ANGLE of ATTACK EFFECT  
ON THE AILERON

DG-2606STN

FIG 3-26

THE BOEING COMPANY

NOTE: \* USE FOR ALL FLAP SETTINGS  
 \* AILERON DEFLECTION PERTAINS TO SURFACE BEHIND SPOILER



MOD C-8A

|      |         |         |         |      |
|------|---------|---------|---------|------|
| CALC | SPITZER | 9-15-70 | REVISED | DATE |
| CHKD |         |         |         |      |
| APP  |         |         |         |      |
|      |         |         |         |      |

SPOILER EFFECTIVENESS

THE BOEING COMPANY

DL-260657

FIG 3-27

PAGE

68

SPOILER ROLLING &  
YAWING MOMENT

$\delta_{sp} = 50^\circ$   
RIGHT  
WING

ROLLING MOMENT  
 $\Delta C_l \delta_{sp} = 50$

$\delta_a \approx 60^\circ$   
45°

30°

15°

0°

$\delta_a \approx -15^\circ$

WING ANGLE OF ATTACK,  $\alpha$  - DEG.

NOTE: 1. USE FOR ALL FLAP DEFLECTIONS  
2. ALLERON DEFLECTIONS  
3. TO SURFACE BLIND SPOT  
4. USE FOR ALL SIDE SLIP ANGLES

$\delta_a \approx 0^\circ$

15°

30°

45°

$\delta_a \approx 60^\circ$

YAWING MOMENT  
 $\Delta C_n \delta_{sp} = 50$

-0.04

|       |         |         |       |      |
|-------|---------|---------|-------|------|
| CALC  | SPITZER | 9-11-70 | 20110 | WING |
| CHECK |         |         |       |      |
| APR   |         |         |       |      |
| APR   |         |         |       |      |

SPOILER ROLLING &  
YAWING MOMENT  
 $\delta_{sp} = 50^\circ$   
RIGHT  
WING

THE BOEING COMPANY

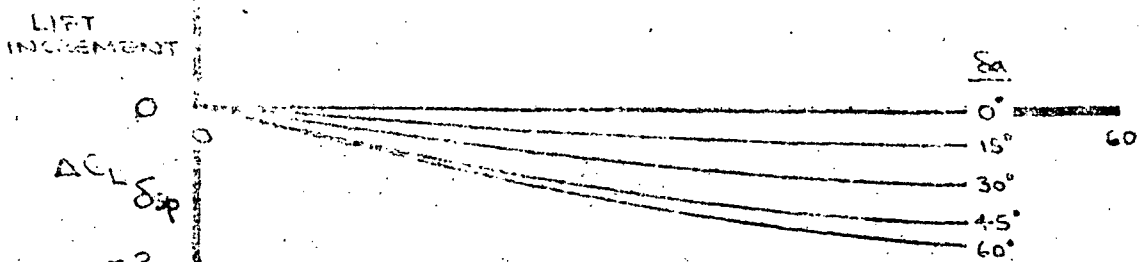
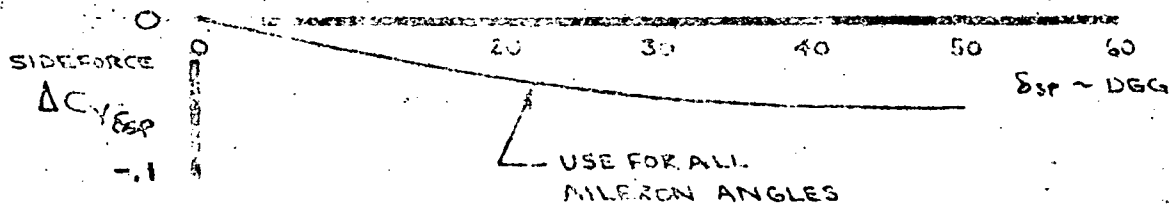
MOD C-2A

D6-26065-12

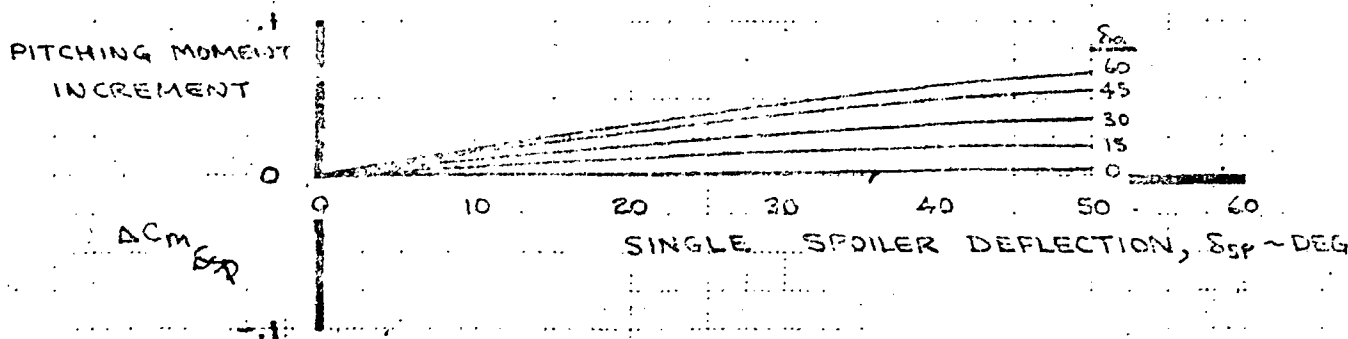
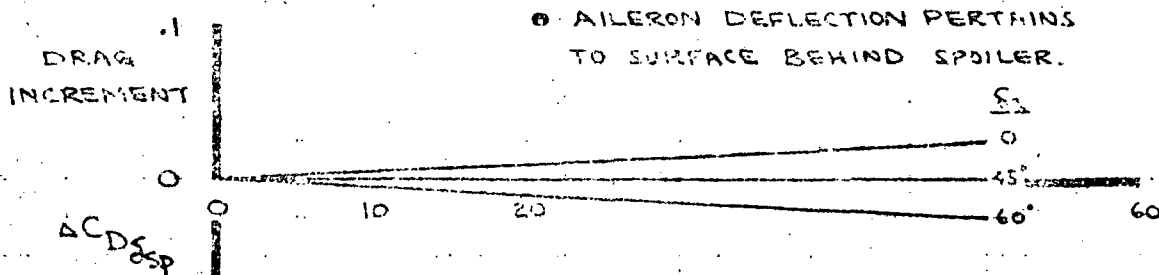
FIG 3-2.5

69

# SPOILER INTERACTIONS



NOTE: \* USE FOR ALL FLAP SETTINGS, ANGLES OF ATTACK AND SIDESLIP.



MOD C-8A

|      |         |         |       |      |
|------|---------|---------|-------|------|
| CALC | SPITZER | 9-16-70 | 18050 | DATE |
| CHKD |         |         |       |      |
| APP  |         |         |       |      |
| APR  |         |         |       |      |

## SPOILER INTERACTIONS

THE BOEING COMPANY

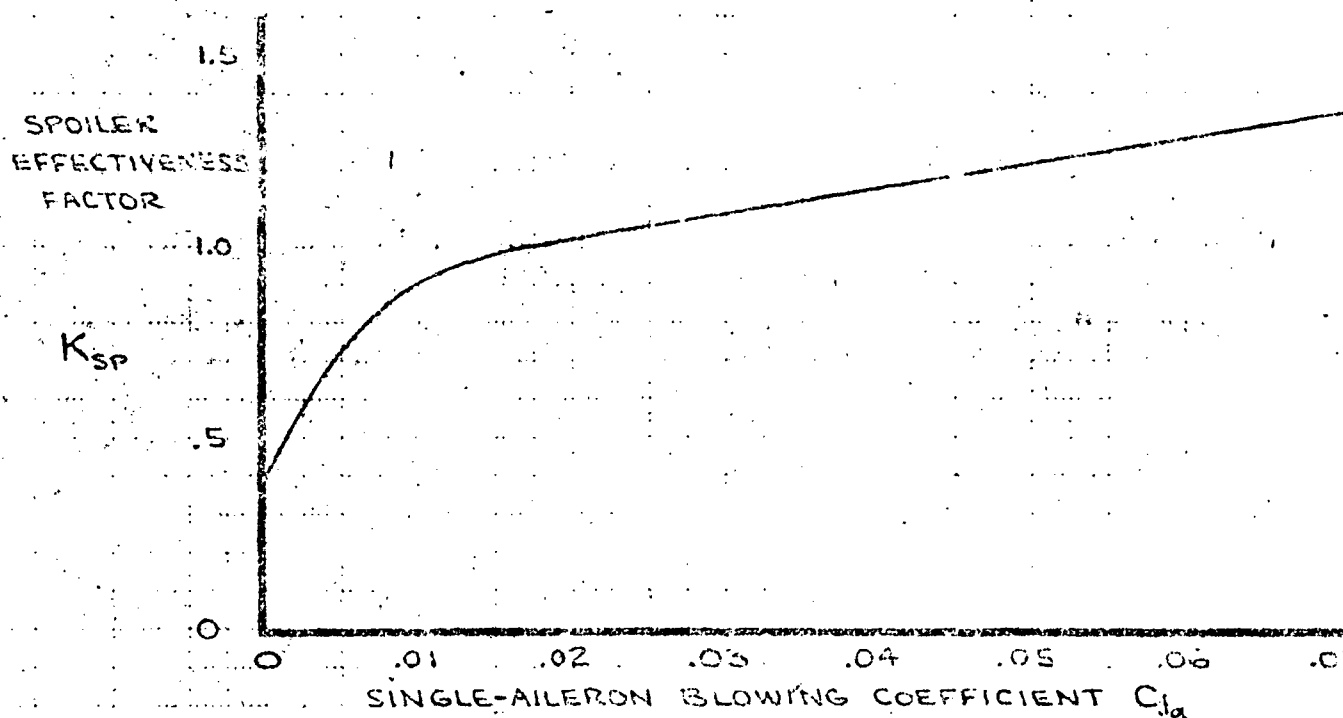
DB-2605TA

FIG 3-29

PAGE

70

EFFECT OF AILERON  
BLOWING ON SPOILER  
EFFECTIVENESS



NOTE:

- AILERON BLOWING PERTAINS TO AILERON SURFACE BEHIND SPOILER

MOD C-82

|       |         |          |       |      |
|-------|---------|----------|-------|------|
| CALC  | SPITZER | 11/12/50 | 30WED | EAIS |
| CHECK |         |          |       |      |
| APR   |         |          |       |      |
| APR   |         |          |       |      |

EFFECT OF AILERON  
BLOWING ON SPOILER  
EFFECTIVENESS

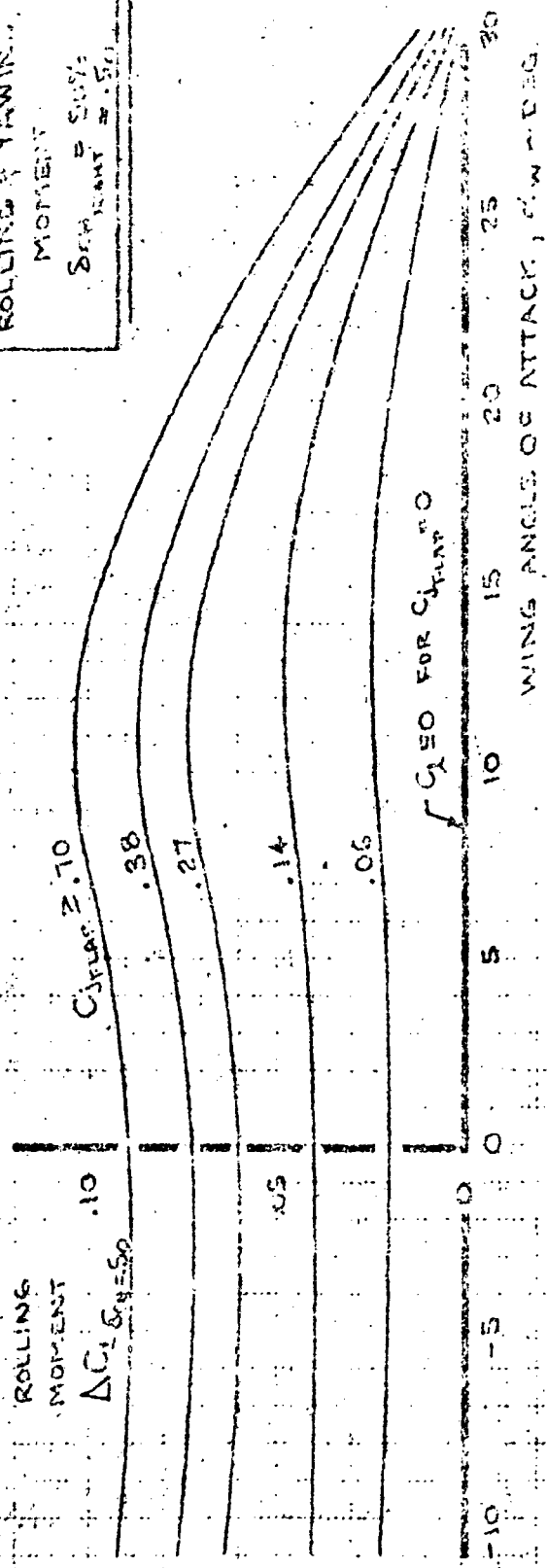
THE BOEING COMPANY

DL-260651

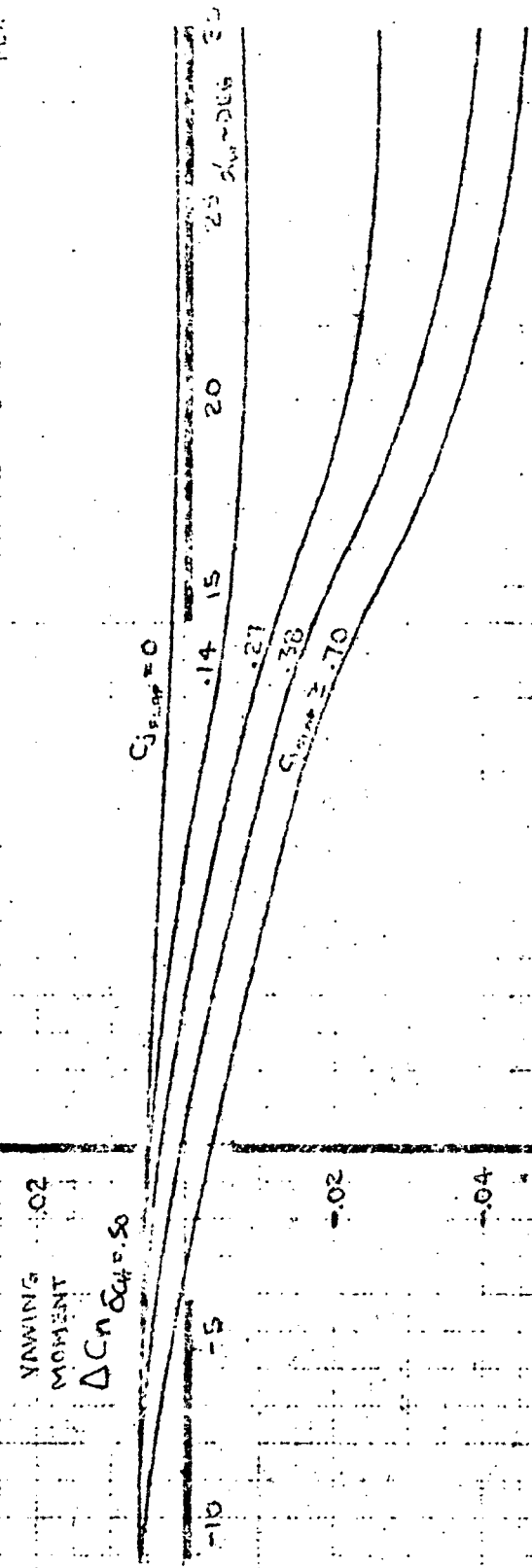
FIG 3-10

DA 15

AUGMENTOR CHOKE  
ROLLING & YAWING  
MOMENT  
 $\delta_{CH} = 50\%$   
 $\delta_{CH} = 50\%$



NOTE: 3 DATA SHOWN FOR  $\delta_{CH} = 70\%$



|       |         |         |         |      |
|-------|---------|---------|---------|------|
| CALC  | SPITZER | 9-11-52 | REVISED | DATE |
| CHECK |         |         |         |      |
| APP   |         |         |         |      |
| APP   |         |         |         |      |

AUGMENTOR CHOKE  
ROLLING & YAWING MOMENT  
 $\delta_{CH} = 50\%$   
THE BOONING COMPANY

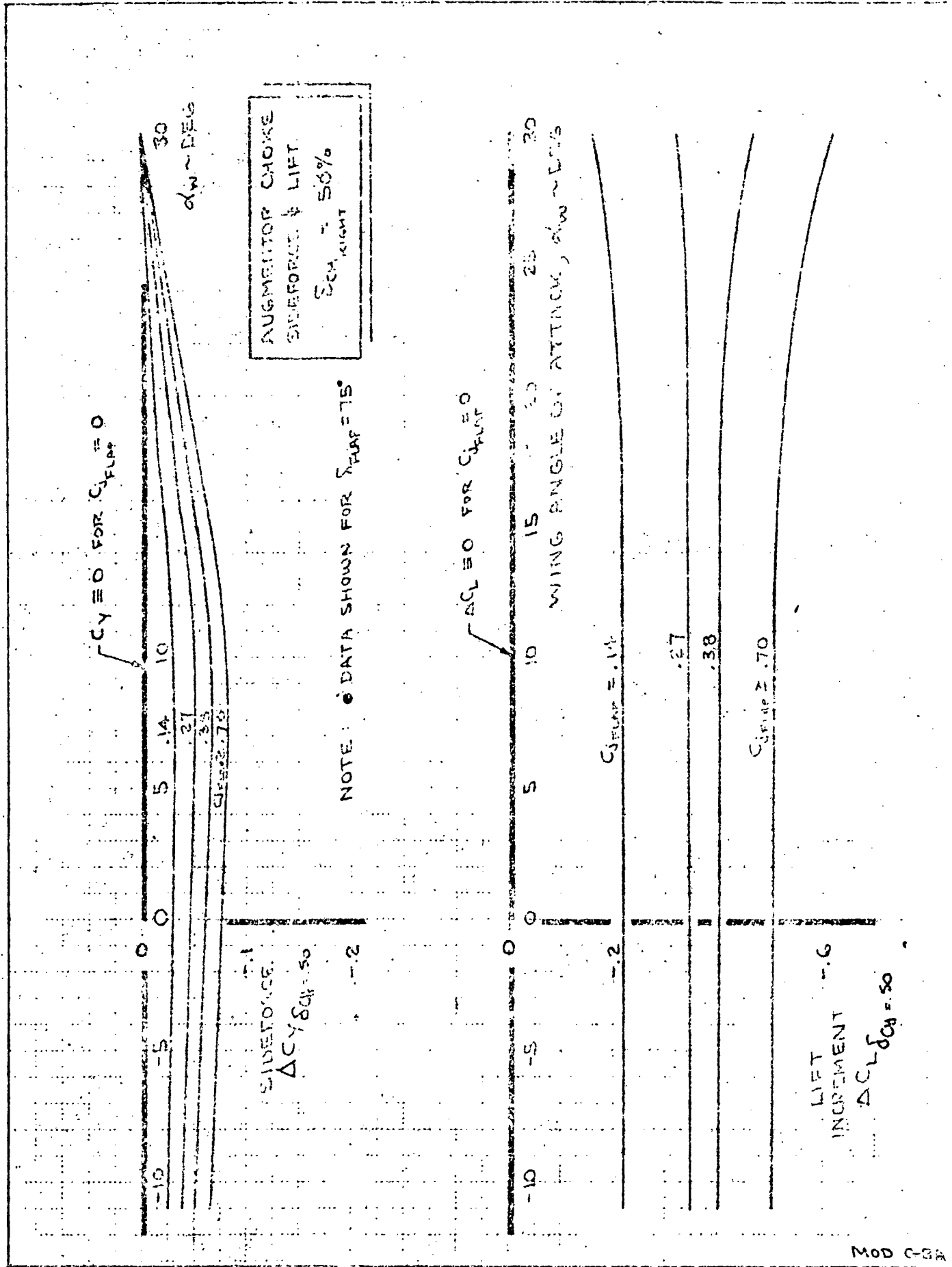
MOD C-80  
D6-26065T  
FIG 3-31  
72

|       |         |      |  |
|-------|---------|------|--|
| CALC  | SP17285 | DATA |  |
| CHECK |         |      |  |
| APR   |         |      |  |
| APR   |         |      |  |

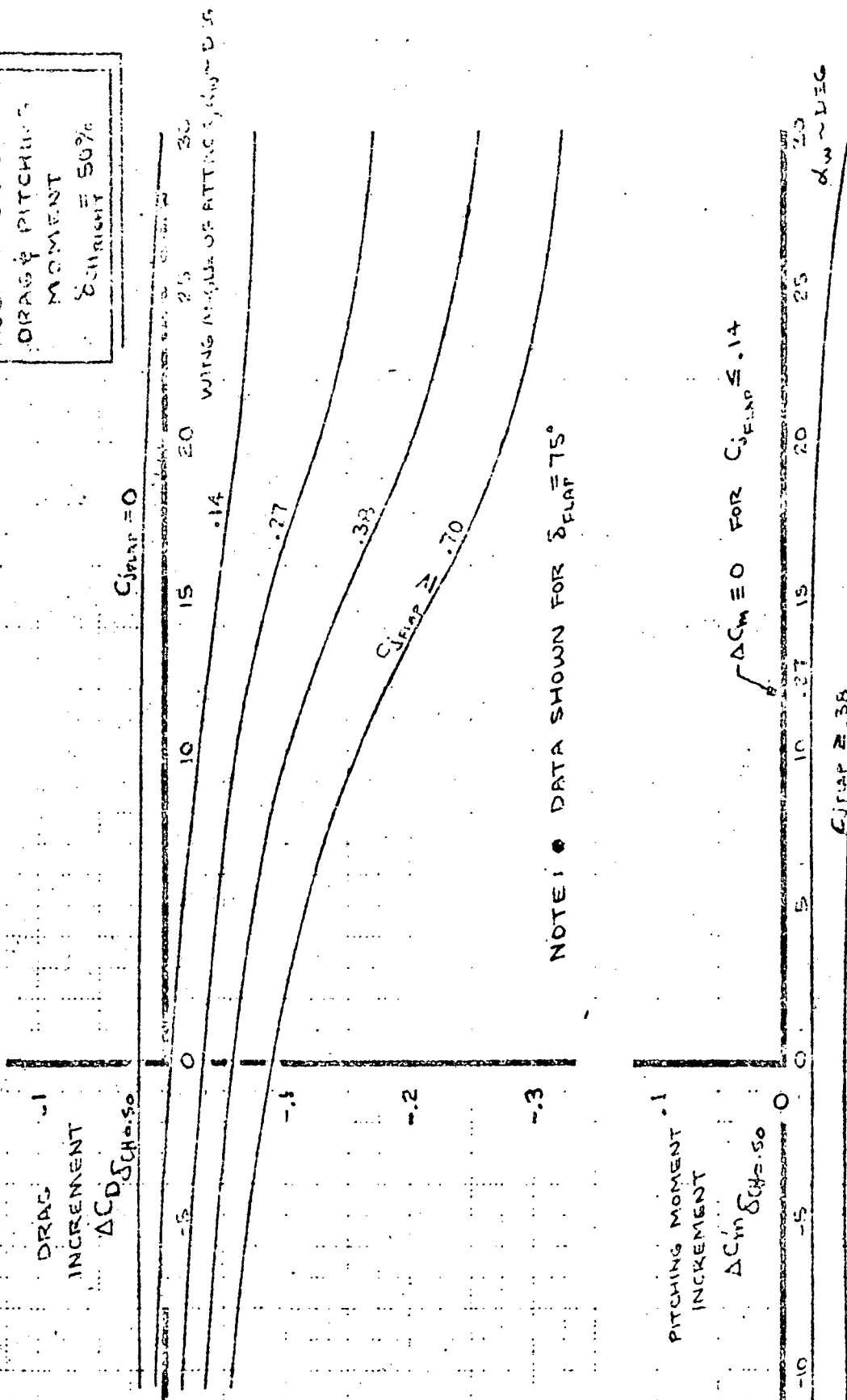
AUGMENTOR CHOKE  
SIDEFORCE & LIFT  
 $\delta_{CH} = 50\%$   
RIGHT

THE ROSSING COMPANY

MOD C-3A  
DC-2005TH  
FIG 3-2  
73



AUGMENTOR CHOKE  
 DRAG & PITCHING  
 MOMENT  
 $\delta_{CHRIGHT} = 50\%$



| CALC. | SPITZER | 1918.75 | 0.0000 | 0.0000 |
|-------|---------|---------|--------|--------|
| CHUCK |         |         |        |        |
| APR   |         |         |        |        |
| APR   |         |         |        |        |

AUGMENTOR CHOKE  
 DRAG & PITCHING  
 MOMENT  
 $\delta_{CHRIGHT} = 50\%$

THE BOEING COMPANY

MOD C-8A

DG-260651

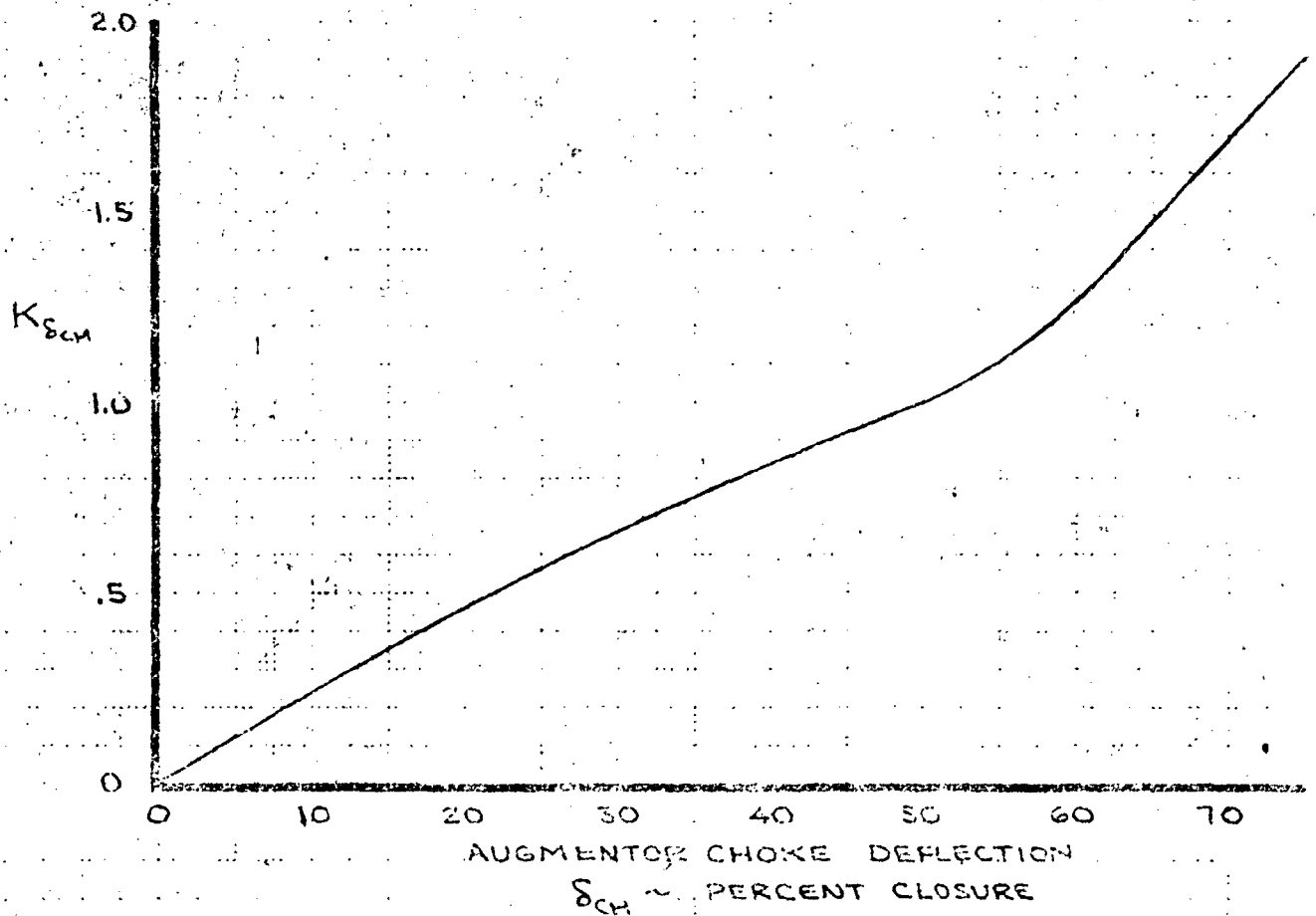
FIG 3-50

PAGE  
 74

# AUGMENTOR CHOKE EFFECTIVENESS WITH CHOKE DEFLECTION

## NOTE:

- USE FOR ALL FLAP SETTINGS
- EFFECTIVENESS MAY BE ERRATIC FOR  $\alpha > 5^\circ$  AND  $\delta_{CH} \geq 75\%$ .



MOD C-8

|      |         |         |      |      |
|------|---------|---------|------|------|
| CALC | SPITZER | 9-16-77 | FWED | DATE |
| CHOK |         |         |      |      |
| APR  |         |         |      |      |
| APR  |         |         |      |      |

AUGMENTOR CHOKE  
EFFECTIVENESS WITH  
CHOKE DEFLECTION

THE SOBINO COMPANY

DG-260857

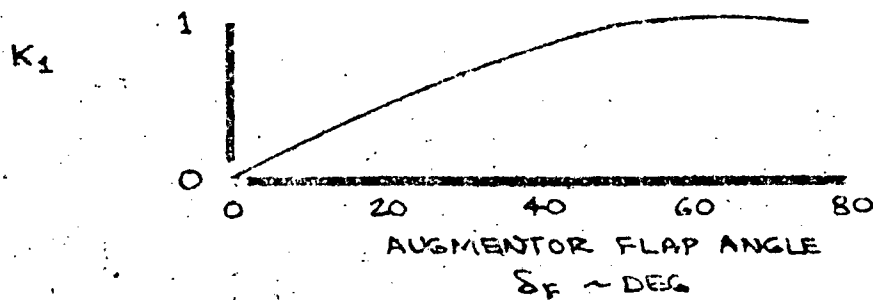
FIG 3-1

PAGE

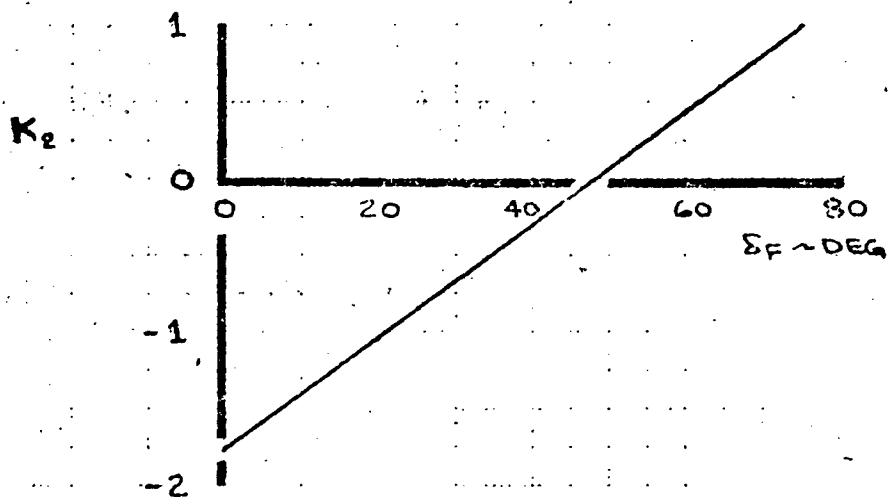
75

AUGMENTOR CHOKE  
EFFECTIVENESS  
VARIATION WITH FLAP  
ANGLE

NOTE: • USE FOR ALL VALUES  
OF  $C_{j\_FLAP} \neq \delta_{CH}$



LIFT, PITCHING  
MOMENT, ROLLING  
MOMENT, SIDEFORCE



DRAG, YAWING  
MOMENT

MOD C-8A

|      |         |         |        |      |
|------|---------|---------|--------|------|
| CALC | SPITZER | 9-21-70 | 354560 | DATE |
| CHKD |         |         |        |      |
| APP  |         |         |        |      |
| ISS  |         |         |        |      |

AUGMENTOR CHOKE EFFECTIVENESS D6-26065T  
VARIATION WITH FLAP ANGLE

FIG 3-53

THE BOEING COMPANY

PAGE

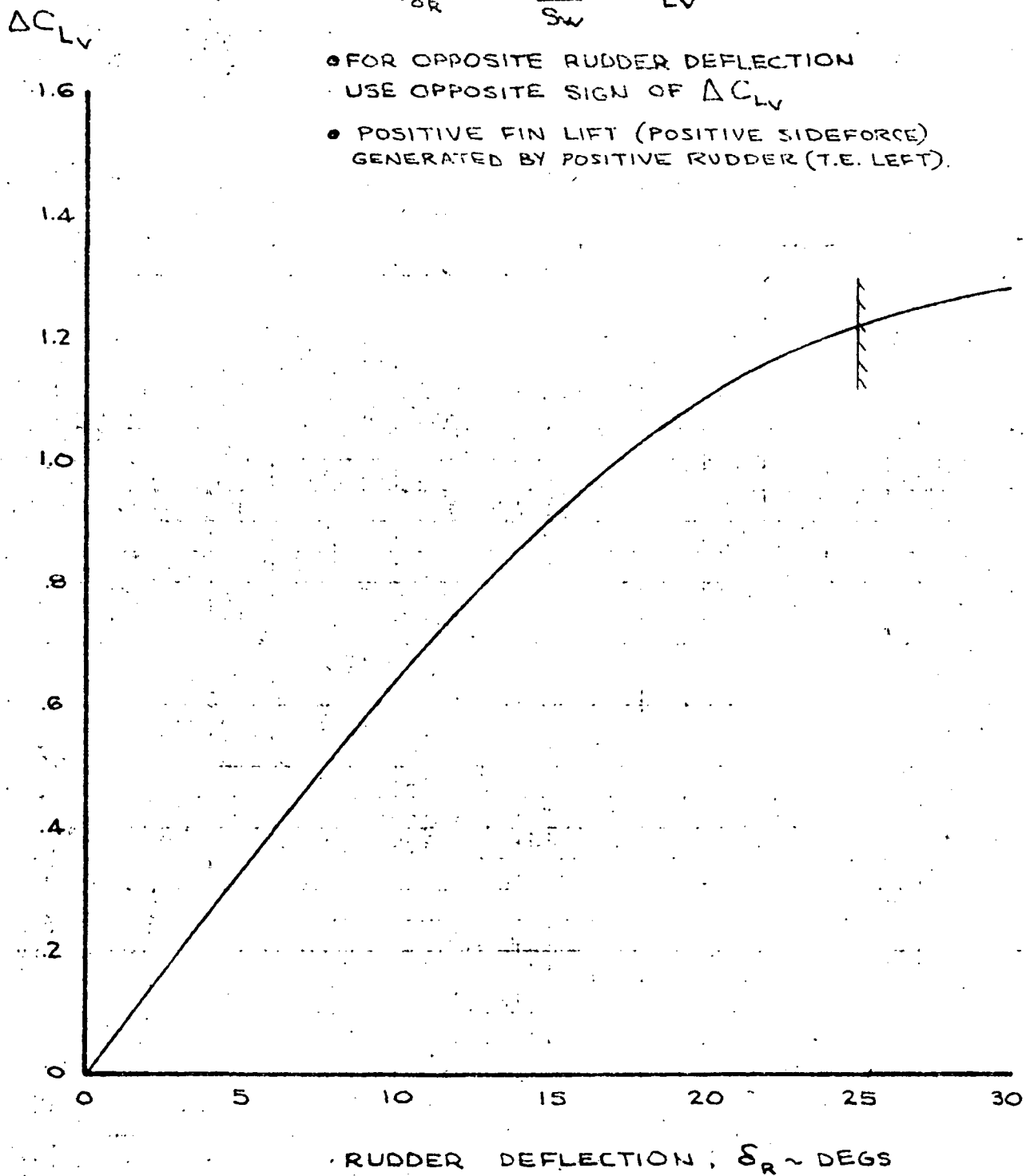
76

\_\_\_\_\_

VERTICAL  
TAIL  
LIFT COEFFICIENT

$$G \Delta C_{Y \delta R} = \frac{S_v}{S_w} \cdot \Delta C_{LV}$$

- FOR OPPOSITE RUDDER DEFLECTION  
USE OPPOSITE SIGN OF  $\Delta C_{Lv}$
- POSITIVE FIN LIFT (POSITIVE SIDEFORCE)  
GENERATED BY POSITIVE RUDDER (T.E. LEFT).



MOD C-8A

|       |        |         |         |      |                      |            |
|-------|--------|---------|---------|------|----------------------|------------|
| CALC  | RUNSEY | 10/2/70 | REVISED | DATE | RUDDER EFFECTIVENESS | D6-26065TN |
| CHECK |        |         |         |      |                      | FIG 3-36   |
| APR   |        |         |         |      |                      | PAGE       |
| APR   |        |         |         |      |                      | 77         |
|       |        |         |         |      | THE BOEING COMPANY   |            |

#### 4.0 Propulsive Forces and Moments

Propulsive forces and moments are added to the aerodynamic forces and moments resolved into the body axes in the same way as in the existing simulation, Reference 2. The new moment arms are given on Table 3-6. The engine model computes gross hot thrust, isentropic cold thrust and inlet mass flow (for ram drag). Ram drag effects have been included in the aerodynamic model, Section 3. Figure 4-1 shows the flow diagram for the engine simulation. The engine arms and all geometric data are given on Table 3-6.

The static relationships between throttle inputs,  $\delta_{TH}$ , and RPM,  $T_c$ ,  $T_H$  and  $\dot{M}$  are shown on Figure 4-2, including all duct and exhaust losses for the nominal engine configuration (engine test stand data corrected for conical nozzle tail pipes). The factors  $C_{5X}$  and  $C_{5Z}$  (see Figure 4-1) give the extra losses due to nozzle rotation angle. The engine acceleration/deceleration dynamics are shown on Figures 4-3 and 4-4. Engine failures will be simulated by a thrust loss of 42% per second.

The augmentor nozzle flow for each wing is derived in part from both the left and right engines through the crossover ducts. The ratio of flow from each engine to each wing may be varied in the simulation in order to investigate how much engine-out moment relief can be gained from asymmetric blowing. The total blowing coefficient is given by:

40 15460



$$C_{J_{TOT}} = \frac{T_{CL} + T_{CR}}{\bar{Q} \cdot SW}$$

Each wing flap receives a proportional coefficient:

$$** \quad C_{J_{FLAP L}} = \frac{.38 T_{CL} + .43 T_{CR}}{\bar{Q} \cdot SW}$$

$$** \quad C_{J_{FLAP R}} = \frac{.43 T_{CL} + .38 T_{CR}}{\bar{Q} \cdot SW}$$

The ailerons are fed from the engine on the opposite wing.

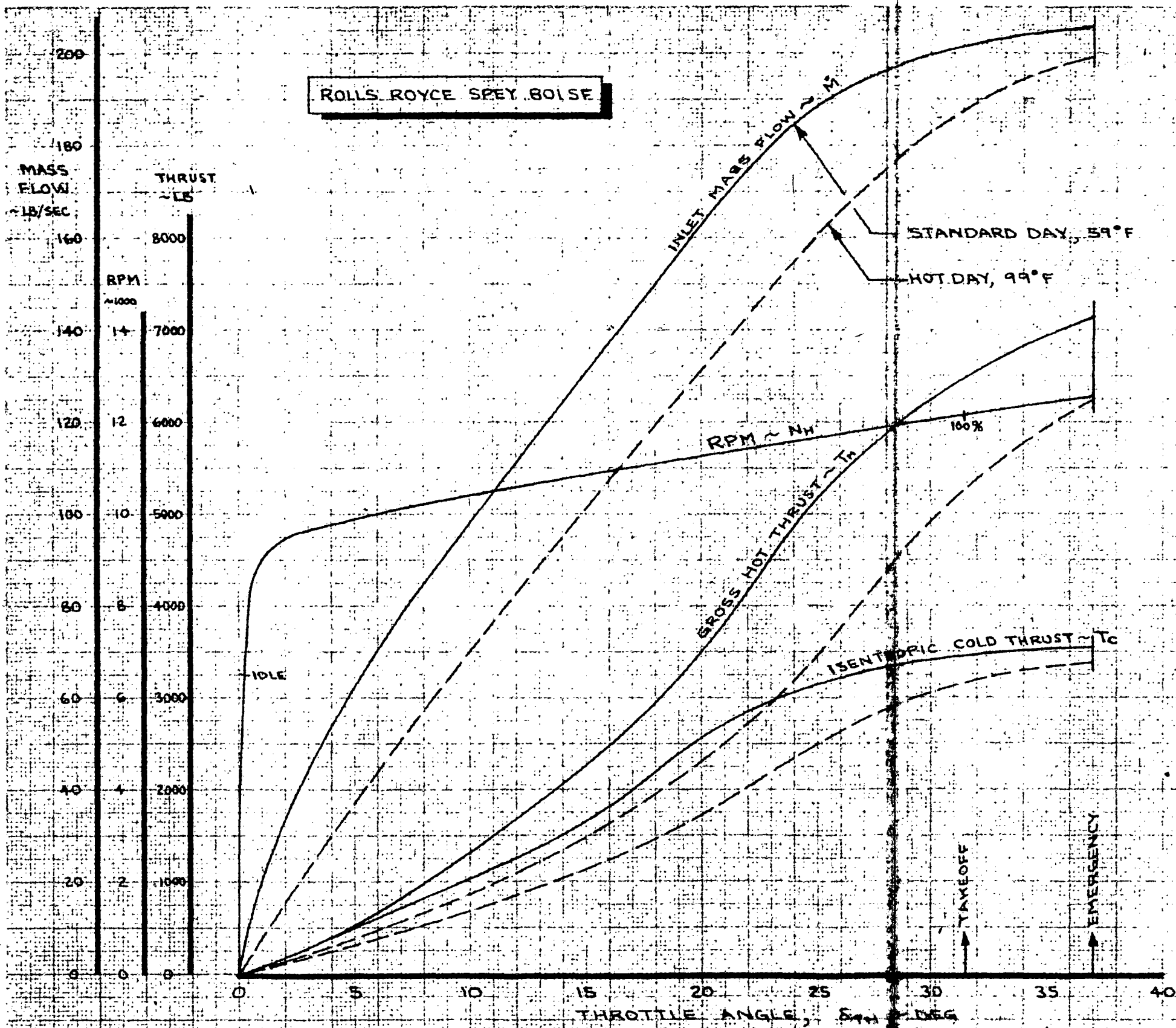
The equations are:

$$C_{JAL} = \frac{.12 T_{CR}}{\bar{Q} \cdot SW}$$

$$C_{JAR} = \frac{.12 T_{CL}}{\bar{Q} \cdot SW}$$

\*\* The duct lag may be simulated by a first order lag. Nominally this lag will be set to zero.





**NOTE:**

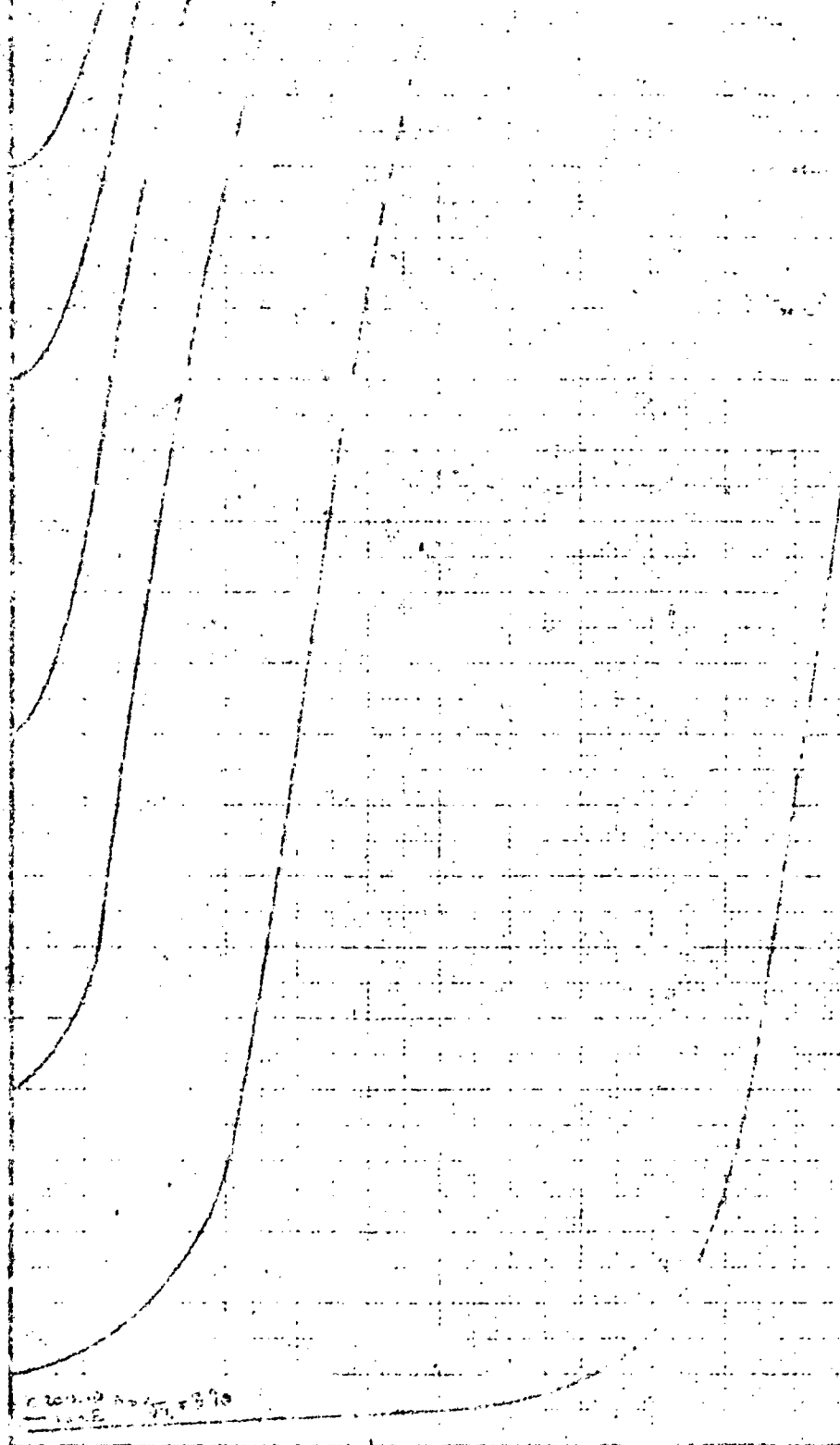
1. SEA LEVEL
2. ENGINE TEST BED DATA EVALUATED AT GOKTS AND CORRECTED TO CONICAL NOZZLE CONFIGURATION.
3. CONICAL NOZZLE GROSS HOT THRUST SHOWN DOES NOT INCLUDE GEOMETRIC EFFECT OF NOZZLE ANGLE.
4. BLEED LOSSES INCLUDED.
5. ISENTROPIC COLD THRUST INCLUDES AIRPLANE DUCT SYSTEM LOSSES. ACTUAL COLD THRUST WOULD INCLUDE AN ADDITIONAL  $C_D$  FACTOR

|       |         |          |         |      |                                                                                     |            |
|-------|---------|----------|---------|------|-------------------------------------------------------------------------------------|------------|
| CALC  | SPITZER | 12-17-71 | REVISED | DATE | ENGINE THRUST CHARACTERISTICS<br>MODIFIED C-8A SIMULATOR DATA<br>THE BOEING COMPANY | 06-26065TH |
| CHECK |         |          |         |      |                                                                                     | FIG 4-2    |
| APPD  |         |          |         |      |                                                                                     | PAGE       |
| APPD  |         |          |         |      |                                                                                     | 81         |

REF. C/S 6-840-11-506,  
SEPT 13, 1971

PERCENTAGE OF TAKEOFF THRUST - 100

100  
80  
60  
40  
20  
0



ADJUSTED AND  
REVISED

|        |         |
|--------|---------|
| DATE   | 2-27-57 |
| ENGINE |         |
| APPROX |         |
| APPROX |         |

ENGINE RECEPTION

82-24065TR  
FIG 4-3

THE BOEING COMPANY

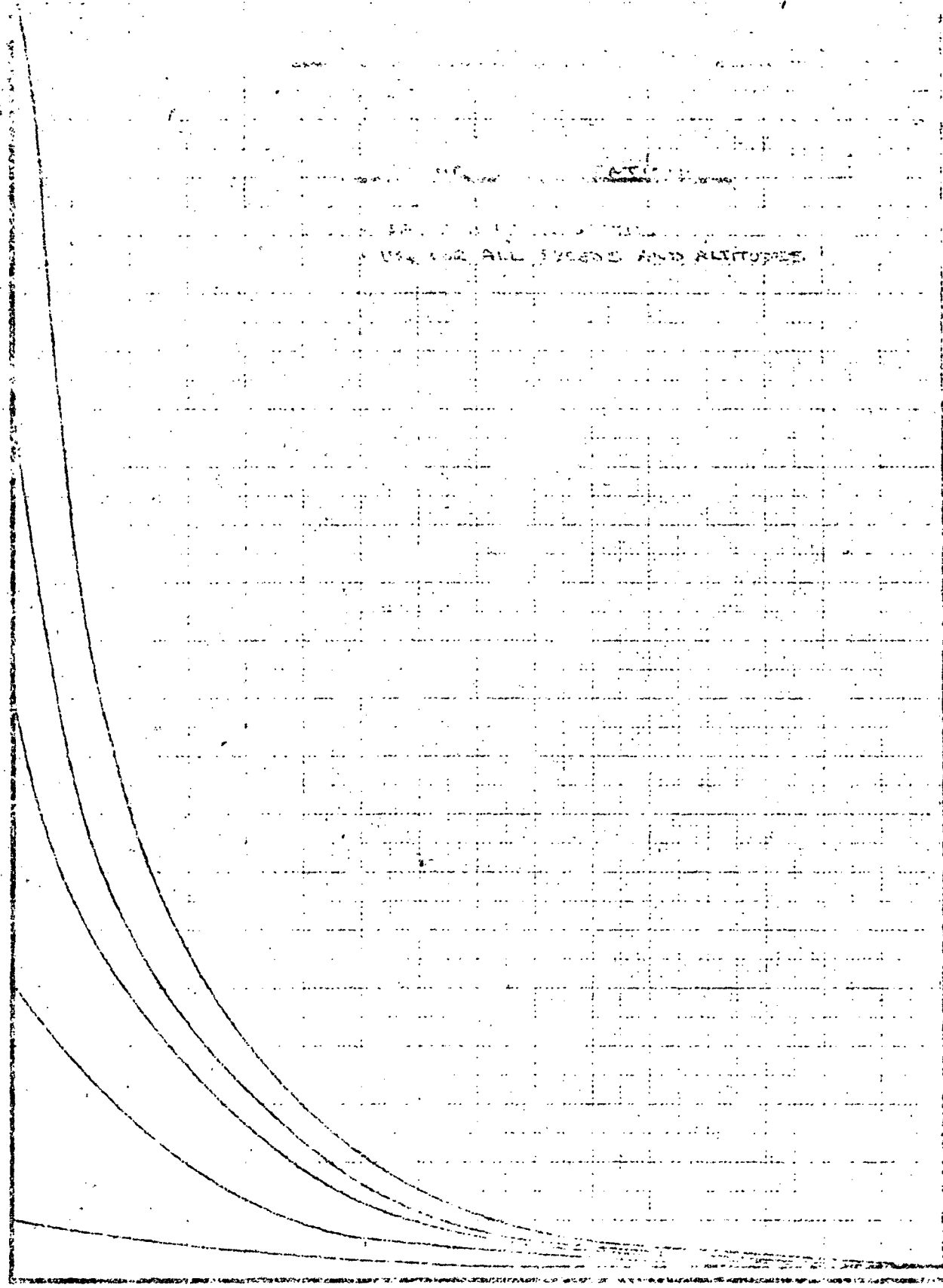
PERCENTAGE OF TAKEOFF THRUST - %

100  
90  
80  
70  
60  
50  
40  
30  
20  
10  
0

ELAPSED TIME - SECS.

0 2 4 6 8 10 12 14

USE FOR ALL SPEEDS AND ALTITUDES



2. 5. 10. 15. 20. 25. 30. 35. 40. 45. 50. 55. 60. 65. 70. 75. 80. 85. 90. 95. 100.

ENGINE DECELERATION

THE ENGINE COMPANY

DG-26057

FIG 4

## 5.0 Airplane Mass and Inertia Characteristics

For each simulated flight the weight and inertia remain constant since fuel burn-off is ignored. There should, however, be capability to change initial weight, c.g. position and inertias prior to each run. The Modified C-8A weight and c.g. characteristics are shown in Figure 5-1. For normal conditions the c.g. varies with gross weight (fuel loading) along the "nominal CG" line. Crew and flight test instrumentation weight are included in the O.E.W. value. Airplane moments of inertia are presented in Figures 5-2 through 5-4. Two roll inertia levels are shown depending on fuel system configuration.

AD 1848D

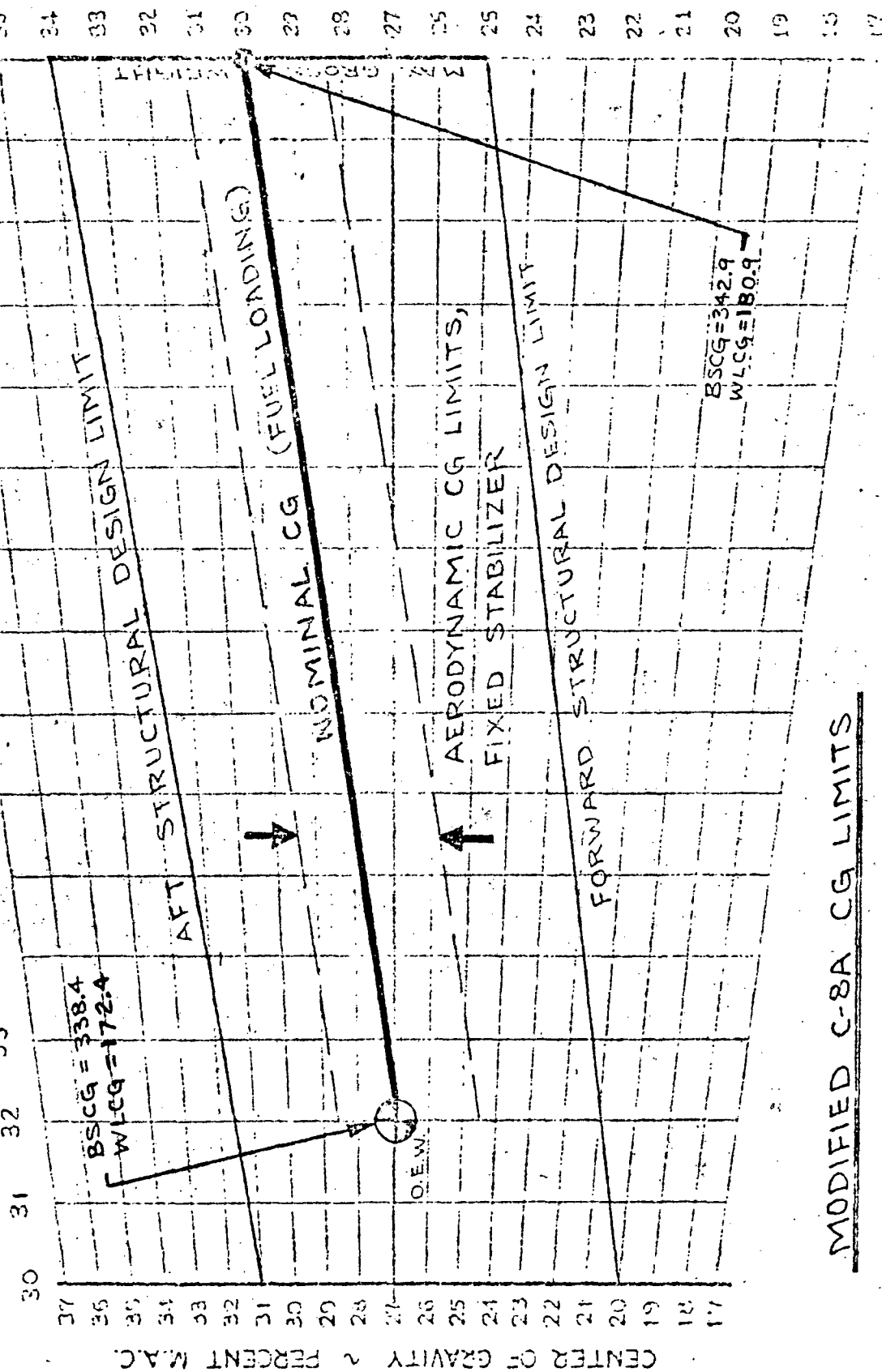


| DATA               | SCALE ONE INCH REPRESENTS |
|--------------------|---------------------------|
| 70% M.A.C.         | W.L.C.G.                  |
| 138.2 IN.          | 2000 LB                   |
| M.A.C. = 149.0 IN. | LE M.A.C. = 298.0 IN.     |

WEIGHT ~ 1000 LBS.

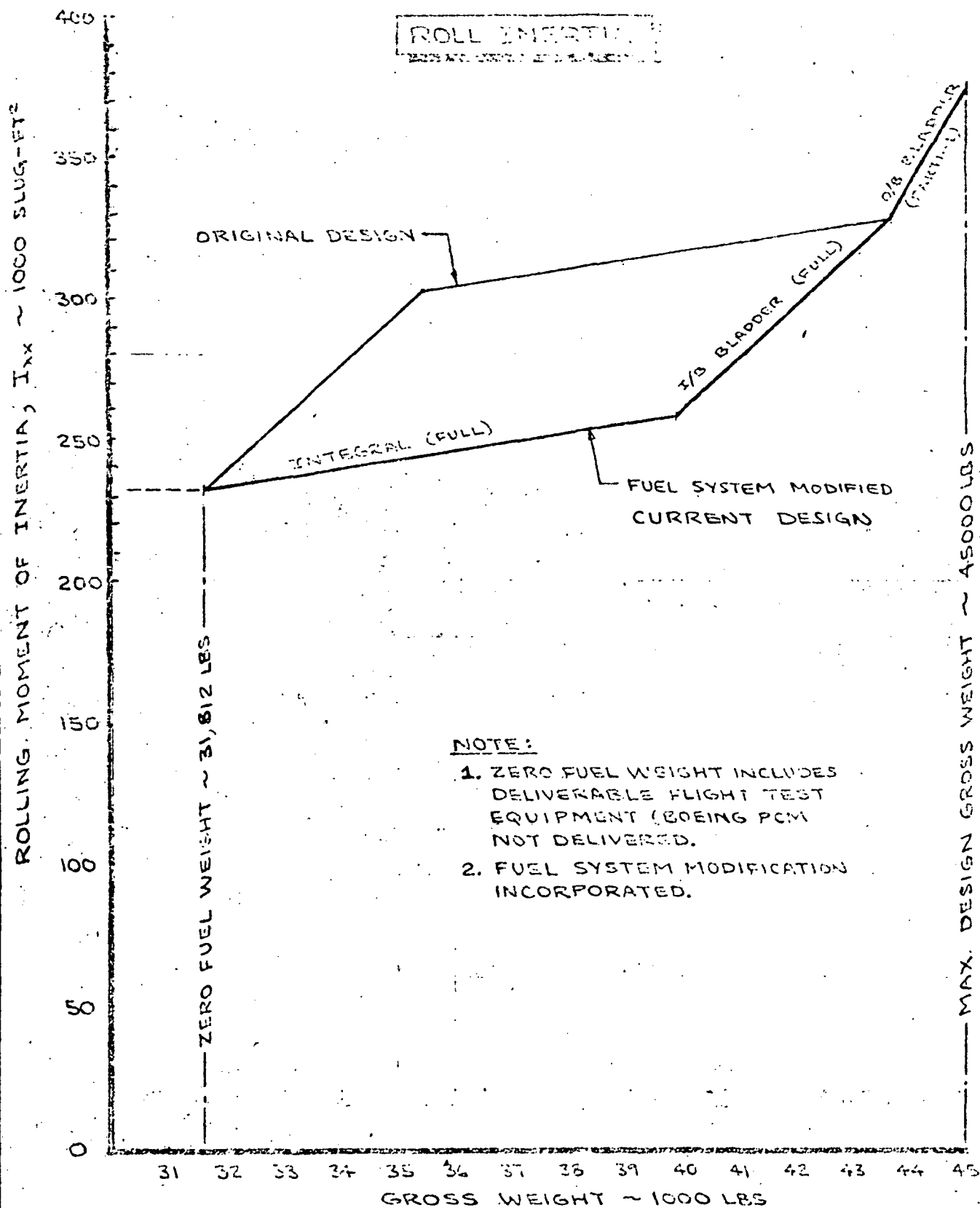
- OEW CG AND FUEL LOADING LINE FROM D6-25416, P. 9.2, REV. B

- STABILIZER INCIDENCE SET AT  $\alpha = 0^\circ$  FOR NOMINAL CG



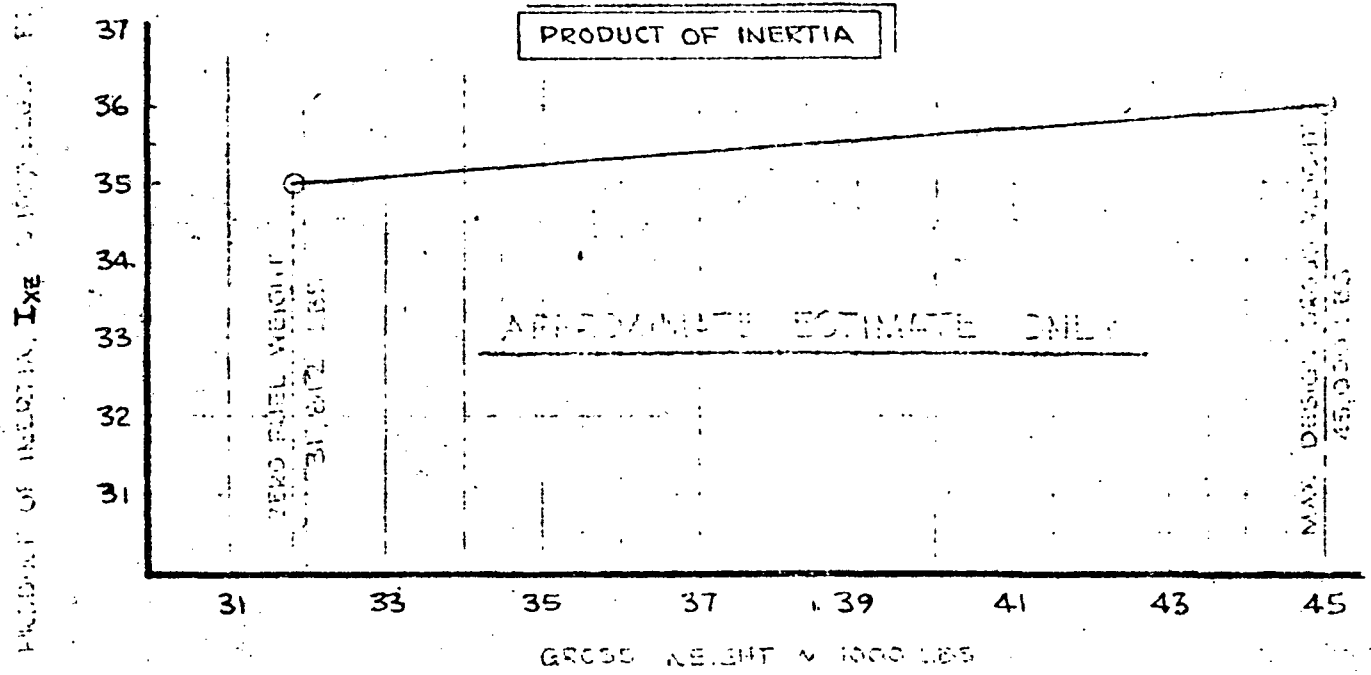
MODIFIED C-8A CG LIMITS

FIG 5-1



|       |                   |          |      |                                             |  |
|-------|-------------------|----------|------|---------------------------------------------|--|
|       |                   |          |      | MOD C-8A                                    |  |
| CALC  | SPITZER 12-1-71   | REVIEWED | DATE | ROLLING MOMENT OF INERTIA FOR MODIFIED C-8A |  |
| CHECK | GRIFFITHS 12-1-71 |          |      |                                             |  |
| APR   |                   |          |      | THE BOEING COMPANY                          |  |
| APR   |                   |          |      |                                             |  |
|       |                   |          |      | D6-260657N                                  |  |
|       |                   |          |      | FIG 5-2                                     |  |
|       |                   |          |      | PAGE 86                                     |  |





|      |          |      |         |                                            |            |
|------|----------|------|---------|--------------------------------------------|------------|
| ITEM | QUANTITY | UNIT | REMARKS | PRODUCT OF INERTIA<br>FOR<br>MODIFIED C-8A | D6-2606STW |
| AFR  |          |      |         |                                            | FIG 5-4    |
|      |          |      |         | THE ABOVE CONTAINS                         | 88         |

## 6.0 Rigid-Body Equations of Motion in the Earth's Gravitational Field, Including Atmospheric Disturbances

These should be the same as those used in the existing NASA simulation, Reference 2, including the special gust model. These special gusts,  $\dot{P}_{DG}$  and  $\dot{R}_{DG}$ , simulate the effects of large amplitude random disturbances in roll and yaw. Additionally, downdrafts and shears will be used to evaluate maximum flight control system rate requirements. Figure 6-1 specifies the shear as a function of altitude and the downdraft as a function of distance from the runway threshold.

## 7.0 Accelerations at the Pilot's Station

These are computed for use in driving the NASA FSAA six-degree-of-freedom motion base. Existing equations derived by NASA should be used. Stall buffet may be added at  $C_{L_{max}}$  if this can easily be added to the existing program.

The pilot is seated a distance  $X_p$  forward of the c.g. and  $Z_p$  above it, where:

$$X_p = \frac{BSCG - 115.5}{12} \quad \text{feet}$$

$$Z_p = \frac{193.85 - WLCG}{12} \quad \text{feet} \quad (\text{eye position})$$

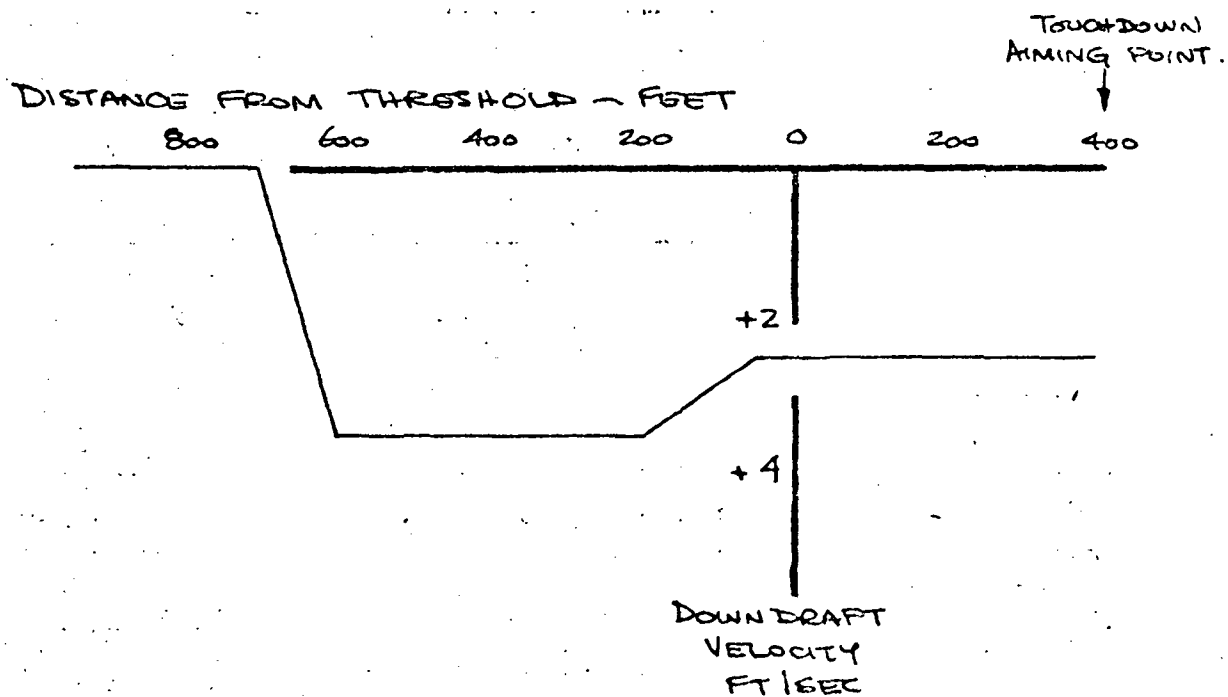
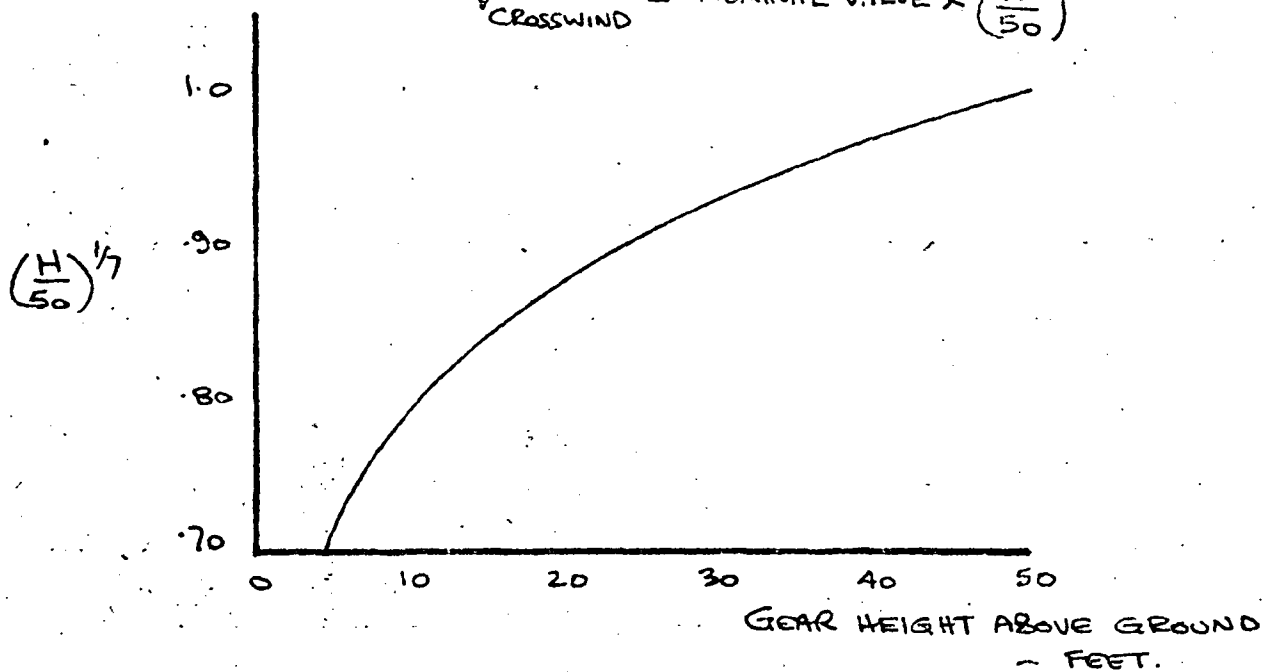
## 8.0 Outside World Visual Presentation to the Pilot

A color television display of a 1/600 scale model is used at NASA to give a representation of the visual scene as would be observed during approach and landing under varying lighting, cloud and visibility conditions. The field of view available should be adjusted to represent as closely as possible that available from the Buffalo.

AD 10490



$$V_{\text{CROSSWIND}} = \text{NOMINAL VALUE} \times \left(\frac{H}{50}\right)^{1/7}$$



MOD C-8A

|                    |        |         |         |      |                                            |             |
|--------------------|--------|---------|---------|------|--------------------------------------------|-------------|
| CALC               | RUMSEY | 9/20/70 | REVISED | DATE | SHEAR AND DOWNDRAFT<br>FOR STOL EVALUATION | DO-2606STN3 |
| CHECK              |        |         |         |      |                                            |             |
| APR                |        |         |         |      |                                            | FIG. 6-1    |
| APR                |        |         |         |      |                                            | PAGE 90     |
| THE BOEING COMPANY |        |         |         |      |                                            |             |

## 9.0 Pilot's Instruments and Cab Environment

The instrumentation layout required for the simulator cab is shown schematically on Figure 9-1.

Raw ILS signals will be fed to the HSI and the ADI command bars. Both the glide slope error and the localizer error signals will be used at 1/3 sensitivity while flying the 7-1/2 degree approach path to compensate for the increased signal sensitivity given by the shorter range to touchdown on these steep glide paths.

The Boom IAS should run one full revolution per 100 knots. The radio altimeter, VSI, T/S, compass, standby altimeter and ASI are all styled as the existing Buffalo instruments.

Instruments indicating trim position for the aileron and the rudder will be provided, preferably placed on the overhead consol as in the Buffalo. A dual function (lateral/longitudinal) trim switch will be provided at the pilot's left hand thumb position. Rudder trim is via a switch situated next to the rudder trim indicator.

Engine noise will be simulated at a level high enough to mask the noise from the motion base.

AD 15460



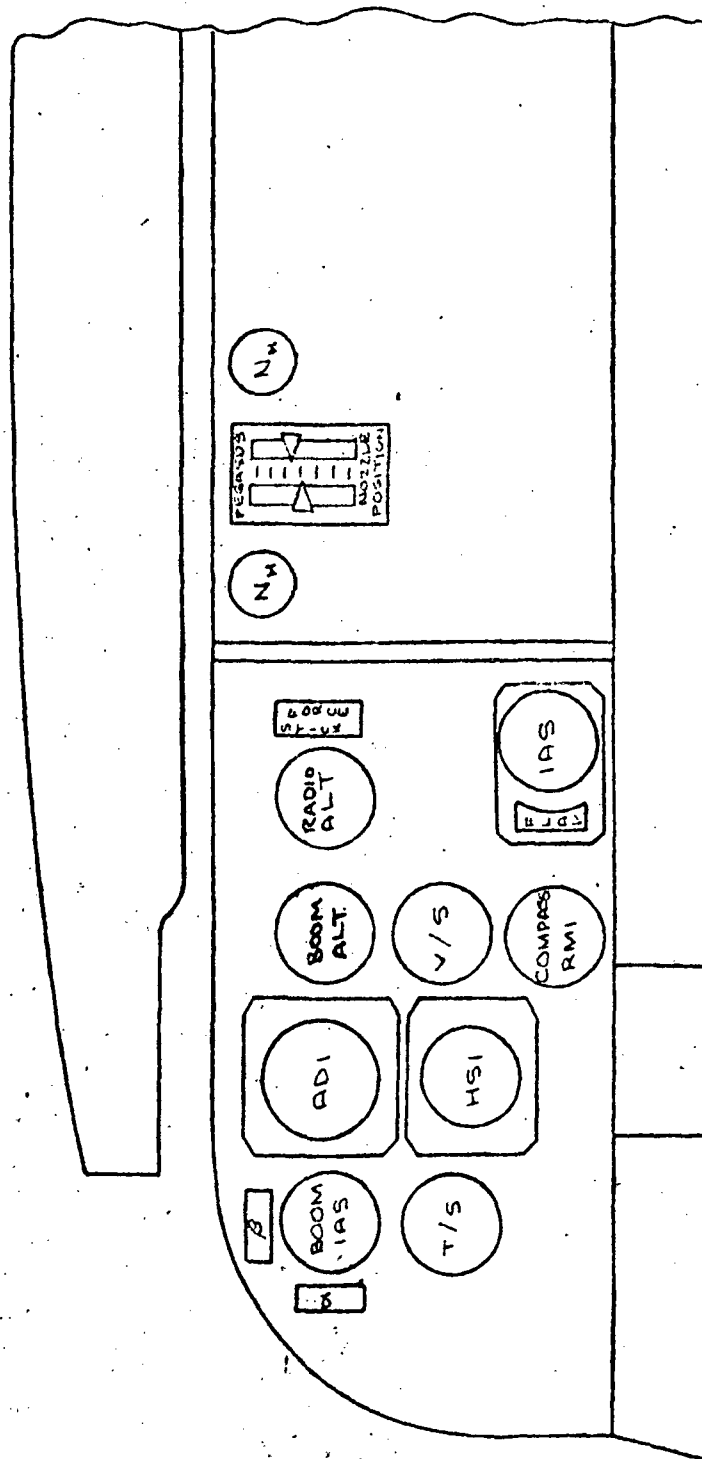
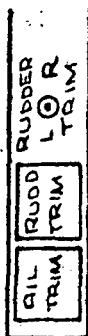


FIG 9-1 INSTRUMENTATION REQUIREMENTS FOR SIMULATOR CAB

|            |                  |
|------------|------------------|
| D6-26065TN |                  |
| Calc       | RUMSEY 9.10.70   |
| REVISED    | SPITZER 12/19/71 |
| FIG 9-1    |                  |
| PAGE 92    |                  |

## 10.0 Lateral-Directional Stability Augmentation System

The lateral-directional stability augmentation system is divided into separate systems for each control axis. Both systems are automatically disengaged above 100 knots.

Figure 10-1 shows the simulation diagram of the lateral SAS. Three feedback signals are used along with the control wheel input described in Section 2.2:

1. Body axis roll rate ( $P_B$ ) for roll mode augmentation
2. Body axis yaw rate ( $P_Y$ ) for spiral mode augmentation
3. Sideslip angle ( $\beta$ ) for variation of dihedral effect

An on/off switch and adjustable gain is provided for each loop. The signals are summed and fed to the 35 rad/sec band-width servo actuator that is rate and position limited. Switch 4 represents the pilots' engage/dis-engage switch and quick disconnect switch. Switches 5 and 6 are used for simulating active failures. The location of switches in the cab should closely approximate their actual location in the airplane. Accuracy is necessary to properly evaluate pilot response to failures.

Lateral SAS inputs are summed in series with pilot's wheel inputs and fed to the ailerons, choke and spoiler actuators. Provisions should be made to permit lateral SAS inputs to the ailerons only (Switch 7).

In the normal operating mode, the lateral SAS gains are:



$$K_{RH} = \begin{cases} 1.50 \text{ deg/deg/sec} & (\text{flap} > 40^\circ) \\ .75 \text{ deg/deg/sec} & (\text{flaps} < 40^\circ) \end{cases}$$

$$K_{SH} = 1.75 \text{ deg/deg/sec}$$

$$K_{DIH} = 0$$

For simulator studies the gains may be varied at will; however, the variable gain feature of the actual SAS permits a range of  $-2.7 \leq K \leq +2.7$  for all channels. Figure 10-2 shows the directional stability augmentation system. Switches 8, 9, 10 control the turn coordinator feedback signals, which consist of wheel position ( $\delta_W$ ), body axis roll rate ( $P_B$ ) and roll attitude ( $\phi$ ). Dutch roll damping uses either body axis yaw rate,  $R_B$ , (Switch 11) or a derived sideslip rate,  $\dot{\beta}_{der}$ , where  $\dot{\beta}_{der} = -R_B + .293\dot{\phi}$  (Switches 11 and 12). The signal may or may not be bandpassed (Switch 13). All signals are summed and fed to a servo actuator as in the lateral axis. Switch 16 is the system engage/disengage, quick disconnect switch and must be synchronized with switch 4 in the lateral SAS.

In the normal operating mode, the directional SAS gains are

$$K_{TC1} = 0$$

$$K_{TC2} = \begin{cases} .60 \text{ deg/deg/sec} & (\text{flap} > 40^\circ) \\ .30 \text{ deg/deg/sec} & (\text{flap} < 40^\circ) \end{cases}$$

$$K_{TC3} = \begin{cases} .10 \text{ deg/deg} & (\text{flap} > 40^\circ) \\ 0 & (\text{flap} < 40^\circ) \end{cases}$$

$$K_{YD} = .40 \text{ deg/deg/sec} \quad (\text{with filter})$$

AD 1546D



All gains may be varied at will; however, the variable gain feature of the actual directional SAS permits the following range.

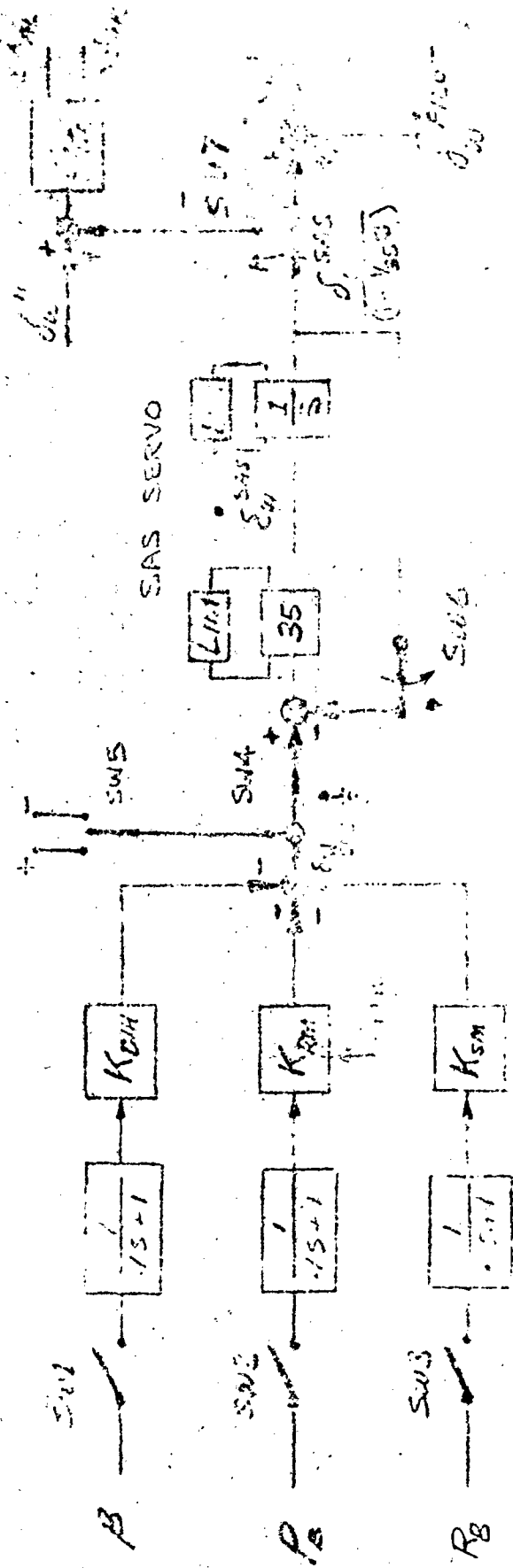
$$-2.7 \leq K_{TC2} \text{ or } K_{YD} \leq +2.7$$

$$-.54 \leq K_{TC1} \text{ or } K_{TC3} \leq +.54$$

AD 18480



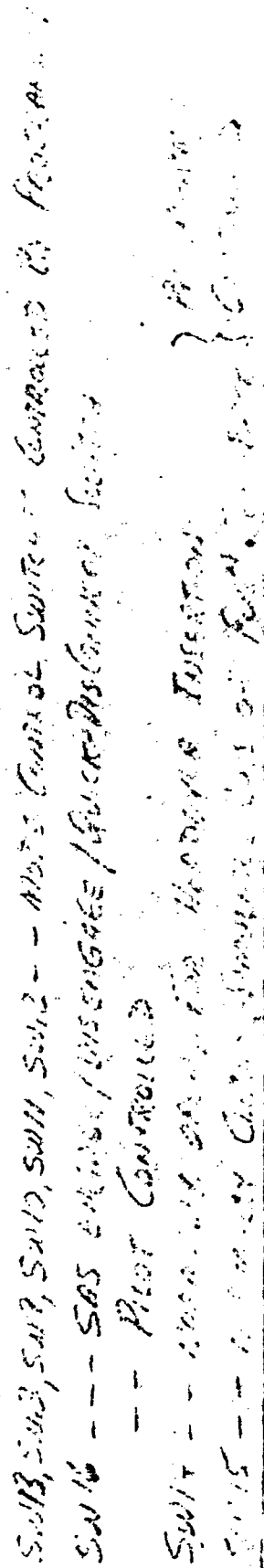
LATERAL SAS  
SIMULATION



SW1, SW2, SW3 --- Mode Control Switches Control to be Proportional  
SW4 --- SAS mode/Proportional/Rate Feedback & Summing  
SW5 --- Feedback Control  
SW6 --- Motor Control  
SW7 --- Motor Control

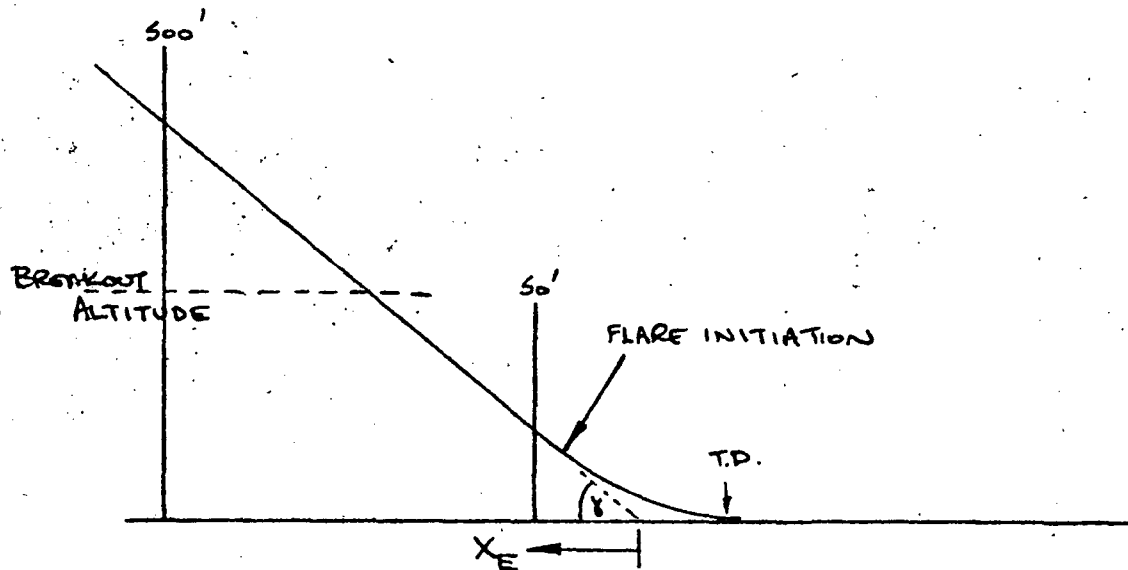
|                   |               |      |            |
|-------------------|---------------|------|------------|
| PROJ.             | REVISED       | DATE | MOD C-BA   |
| CHECK             | SPITZER 7 8/7 |      | D6-2606ST1 |
| APR               |               |      | FIG 10-1   |
| APR               |               |      | 96         |
| THE BOOTH COMPANY |               |      |            |

LATERAL SAS SIMULATION



## 11.0 Additional Simulation Print-Out Requirements

11.1 As an aid in evaluating the airplane performance and the pilot workload for each simulated landing or failure case, the following parameters should be stored in the computer during each run and unloaded in printed form at the end of the run, with the date and relevant run number attached to the list.



T.D. is defined as either main wheel touching the runway. If the nose wheel touches first, this fact should be noted on the print-out.

At Break-out Altitude (to be defined) record:

$V_A, h, X_E, Y_E, \psi$ .

At 50' altitude (assuming this is prior to flare initiation) record:

$V_A, h, X_E, Y_E, \psi + \theta, \rho, \alpha, \beta$ .

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At touchdown record:

$$V_A, h, x_E, y_E, \psi, \theta, \phi, \alpha, \beta, \ddot{h}.$$

From 500 feet down to 50 feet calculate the following rms error values.

Glide slope error, localiser error, speed error from reference (or  $\Delta_{\text{error}}$ ), pilot's stick force, pilot's wheel force, pilot's rudder pedal force, thrust vector angle displacement from trim position.

11.2 To help in identifying expected structural loading conditions, during each piloted simulator run a buzzer should sound in the cockpit if the load factor is outside the bounds 0 to 2.0.

11.3 As an aid to check-out of the program, and for producing trim data for various flight conditions, the following data should be added to the PRINT statement:

Dynamic pressure,  $\bar{Q}$  lb/ft<sup>2</sup>

Computed lift coefficient,  $C_L = \frac{W}{\bar{Q} \cdot S_W}$

Aerodynamic lift coefficient,

$$\begin{aligned} C_{LAERO} = & \Delta C_{LWB} + \Delta C_{LLAT.CONT} + C_{LTAIL} \\ & + C_{L\hat{\alpha}} WING \frac{\alpha \cdot \bar{Q} W}{2 V_A} + C_{LQ} WING \frac{Q S_{cw}}{2 V_A} \\ & + \Delta C_{LGROUND EFFECT} \end{aligned}$$

Cold thrust blowing coefficient,  $C_{JTOT} = \frac{T_{CL} + T_{CR}}{\bar{Q} \cdot S_W}$

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Hot thrust blowing coefficient,  $C_{T\text{TOT}} = \frac{T_{\text{HOT L + R}}}{\bar{Q} \cdot S_W}$

Tail incidence,  $\alpha_{\text{TAIL}}$  degrees

Downwash,  $\epsilon$  - degrees

Differential Aileron angle,  $\Delta\delta_a = \frac{\Delta\delta_{aR} - \Delta\delta_{aL}}{2}$

Aileron droop,  $\delta_{a\text{droop}}$  degrees

Augmentor choke,  $\delta_{\text{CH}} = \delta_{\text{CHR}} - \delta_{\text{CHL}}$

Aircraft c.g. position - fraction of m.a.c.

Aircraft weight,  $W$  lbs.

Body mounted normal accelerometer,  $n_{ZB} = -\frac{Z_B}{W}$

Time continuous traces of various parameters should also be taken during each simulated landing. These parameters will be specified at a later date.

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## APPENDIX A

### AN EXPANDED SIMULATION OF THE HYDRAULIC SYSTEM AND THE POWER CONTROL ACTUATORS

It is possible by use of an expanded simulation to investigate the effect on vehicle handling qualities due to hydraulic pump saturation in emergency conditions. Should the Augmentor Wing FTV design become critical in this area, the simulator models in this Appendix will be used to make this evaluation.

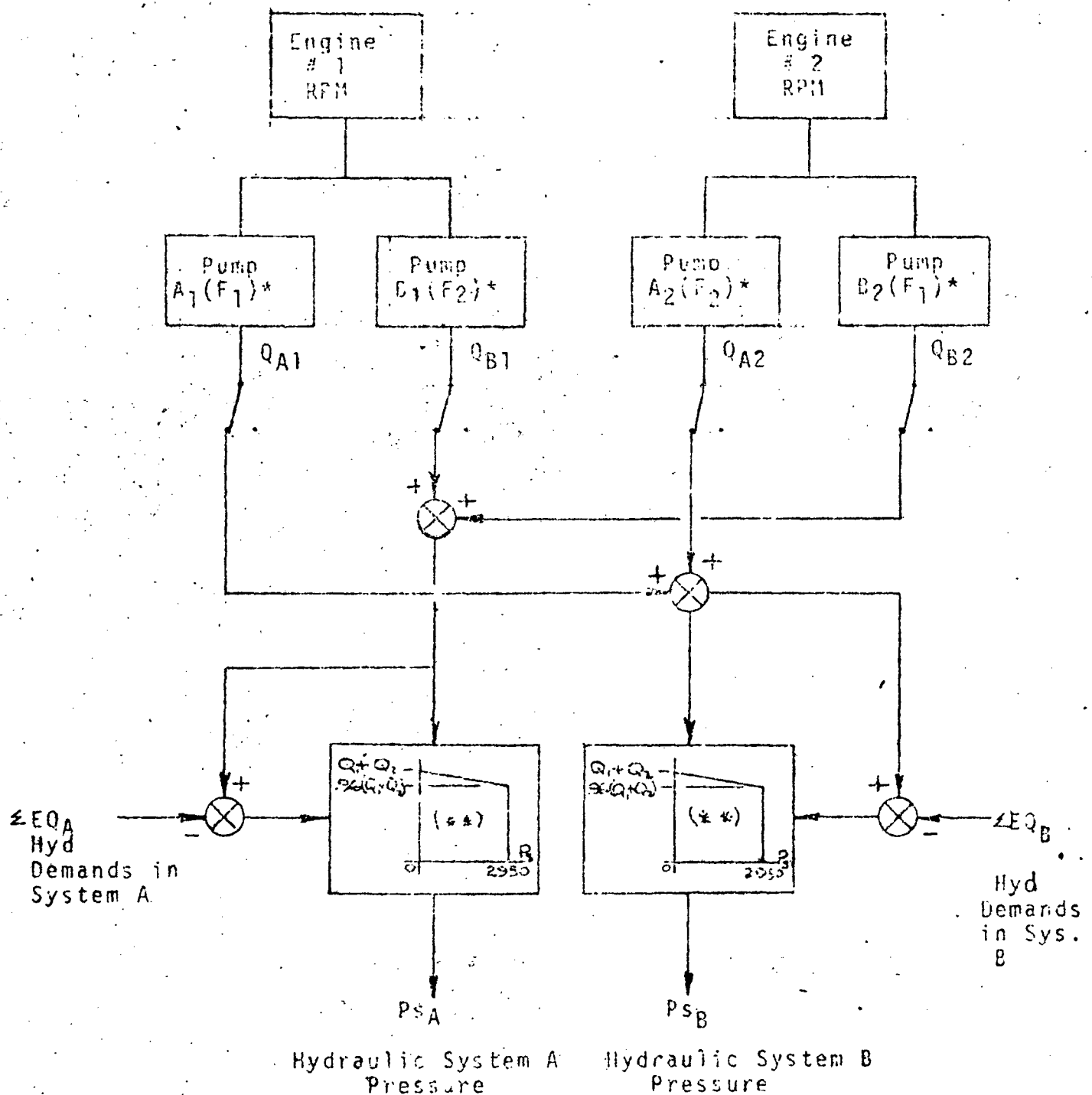
Figure A1 describes the flow available from each engine driven source and computes the system pressures as they depend on hydraulic demands from each system.

Figure A2 schematically shows the summation of all flow requirements in the flying controls system.

Figure A3 shows the type of actuator model which would be used in conjunction with this hydraulic system model. This type of simulation provides an indication of rate and position of the servo output as a function of input signal, available hydraulic supply and actuator loading.

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Figure A,  
HYDRAULIC POWER DISTRIBUTION



$$*F_1 \& F_2 = f(N_p, D_p)$$

$N_p$  = Pump Speed

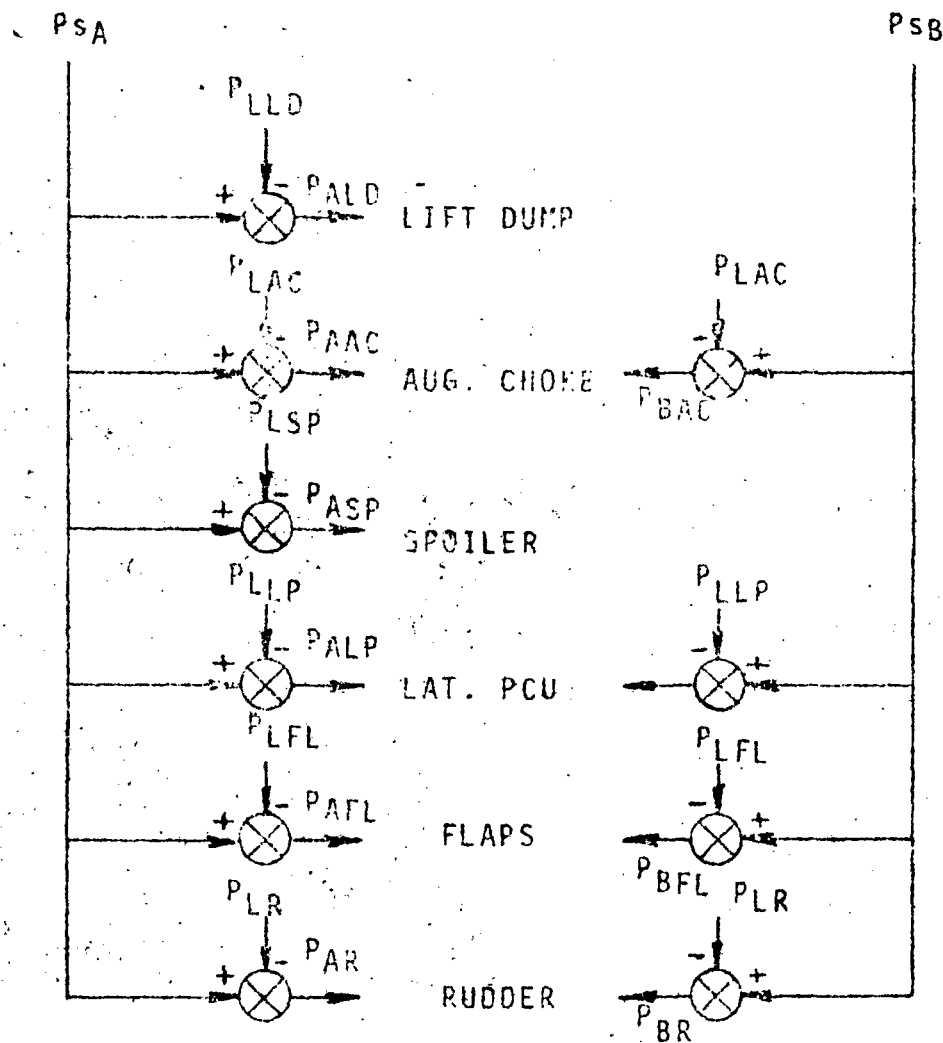
$D_p$  = Pump Displacement

$$** P_S = 2950 \begin{cases} \text{---} \Delta EQ_D = .96(Q_1 + Q_2) \\ \text{---} \Delta EQ_D = 0 \end{cases}$$

$$** P_S = K(Q_1 + Q_2 - EQ_D) \begin{cases} \text{---} \Delta EQ_D = Q_1 + Q_2 \\ \text{---} \Delta EQ_D = .96(Q_1 + Q_2) \end{cases}$$

$$K = \frac{2950}{.04(Q_1 + Q_2)}$$

Figure 12  
HYDRAULIC SYSTEM DISTRIBUTION  
PRESSURE SIGNALS



Where:  $P_L = K (Q)^{1.75} = \text{Line Loss}$

Constants depending on line length  
and diameter

Figure A3  
HYDRAULICS AND SURFACE SITUATION

