THE SIMULATION OF A LARGE JET TRANSPORT AIRCRAFT
Volume I: Mathematical Model

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THE BOEING COMPANY
Wichita, Kans.
for Ames Research Center

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The mathematical models used in the manned simulation of a jumbo jet transport aircraft are described. Included are the models of the basic airframe, the longitudinal lateral and directional control systems, the high lift system, the propulsion system, and the landing gear system. In addition, the low speed buffet characteristics and atmospheric model are described.

Included is a list of necessary tests of the simulation to insure validity of the model, of the computer program, and of the quantitative values.
This report summarizes all work conducted by The Boeing Company under Task II of Contract NAS2-5524, "Design for the Simulation of Advanced Aircraft". The National Aeronautics and Space Administration Technical Monitor was Mr. John Dusterberry of the Simulation Science Division. The Boeing Company Project Leader was Mr. C. Rodney Hanke of the Wichita Division Stability, Control and Flying Qualities Organization. Technical assistance was provided by Mr. Robert A. Curnutt of the 747 Aerodynamics Staff, Everett, Washington.
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INTRODUCTION

The Boeing Company provided NASA-Ames Research Center with mathematical models and data to simulate the flying qualities and characteristics of the Boeing 747 on the NASA Flight Simulator For Advanced Aircraft (FSAA).

The contractual report is divided into two volumes. Volume I includes a description of:

1. The work performed under the contract.
2. Generalized equations and approximations used in the simulation.
3. The form of the data furnished to NASA.
4. Nomenclature used for the report.

Volume II contains only limited rights data. These data are to be retained within the Government until The Boeing Company chooses to treat the data as non-proprietary or until September 15, 1971, whichever occurs first.

Boeing and NASA personnel established ground rules for the simulation program; these ground rules resulted in the following conditions of simulation:

1. Flaps-up and flaps-down data were incorporated into one computer program for simulation capability throughout the complete flight envelope.
2. The significant stability derivatives including control effectiveness derivatives were capable of being modified by multiplication factors (normally equal to 1.0).
3. Hydraulic system failures could be simulated by reduced control capability with the multiplication factors described in Item 2.
4. Data describing aircraft system malfunctions (asymmetric flaps, floating control surfaces, etc.) were provided, however, these data were not incorporated in the simulation.
5. Existing NASA digital computer programs were used where possible.
6. The aerodynamic data cards from the Boeing 747 digital simulation were duplicated for NASA and modified for the digital computer format at NASA-Ames. The stored data were observed visually on a scope as a part of the computer program checkout.
7. A single yaw damper system was programmed to drive both rudder segments.
8. Significant blowdown limits were included in the simulation.
9. The rudder ratio changer (limiter) was incorporated in the simulation. Rudder blowdown data were provided but were not required for the simulation.

10. The existing NASA engine simulation program was modified to simulate the 747 propulsion system. The engine simulation included forward, idle, and reverse thrust with first stage compressor RPM, engine pressure ratio, exhaust gas temperature and windmilling drag characteristics.

11. The existing NASA landing gear simulation program was modified to simulate a simplified model of the 747 landing gear. The load equalization system was accounted for by assuming that takeoff rotation occurred about a point midway between the main wing gear and body gear struts. The mathematical model of the main landing gear did not incorporate body gear steering or wing landing gear tilt.

12. Altitude and airspeed system position errors were neglected.

13. The 747 automatic flap retraction system was not incorporated in the simulation. A description of the system was provided to NASA.

14. The 747 autopilot was not included in the simulation but descriptive data on the autopilot were provided to NASA.

15. Constant fuel load was assumed for any simulation run.
NOMENCLATURE

A, B, C

Constants used in stick force program

A.O.T.

Aft quadrant travel, in.

a

Speed of sound, ft/sec

a₀

Speed of sound, sea level standard, ft/sec

b

Wing span, ft

\( \bar{c} \)

Wing mean aerodynamic chord, ft

\( C_D \)

Airplane drag coefficient, \( C_D = \frac{\text{Drag}}{\rho_D S} \)

\( C_{D\text{Basic}} \)

Basic drag coefficient for the rigid airplane at \( \alpha_{F.R.L} = 0^\circ \), in free air and with the landing gear retracted

\( C_{D_M} \)

Drag coefficient at Mach number

\( C_L \)

Airplane lift coefficient, \( C_L = \frac{\text{Lift}}{\rho_D S} \)

\( C_{L\text{Basic}} \)

Basic lift coefficient for the rigid airplane at \( \alpha_{F.R.L} = 0^\circ \), in free air and with the landing gear retracted

\( C_L (c.g. :25) \)

Change in pitching moment coefficient due to c.g. variation from 25% M.A.C.

\( C_f \)

Airplane rolling moment coefficient, \( C_f = \frac{\text{Rolling Moment}}{\rho_D S b} \)

\( C_m \)

Airplane pitching moment, \( C_m = \frac{\text{Pitching Moment}}{\rho_D S \bar{c}} \)

\( C_{m.25 \text{Basic}} \)

Basic pitching moment coefficient for the rigid airplane at \( \alpha_{F.R.L} = 0^\circ \) in free air, with the landing gear retracted, and with the c.g. = 25% M.A.C.

\( C_n \)

Airplane yawing moment coefficient, \( C_n = \frac{\text{Yawing Moment}}{\rho_D S b} \)
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<tr>
<td>$C_v$</td>
<td>Airplane side force coefficient, $\frac{\text{Side Force}}{q_D S}$</td>
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<tr>
<td>c.g.</td>
<td>Airplane center of gravity position as a fraction of the wing mean aerodynamic chord</td>
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<td>$C_i$</td>
<td>Landing gear damping constant</td>
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<tr>
<td>EGT</td>
<td>Exhaust gas temperature</td>
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<td>EPR</td>
<td>Engine pressure ratio</td>
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<td>$F_{B_i}$</td>
<td>Tire braking force, lb</td>
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<tr>
<td>$F_{GZ}$</td>
<td>Vertical oleo strut force, lb</td>
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<tr>
<td>$F_{RX}$, $F_{RY}$, $F_{RZ}$</td>
<td>Total body axis drag, side, and normal force exerted through the oleo struts, lb.</td>
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<tr>
<td>$F_{RX_{pi}}$, $F_{RY_{pi}}$, $F_{RZ_{pi}}$</td>
<td>Body axis drag, side and normal tire force for each oleo strut, lb</td>
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<td>$F_N$</td>
<td>Net thrust, lb</td>
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<td>$F_{NG}$</td>
<td>Tire force normal to runway, lb</td>
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<tr>
<td>$F_s$, $F_{si}$</td>
<td>Tire side force, lb</td>
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<td>F.Q.T.</td>
<td>Forward quadrant travel, in</td>
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<tr>
<td>FS</td>
<td>Stick force, lb</td>
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<tr>
<td>$F_{S\text{Mass Unbalance}}$</td>
<td>Stick force due to the mass unbalance of the column, lb</td>
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<tr>
<td>FSAA</td>
<td>Flight Simulator for Advanced Aircraft</td>
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<td>$F_{\mu}$, $F_{\mu_i}$</td>
<td>Tire drag force, lb</td>
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<tr>
<td>FST</td>
<td>Flap screw travel</td>
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<tr>
<td>F.U.T.</td>
<td>Feel unit torque, in-lb</td>
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<tr>
<td>$F_{WD}$</td>
<td>Engine Windmilling Drag, lb</td>
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<tr>
<td>$F_{XB_O}$, $F_{YB_O}$, $F_{ZB_O}$</td>
<td>Tire forces in body axis for no pitch or roll rotation, lb</td>
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FXB, FYB, FZB:
  Tire forces in body axis due to pitch rotation only, lb
FXB, FYB, FZB:
  Tire forces in body axis due to pitch and roll rotation, lb
\( g \)
  Acceleration due to gravity, ft/sec^2
GTi, HTi:
  Constants used to determine wheel side force
h:
  Pressure altitude of the airplane c.g., ft
h_{BC,g_i}:
  Vertical distance from c.g. to force vector created by the tires in contact with the runway, ft
h_G:
  Height of the tire relative to the runway, ft
h_r:
  Runway altitude, ft
h_\theta:
  Height of the tire relative to the c.g. due to the pitch angle, ft
h_\phi:
  Change in height of the tire relative to the c.g. due to the bank angle, ft
ICAO:
  International Civil Aviation Organization
K:
  Drag coefficient interpolation factor
KB:
  Braking constant
KBM:
  Maximum value of braking constant
K_\alpha:
  Effectiveness factor for elevator and stabilizer
K_{EGT}:
  Proportionality constant relating exhaust gas temperatures to \( N_1 \)
KT:
  Tire deflection constant
M:
  Mach number
MA:
  Mechanical advantage
M.A.C.:
  Wing mean aerodynamic chord, ft
MRX, MRY, MZR:
  Total body axes moments computed from the body axes forces and their distances from the c.g., ft-lb
N_1:
  Low pressure compressor rotor speed
n_Z:
  Airplane normal load factor along the Z-axis
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<td>$P_f$</td>
<td>Artificial feel unit pressure, lb/in$^2$</td>
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<tr>
<td>$P_o$</td>
<td>Static pressure, sea level standard day, lb/ft$^2$</td>
</tr>
<tr>
<td>$p, q, r$</td>
<td>Roll, pitch and yaw rates about a reference axes system, radians/sec</td>
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<tr>
<td>$q_c$</td>
<td>Impact pressure, $q_c = P_{total} - P_{static}$, lb/ft$^2$</td>
</tr>
<tr>
<td>$q_D$</td>
<td>Dynamic pressure, $q = \frac{1}{2} \rho V^2$, lb/ft$^2$</td>
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<tr>
<td>$r$</td>
<td>Tire radius, ft</td>
</tr>
<tr>
<td>RPM</td>
<td>Revolutions per minute</td>
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<td>$S$</td>
<td>Wing area, ft$^2$</td>
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<td>$\alpha_{F.R.L.}$</td>
<td>Horizontal stabilizer angle relative to the fuselage reference line, degrees</td>
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<td>$T_{AM}$</td>
<td>Ambient temperature, $^\circ R$</td>
</tr>
<tr>
<td>$T_o$</td>
<td>Static temperature, sea level standard day, $^\circ R$</td>
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<tr>
<td>$T_{t_2}$</td>
<td>Total temperature at compressor face, $^\circ R$</td>
</tr>
<tr>
<td>$T_{t_7}$</td>
<td>Turbine discharge total temperature, $^\circ F$</td>
</tr>
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<td>$V$</td>
<td>True airspeed, ft/sec</td>
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<td>$V_c$</td>
<td>Calibrated airspeed, knots</td>
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<tr>
<td>$V_E$</td>
<td>Equivalent airspeed, knots</td>
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<td>$V_{F_i}$</td>
<td>Landing gear spring force, lb</td>
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<tr>
<td>$V_{G}$</td>
<td>Ground speed, knots</td>
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<td>$W$</td>
<td>Airplane weight, lb</td>
</tr>
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<td>$X_{L_i}, Y_{L_i}, Z_{L_i}$</td>
<td>Distance from c.g. to the end of the fully extended landing gear strut, ft</td>
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<tr>
<td>$Y_{E_i}$</td>
<td>Effective inboard engine yawing moment arm, ft</td>
</tr>
<tr>
<td>$Y_{E_O}$</td>
<td>Effective outboard engine yawing moment arm, ft</td>
</tr>
<tr>
<td>$Z_{E_i}$</td>
<td>Effective inboard engine pitching moment arm, ft</td>
</tr>
<tr>
<td>$Z_{E_O}$</td>
<td>Effective outboard engine pitching moment arm, ft</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>Angle of attack relative to fuselage reference line, degrees</td>
</tr>
<tr>
<td>$\alpha_{W.D.P.}$</td>
<td>Airplane angle of attack relative to the wing design plane, degrees</td>
</tr>
<tr>
<td>Symbol</td>
<td>Description</td>
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<tr>
<td>$\beta$</td>
<td>Airplane sideslip angle, degrees</td>
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<tr>
<td>$\beta_G$</td>
<td>Airplane sideslip angle relative to ground velocity vector, degrees</td>
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<td>$\delta_{AM}$</td>
<td>Ambient pressure ratio, $P/P_o$</td>
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<td>$\delta_{A1}$</td>
<td>Inboard aileron deflection angle, degrees</td>
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<td>$\delta_{AO}$</td>
<td>Outboard aileron deflection angle, degrees</td>
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<td>$\delta_B$</td>
<td>Brake pedal deflection, in</td>
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<td>$\delta_{\text{column}}$</td>
<td>Control column deflection angle, degrees</td>
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<td>$\delta_e$</td>
<td>Elevator deflection angle, degrees</td>
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<tr>
<td>$\delta_{e1}$</td>
<td>Inboard elevator deflection angle, degrees</td>
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<tr>
<td>$\delta_{eO}$</td>
<td>Outboard elevator deflection angle, degrees</td>
</tr>
<tr>
<td>$\delta_F$</td>
<td>Flap position</td>
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<td>$\delta_P$</td>
<td>Rudder pedal deflection, in</td>
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<td>$\delta_R$</td>
<td>Rudder deflection angle, degrees</td>
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<td>$\delta_{R\text{command}}$</td>
<td>Commanded rudder deflection angle, degrees</td>
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<tr>
<td>$\delta_{R\text{lower}}$</td>
<td>Lower rudder deflection angle, degrees</td>
</tr>
<tr>
<td>$\delta_{R\text{max}}$</td>
<td>Maximum rudder deflection angle, degrees</td>
</tr>
<tr>
<td>$\delta_{R\text{upper}}$</td>
<td>Upper rudder deflection angle, degrees</td>
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<td>$\delta_S$</td>
<td>Nose wheel steering angle, degrees</td>
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<td>$\delta_{SBH}$</td>
<td>Speed brake handle position, degrees</td>
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<tr>
<td>$\delta_{SP}$</td>
<td>Spoiler deflection angle, degrees</td>
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<tr>
<td>$\delta_T$</td>
<td>Tire deflection, in</td>
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<tr>
<td>$\delta_W$</td>
<td>Control wheel deflection angle, degrees</td>
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<tr>
<td>$\theta$</td>
<td>Temperature ratio, $T/T_o$</td>
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<td>$\theta_B$</td>
<td>Airplane body axis pitch angle, radians</td>
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<tr>
<td>$\theta_{t2}$</td>
<td>Temperature ratio, $T_{t2}/T_o$</td>
</tr>
<tr>
<td>$\phi_B$</td>
<td>Airplane body axis roll angle, radians</td>
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\( \mu_B \)
Coefficient of braking friction

\( \mu_{\text{roll}} \)
Coefficient of rolling friction

\( \frac{\partial \delta_s}{\partial \delta_p} \)
Nose wheel steering gearing constant

\( \Delta S_{T_i} \)
Landing gear oleo strut compression, in

\( \frac{dC_D}{d\alpha} F.R.L. \)
Change in basic drag coefficient due to change in stabilizer angle from \( \alpha_{F.R.L.} = 0^\circ \)

\( \Delta C_{D_{\text{spoilers}}} \)
Change in drag coefficient due to spoiler or speedbrake deflection.

\( \Delta C_{D_{\text{landing gear}}} \)
Change in drag coefficient due to landing gear extension

\( \Delta C_{D_{\text{ground effect}}} \)
Change in drag coefficient due to ground effect

\( \Delta C_{D_{\text{sideslip}}} \)
Change in drag coefficient due to angle of sideslip

\( \Delta C_{D_{\text{rudders}}} \)
Change in drag coefficient due to rudder deflection

\( (\Delta C_L)_{\alpha_{W.D.P.} = 0} \)
Change in basic lift coefficient due to aeroelasticity at \( \alpha_{W.D.P.} = 0^\circ \)

\[ \Delta \left( \frac{dC_L}{d\alpha} \right)_{\alpha_{W.D.P.}} \]
Change in basic lift coefficient due to the aeroelastic effect on the rigid airplane basic lift coefficient curve slope

\[ \frac{dC_L}{d\alpha} \left( \frac{\alpha e}{2V} \right) \]
Change in lift coefficient due to rate of change of angle of attack

\[ \frac{dC_L}{d\dot{q}} \left( \frac{q e}{2V} \right) \]
Change in lift coefficient due to pitch rate

\[ \frac{dC_L}{dnz} \]
Change in lift coefficient due to aeroelastic inertia relief caused by normal load factor

\( K_{\alpha} \frac{dC_L}{d\alpha} F.R.L. \)
Change in lift coefficient due to change in stabilizer angle from \( \alpha_{F.R.L.} = 0^\circ \)

Change in lift coefficient due to change in inboard elevator angle from \( \delta_{e_1} = 0^\circ \)

\( K_{\alpha} \frac{dC_L}{d\delta_{e_0}} \delta_{e_0} \)
Change in lift coefficient due to change in outboard elevator angle from \( \delta_{e_0} = 0^\circ \)
\[ \Delta C_{L_{\text{spoiler}}} \]
Change in lift coefficient due to spoiler or speedbrake deflection

\[ \Delta C_{L_{\text{outboard ailerons}}} \]
Change in lift coefficient due to outboard aileron deflection

\[ \Delta C_{L_{\text{landing gear}}} \]
Change in lift coefficient due to landing gear extension

\[ \Delta C_{L_{\text{ground effect}}} \]
Change in lift coefficient due to ground effect

\[ \frac{dC_{m}}{d\beta} \]
Rolling moment coefficient due to angle of sideslip

\[ \frac{dC_{m}}{d\dot{\beta}} \left( \frac{p_s b}{2V} \right) \]
Rolling moment coefficient due to roll rate about the stability axis

\[ \frac{dC_{m}}{d\ddot{\beta}} \left( \frac{p_s b}{2V} \right) \]
Rolling moment coefficient due to yaw rate about the stability axis

\[ \Delta C_{\beta_{\text{spoiler}}} \]
Rolling moment coefficient due to spoiler deflection

\[ \Delta C_{\beta_{\text{inboard ailerons}}} \]
Rolling moment coefficient due to inboard aileron deflection

\[ \Delta C_{\beta_{\text{outboard ailerons}}} \]
Rolling moment coefficient due to outboard aileron deflection

\[ \Delta C_{\beta_{\text{rudders}}} \]
Rolling moment coefficient due to rudder deflection

\[ \left( \Delta C_{m_{.25}} \right) \alpha_{\text{W.D.P.} = 0} \]
Change in pitching moment coefficient at \( \alpha_{\text{W.D.P.} = 0} \) due to aeroelasticity

\[ \Delta \left( \frac{dC_{m}}{d\alpha} .25 \right) \alpha_{\text{W.D.P.}} \]
Change in pitching moment coefficient due to the aeroelastic effect on the rigid airplane basic pitching moment coefficient curve slope

\[ \frac{dC_{m}}{d\dot{\alpha}} \left( \frac{\dot{\alpha} \bar{c}}{2V} \right) \]
Change in pitching moment coefficient due to rate of change of angle of attack

\[ \frac{dC_{m}}{d\ddot{\alpha}} \left( \frac{\alpha \bar{c}}{2V} \right) \]
Change in pitching moment coefficient due to pitch rate

\[ \frac{dC_{m}}{dn_z} n_z \]
Change in pitching moment coefficient due to aeroelastic inertia relief caused by normal load factor
\[ K \alpha \frac{dC_m}{d\alpha} \cdot 0.25 \text{ F.R.L. Change in pitching moment coefficient due to change in stabilizer angle from } \text{ F.R.L.} = 0^\circ \]

\[ F.R.L = 0^\circ \]

\[ K \alpha \frac{dC_m}{d\delta e_1} \cdot 0.25 \] Change in pitching moment coefficient due to change in inboard elevator angle from \( \delta e_1 = 0^\circ \)

\[ K \alpha \frac{dC_m}{d\delta e_o} \cdot 0.25 \] Change in pitching moment coefficient due to change in outboard elevator angle from \( \delta e_o = 0^\circ \)

\[ \Delta C_m \cdot 0.25 \text{ spoilers} \]

Change in pitching moment coefficient due to spoiler or speed brake deflection

\[ \Delta C_m \cdot 0.25 \text{ inboard ailerons} \]

Change in pitching moment coefficient due to inboard aileron deflection

\[ \Delta C_m \cdot 0.25 \text{ outboard ailerons} \]

Change in pitching moment coefficient due to outboard aileron deflection

\[ \Delta C_m \cdot 0.25 \text{ landing gear} \]

Change in pitching moment coefficient due to landing gear extension

\[ \Delta C_m \cdot 0.25 \text{ ground effect} \]

Change in pitching moment coefficient due to ground effect

\[ \Delta C_m \cdot 0.25 \text{ sideslip} \]

Change in pitching moment coefficient due to angle of sideslip

\[ \Delta C_m \cdot 0.25 \text{ rudders} \]

Change in pitching moment coefficient due to rudder deflection

\[ \frac{dC_n}{d\beta} \] Yawing moment coefficient due to angle of sideslip

\[ \frac{dC_n}{d\beta} \left( \frac{\dot{\beta} b}{2V} \right) \] Yawing moment coefficient due to rate of change of sideslip angle

\[ \frac{dC_n}{d\dot{\beta}} \left( \frac{b \dot{\beta}}{2V} \right) \] Yawing moment coefficient due to roll rate about the stability axis

\[ \frac{dC_n}{d\dot{\phi}} \left( \frac{r \phi}{2V} \right) \] Yawing moment coefficient due to yaw rate about the stability axis

\[ x \]
\[ \Delta C_{n_{\text{spoilers}}} \] Yawing moment coefficient due to spoiler deflection

\[ \Delta C_{n_{\text{inboard ailerons}}} \] Yawing moment coefficient due to inboard aileron deflection

\[ \Delta C_{n_{\text{outboard ailerons}}} \] Yawing moment coefficient due to outboard aileron deflection

\[ \Delta C_{n_{\text{rudders}}} \] Yawing moment coefficient due to rudder deflection

\[ \frac{dC_y}{d\beta} \] Side force coefficient due to angle of sideslip

\[ \frac{dC_y}{d\dot{p}} \left( \frac{p_s b}{2V} \right) \] Side force coefficient due to roll rate about the stability axis

\[ \frac{dC_y}{d\dot{\phi}} \left( \frac{r_s b}{2V} \right) \] Side force coefficient due to yaw rate about the stability axis

\[ \Delta C_{v_{\text{spoilers}}} \] Side force coefficient due to spoiler deflection

\[ \Delta C_{v_{\text{rudders}}} \] Side force coefficient due to rudder deflection

\[ \frac{d}{dt} = \{\cdot\} \] Time derivative operation

**Landing Gear Designation**

\[ i = 1 \] Nose Gear

\[ i = 2 \] Left main landing gear

\[ i = 3 \] Right main gear
GENERAL AIRCRAFT DESCRIPTION

The Boeing 747 is a four-fanjet intercontinental transport designed to operate from existing international airports. High lift for low speed flight is obtained with wing triple-slotted trailing flaps and Krueger type leading edge flaps. The Krueger flaps outboard of the inboard nacelle are cambered and slotted while the inboard Krueger flaps are standard unslotted. A movable stabilizer with four elevator segments provides longitudinal control for the aircraft. The lateral control is obtained with five spoiler panels, an inboard aileron between the inboard and outboard flaps, and an outboard aileron which only operates when the flaps are down. The five spoiler panels on each wing which are used for lateral control also operate symmetrically as speedbrakes in conjunction with the sixth spoiler panel. Directional control is obtained with a two-segment rudder. A general arrangement drawing of the controls and geometric dimensions of the aircraft is shown in Figure 1. A summary of the basic reference areas and geometric dimensions for the simulation is shown in Table 1.
747 THREE-VIEW

FIGURE 1
<table>
<thead>
<tr>
<th>ITEM</th>
<th>VALUE</th>
<th>DIMENSION</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing Area (S)</td>
<td>5500</td>
<td>Ft.²</td>
</tr>
<tr>
<td>Wing Mean Aerodynamic Chord (MAC)</td>
<td>27.3</td>
<td>Ft.</td>
</tr>
<tr>
<td>Wing Span (b)</td>
<td>195.7</td>
<td>Ft.</td>
</tr>
<tr>
<td>Wheel Base</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wing Gear</td>
<td>79</td>
<td>Ft.</td>
</tr>
<tr>
<td>Body Gear</td>
<td>89</td>
<td>Ft.</td>
</tr>
<tr>
<td>Wheel Tread</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wing Gear</td>
<td>36.2</td>
<td>Ft.</td>
</tr>
<tr>
<td>Body Gear</td>
<td>12.5</td>
<td>Ft.</td>
</tr>
<tr>
<td>Pilots Station to .25 MAC (ΔX)</td>
<td>86</td>
<td>Ft.</td>
</tr>
<tr>
<td>Pilots Station to Aircraft Centerline (ΔY)</td>
<td>1.7</td>
<td>Ft.</td>
</tr>
<tr>
<td>Pilots Station to Aircraft c.g. (ΔZ)</td>
<td>10</td>
<td>Ft.</td>
</tr>
</tbody>
</table>

**Summary of Areas and Dimensions**

Table 1
AIRFRAME SIX DEGREE-OF-FREEDOM MODEL

This section contains the general form of equations for computing force and moment coefficients created by aerodynamic loads on the airplane. Each equation is written as a function of its significant variables. The NASA multiplication factors which can be used to modify the basic stability derivatives (ground rule 2 of the Introduction) are not included in these equations. The aerodynamic data in stability axes for these equations are included in Volume II of this report. The dimensionless force and moment coefficients \((C_L, C_D, C_m, C_{\ell}, C_n, \text{and } C_Y)\) were converted into dimensional forces and moments. These forces and moments were used in an existing NASA digital computer program to obtain six degree-of-freedom airframe response. Sign conventions for the control surface deflections and the aerodynamic coefficients are shown in Figure 2.

Lift Force Coefficient

The dimensionless aerodynamic lift force coefficient was computed by the following general equation for a given angle of attack:

\[
C_L = C_{L\text{Basic}} + \Delta C_L \alpha_{W.D.P.} = 0 + \frac{\Delta C_L}{d \alpha} \alpha_{W.D.P.} + \frac{d C_L}{d \alpha} \left( \frac{\alpha}{2V} \right) + \frac{d C_L}{d q} \left( \frac{q \alpha}{2V} \right) + \frac{d C_L}{d n_z} n_z + \frac{K_C d C_L}{d \alpha} \Delta F.R.L.
\]

\[+ K_{\alpha} \frac{d C_L}{d \delta_{\alpha_1}} \delta_{\alpha_1} + K_{\alpha} \frac{d C_L}{d \delta_{\alpha_0}} \delta_{\alpha_0} + \Delta C_L \text{spoilers} + \Delta C_L \text{outboard ailerons} + \Delta C_L \text{landing gear} + \Delta C_L \text{ground effect}.
\]

The aerodynamic lift force data and detailed equations for obtaining the stability derivatives and control surface parameters are included in Section 2, Volume II of this report.
ALL CONTROL SURFACE TRAILING EDGE DOWN DEFLECTIONS ARE POSITIVE
RIGHT WING SPOILER DEFLECTION IS POSITIVE
SIGN CONVENTION
(STABILITY AXES)
FIGURE 2
Drag Force Coefficient

The dimensionless aerodynamic drag force coefficient was computed by the following general equation for a given angle of attack:

\[
C_D = K \left[ C_{D\text{Basic}} + \frac{dC_D}{d\alpha} \right] + [1 - K] \left[ C_D \right]_M + \Delta C_{Ds\text{poilers}} + \Delta C_{D\text{landing gear}} + \Delta C_{D\text{ground effect}} + \Delta C_{Ds\text{sideslip}} + \Delta C_{Dr\text{udders}}.
\]

The aerodynamic drag force data and detailed equations for obtaining the stability derivatives and control surface parameters are included in Section 3, Volume II of this report.

Pitching Moment Coefficient

The dimensionless aerodynamic pitching moment coefficient was computed by the following general equation for a given angle of attack:

\[
C_m = C_{m,25\text{Basic}} + \left( \Delta C_{m,25} \right) \alpha_{W.D.P.} = 0 + \Delta \left( \frac{dC_m}{d\alpha} .25 \right) \alpha_{W.D.P.} + C_L (c.g.-25) + \frac{dC_m}{d\alpha} \left( \frac{\hat{\alpha} \hat{z}}{2V} \right) + K_{\alpha} \frac{dC_m}{d\delta_{e_0}} .25 + K_{\alpha} \frac{dC_m}{d\delta_{e_1}} .25 + K_{\alpha} \frac{dC_m}{d\delta_{e_1}} .25 + \Delta C_{m,25\text{spoilers}} + \Delta C_{m,25\text{inboard ailerons}} + \Delta C_{m,25\text{outboard ailerons}} + \Delta C_{m,25\text{landing gear}} + \Delta C_{m,25\text{ground effect}} + \Delta C_{m,25\text{sideslip}} + \Delta C_{m,25\text{rudders}}.
\]

The aerodynamic pitching moment data and detailed equations for obtaining the stability derivatives and control surface parameters are included in Section 4, Volume II of this report.
Rolling Moment Coefficient

The dimensionless aerodynamic rolling moment coefficient was computed by the following general equation for a given angle of attack:

\[
C_\ell = \frac{dC_\ell}{d\beta} \beta + \frac{dC_\ell}{d\beta} \left( \frac{P_s b}{2V} \right) + \frac{dC_\ell}{d\gamma} \left( \frac{r_s b}{2V} \right) + \Delta C_\ell_{\text{spoilers}} + \Delta C_\ell_{\text{inboard ailerons}} + \Delta C_\ell_{\text{outboard ailerons}} + \Delta C_\ell_{\text{rudders}}.
\]

The aerodynamic rolling moment data and detailed equations for obtaining the stability derivatives and control surface parameters are included in Section 5, Volume II of this report.

Yawing Moment Coefficient

The dimensionless aerodynamic yawing moment coefficient was computed by the following general equation for a given angle of attack:

\[
C_n = \frac{dC_n}{d\beta} \beta + \frac{dC_n}{d\beta} \left( \frac{\dot{\beta} b}{2V} \right) + \frac{dC_n}{d\beta} \left( \frac{P_s b}{2V} \right) + \frac{dC_n}{d\gamma} \left( \frac{r_s b}{2V} \right) + \Delta C_n_{\text{spoilers}} + \Delta C_n_{\text{inboard ailerons}} + \Delta C_n_{\text{outboard ailerons}} + \Delta C_n_{\text{rudders}}.
\]

The aerodynamic yawing moment data and detailed equations for obtaining the stability derivatives and control surface parameters are included in Section 6, Volume II of this report.

Side Force Coefficient

The dimensionless aerodynamic side force coefficient was computed by the following general equation for a given angle of attack:
\[ C_Y = \frac{dC_Y}{d\beta} \beta + \frac{dC_Y}{d\dot{\rho}} \left( \frac{p_s b}{2V} \right) + \frac{dC_Y}{d\dot{\theta}} \left( \frac{r_s b}{2V} \right) \\
+ \Delta C_{Y_{\text{spoilers}}} + \Delta C_{Y_{\text{rudders}}} \]

The aerodynamic side force data and detailed equations for obtaining the stability derivatives and control surface parameters are included in Section 7, Volume II of this report.
LONGITUDINAL CONTROL SYSTEM

General Description

The longitudinal control system consists of two inboard and two outboard elevators and a movable horizontal stabilizer.

Under normal operation, the outboard elevator angle is equal to the inboard elevator angle at speeds below blowdown. The elevators are downrigged 2° from the faired position. The control column is directly linked to the forward quadrant control mechanism. The mechanical rotation of the forward quadrant is transmitted through the control cables to the aft quadrant which controls the elevator actuators. The elevators are fully powered through irreversible hydraulic actuators connected directly to the elevators. Actual elevator deflection is limited by the hydraulic rate, the mechanical stop, or the pressure blowdown.

Control forces are synthesized in an artificial force feel unit which operates as a function of dynamic pressure and stabilizer position.

Pitch trim is available through the movable horizontal stabilizer. The stabilizer position is controlled by thumb switches on the pilot and copilot control wheels. The trim command actuates hydraulic motors which drive a jackscrew mechanism at a rate which is a function of impact pressure.

NASA Longitudinal Control Model and Approximations

A block diagram of the simulated elevator control system and the computed stick force is shown in Figure 3. The column travel in the FSAA was slightly less than the column travel in the 747. For this simulation, the column deflection of the FSAA was scaled up in the computer program such that full deflection of the FSAA column represented full deflection of the 747 column. The control column deflection in the simulator cab was converted into an equivalent forward quadrant angle on the digital computer. Aft quadrant travel was equal to the forward quadrant travel minus the amount due to cable stretch. The cable stretch was proportional to the torque generated by the artificial feel unit. The aft quadrant was converted into an elevator command which was converted into true elevator position through rate and blowdown limiters. The torque of the force feel unit was obtained as a function of aft quadrant travel and feel unit pressure. The feel unit torque was converted into cable stretch and cable load by conversion constants. The stick force produced by the cable load and the mechanical advantage was added to the stick force from the column mass balance to form the computed stick force. The computed stick force was input to the FSAA control loader to provide the pilot with the proper longitudinal force feel characteristics.

The stall warning was provided by a stick shaker model which was activated as a function of angle of attack and flap angle. The data are included in Section 2, Volume II of this report. The frequency and amplitude of the stick shaker model were varied until the evaluation pilot felt the control column "shake" was representative of the 747.

Data for the stick force simulation is presented in Section 8, Volume II of this report.

The stabilizer position was controlled by thumb switches on the pilot and copilot control wheels. A trim command drove the stabilizer trim rate as a function of impact pressure, Figure 4.
I.

NASA DIGITAL COMPUTER

THUMBSWITCH

CAB

F.R.L.

SET BY COMPUTER TRIM PROGRAM

STABILIZER TRIM

FIGURE 4
LATERAL CONTROL SYSTEM

General Description

The lateral control system is a combination of inboard ailerons, outboard ailerons and spoilers. The spoilers can also be used as speedbrakes. Pilot input to the dual tandem central control actuators is provided by a cable system from each of the pilot and copilot control wheels. The actuators drive independent cable systems to the left and right wing lateral control surfaces.

The inboard ailerons operate in all flight regimes while the outboard ailerons operate only when the flaps are down. The outboard ailerons are locked out when the flaps are up.

There are six spoiler panels on each wing. Five panels are modulated by wheel and speedbrake commands and one panel on each wing is controlled by the speedbrake commands. The speedbrake handle is located on the center aisle stand. The five modulated panels on each wing are controlled from “mixer” boxes which sum the inputs from the pilot’s wheel and speedbrake inputs. Wheel inputs will move the modulated spoiler panels up or down within the travel limits at any speedbrake setting.

Control wheel force is synthesized by a spring, as a function of wheel deflection.

NASA Lateral Control Model And Approximations

A block diagram of the simulated lateral control system is shown on Figure 5. The wheel deflection in the simulator cab was converted into an equivalent spoiler and aileron angle on the digital computer. Each spoiler bank deflection was computed separately. The block diagram of Figure 5 illustrates the deflection of only one spoiler panel as a function of wheel deflection for speedbrakes up and down. The commanded spoiler angle was input to the spoiler rate and deflection limiter and the output was the actual deflection for each spoiler panel.

Also shown in Figure 5 is the aileron command as a function of wheel deflection. The ailerons were rate limited. Aileron blowdown was neglected for this simulation. Aileron blowdown information is presented in Section 9 Volume II of this report. On the aircraft, the inboard ailerons are not blowdown limited during normal operations, however, the outboard ailerons can have slightly reduced authority due to blowdown near the flap placard speeds.

For the NASA simulation, wheel forces were approximated by a breakout force and a constant gradient programmed on the FSAA control loader.
LATERAL CONTROL SYSTEM MODEL

FIGURE 5
DIRECTIONAL CONTROL SYSTEM

General Description

The directional control system consists of two rudder segments, upper and lower, each being controlled by a dual tandem actuator. Both rudder segments move together under normal operation. Rudder pedal movement is converted to rudder deflection by cables linking the forward quadrant mechanism to two ratio changers located in the empennage. The rudder ratio changer is programmed to limit maximum rudder travel to ensure safe structural limits at high airspeeds. The rudders are fully powered through irreversible dual source hydraulic actuators which are controlled by the rudder ratio changers. The actual rudder deflection is limited by the hydraulic rate limit, the pressure limit and the ratio changer.

A yaw damper is incorporated which commands rudder proportional to yaw rate and bank angle. For normal operation, the yaw damper drives both rudder segments together. Turn coordination, operative only with flaps down, is achieved by deflecting the rudder through the yaw damper. The yaw damper has its own rudder rate and deflection limits. The rudder commanded by the yaw damper does not result in any movement of the rudder pedals and does not affect the normal operation of the rudder.

Rudder pedal feel forces are provided by mechanical springs and are proportional to rudder pedal movement. Since the ratio changer reduces the rudder deflection with pedal movement as a function of airspeed, the pedal displacement and force to obtain a given rudder deflection is variable with airspeed.

NASA Directional Control Model and Approximations

A block diagram of the simulated rudder control system is shown in Figure 6. The rudder pedal deflection in the cab was converted into a fractional rudder command on the digital computer (\( \delta R / \delta R_{\text{MAX}} \)). The maximum rudder deflection allowed by the rudder ratio changer was programmed as a function of calibrated airspeed. The commanded rudder deflection was obtained by multiplying the fractional rudder command by the maximum allowable rudder angle as determined by the ratio changer. The pilot's rudder command was then rate limited to obtain actual rudder deflection (less yaw damper inputs). For the NASA simulation, the maximum rudder deflection was assumed to be determined by the rudder ratio changer and not the hydraulic pressure limit (blowdown). Under most flight conditions, the actual aircraft rudder is ratio changer limited instead of hydraulic pressure limited. Rudder blowdown data is included in Section 10, Volume II of this report.

For the NASA simulation, rudder pedal forces were approximated by a breakout force and a constant gradient programmed on the FSAA control loader.

The rudder deflection due to the yaw damper system was position and rate limited, and then added to the rudder deflection due to the pilot inputs to obtain the total rudder deflection. The upper and lower rudder angles were both computed by the same mathematical functions. A block diagram with transfer functions of the yaw damper and turn coordinator systems are included in Section 10, Volume II of this report.
DIRECTIONAL CONTROL MODEL

FIGURE 6
HIGH LIFT SYSTEM

General Description

The 747 flap system consists of two inboard and two outboard triple slotted trailing edge flaps and Krueger type leading edge flaps, Figure 7. The Krueger flaps outboard of the inboard nacelle are cambered and slotted while the inboard Krueger flaps are standard unslotted. During extension, the outboard Krueger flaps alter in shape, flexing from a near-flat to a curved surface. The leading edge flaps are either in the full extended or retracted position as a function of the trailing edge flap position.

NASA High Lift Model and Approximations

The 747 flap control lever has seven detents, 0, 1, 5, 10, 20, 25 and 30 for flap position commands. The FSAA flap control lever had five detents. Because position 1 and 5 are leading edge check positions, the five available detents were used to command flap positions 0, 10, 20, 25 and 30. FSAA flap control lever positions between 0 and 10 gave a continuous flap position between 0 and 10. For FSAA flap control lever position greater than 10, flap positions were discretely commanded for the nearest detent position.
A block diagram of the simulated flap control system is shown in Figure 8. The FSAA flap control lever commanded flap screw travel as a function of commanded flap position. The simulated flap screw was driven at a constant rate during flap extension and retraction.

All flaps down aerodynamic data presented in Volume II of this report include effects of leading edge flaps.

Aerodynamic data to simulate asymmetric flap conditions are presented in Volume II of this report. However, these data were not required for the simulation.

The flap auto-retraction system was not included in the simulation.
LOW SPEED BUFFET MODEL

The NASA buffet program was tailored to match the 747 buffet characteristics. Cab buffet intensity was a function of angle of attack, flap position, and spoiler deflection. The intensity associated with angle of attack and spoiler deflection was varied until the pilot felt the buffet was representative of the 747.

Airplane buffet data is presented in Volume II of this report.
PROPULSION SYSTEM

General Description

The 747 aircraft is powered by four Pratt and Whitney JT9D-3 engines having a takeoff thrust rating of 43,500 pounds on a standard sea level day. The JT9D-3 engine is an axial flow, twin spool, high compression, high bypass ratio, turbofan engine. The engines are pod mounted at approximately 40 and 71 percent of the wing semi-span.

Two thrust reversers are provided on each engine, one to reverse the bypass fan exhaust and one to reverse the turbine exhaust. A general description of the engine in normal and reverse thrust modes is shown in Figure 9.

![Figure 9: Diagram of engine in cruise and reverse modes](image)

Both thrust reversers have a translating ring which is located in the outer wall of the nozzle annulus. The plug or centerbody forms the inner wall of the nozzle. During thrust reversal, the translating ring moves to the rear to block the rearward flow of the exhaust and to open the reverser cascades for flow redirection. Tailpipe blockage is accomplished by blocking doors hinged to the translating ring. Engine thrust is controlled from the engine throttle console. The reverse thrust lever is hinged to the forward thrust lever as shown in the schematic of Figure 10. On the 747 airplane, a lockout provision in the thrust lever assembly prevents simultaneous activation of the forward and reverse thrust levers. To obtain reverse thrust, the forward thrust lever must be in the idle position. The
reverse thrust lever is then moved upward to the reverse idle position. A linear time delay is required for the thrust to change from forward idle to reverse idle with no change taking place in the engine parameters (EPR, EGT, etc.).

ENGINE THROTTLE CONSOLE

FIGURE 10

NASA Propulsion Model and Approximations

Engine pressure ratio was computed from the engine throttle angle as shown by the block diagram in Figure 11. The engine throttle angle was converted to engine power lever angle (fuel control angle at the engine). The hysteresis computation between the throttle angle and the power lever angle was neglected for the NASA simulation. The hysteresis in the physical mechanism of the NASA throttle mechanism was assumed to equal the hysteresis between the pilot's throttle and the fuel control on the actual airplane. The static EPR was computed as a function of ambient temperature ($T_{AM}$) for a known power lever angle. The static EPR was input to the NASA engine transient program. An incremental EPR ($\Delta$EPR) was added to the output of the transient program to account for effects of altitude and Mach number. The 747 flight idle limiter was incorporated into the NASA simulation by increasing the actual EPR by a constant when the flaps were in the 25 or greater position and the landing gear was down but not on the ground.

JT9D-3 engine parameter time histories of fast and normal accelerations and decelerations for various airspeeds and altitudes were given to NASA. NASA's engine transient program was modified to approximate these characteristics. The time histories are included in Section 12, Volume II of this report.

The engine thrust simulation is shown in Figure 12. Forward net thrust was computed from EPR, Mach number and ambient pressure ratio. Reverse net thrust was assumed to be only a function of ambient pressure ratio and EPR since thrust reversers are only used from touchdown speed to 100 knots. The thrust mode (forward thrust or reverse thrust) depended upon the positioning of the throttle controls in the cab.
ENGINE PRESSURE RATIO SIMULATION

FIGURE 11
A linear transition with time was used to account for actuation of the thrust reverser mechanism when changing from forward idle thrust to reverse idle thrust.

The engine parameters displayed in the cab were EPR, exhaust gas temperature (EGT) and percent compressor RPM ($%N_1$). A detailed method for generation of EGT and $%N_1$ is shown in Figure 13. The temperature correction as a function of altitude was small and the compressor RPM was roughly proportional to the EGT data. Since the EGT and RPM gages were not essential to the simulation, approximations were made for the NASA simulation as shown in Figure 14.

The detailed 747 engine parameter data and the correlation of NASA thrust simulation data with actual engine data are included in Volume II of this report, Sections 12 and 14 respectively.
PARAMETERS FOR 747 ENGINE GAUGE DISPLAY

FIGURE 13
APPROXIMATIONS FOR NASA ENGINE GAUGE DISPLAY

FIGURE 14
LANDING GEAR SYSTEM

General Description

The 747 employs a four strut main landing gear system with a single strut nose gear, as illustrated in Figure 15.

LANDING GEAR GEOMETRY

FIGURE 15

Two of the main gear struts are wing mounted and two are body mounted. Each main gear strut has a four-wheeled truck. The nose gear strut has two wheels. Because of the geometry of the wheel wells, the wing gear trucks are tilted upward 53 degrees after liftoff for retraction. The body gear trucks are tilted upward 7 degrees for retraction. Since the main landing gears are not aligned longitudinally, a load equalizer is employed to give an effective pitch rotation point midway between the wing and body gear struts.

Nose gear steering is controlled by a tiller and the rudder pedal controls. Full tiller deflection turns the nose gear 70 degrees. Full rudder-pedal travel turns the nose gear 10 degrees.

The rudder pedals command the braking system. A force of approximately 80 pounds is required to obtain maximum braking. Each wheel on the main gear struts is equipped with an anti-skid braking system.

NASA Landing Gear Model and Approximations

The following ground rules were used for the NASA simulation of the 747 landing gear system:

1. The main body and wing gear struts were combined into an equivalent landing gear.
2. Braking capability was provided on the equivalent main gear.
3. Nose wheel steering as a function of rudder pedal travel was incorporated. Nose wheel
steering with a tiller was incorporated but not calibrated to the airplane.

(4) All forces generated by the landing gear contacting the runway were resolved into body axes forces and moments for computation of the airframe response.

(5) Small angle approximations were used for computing landing gear compression, compression rate, and body axes force and moment resolution.

(6) The NASA landing gear simulation was used wherever possible.

The equivalent landing gear model is shown in Figure 16.

![Equivalent Landing Gear Diagram](image)

**EQUIVALENT LANDING GEAR**

**FIGURE 16**

The subscripts 1 through 3 denote the nose gear, main left gear, and main right gear respectively. The subscript "i" in the landing gear equations denotes application of the equation to simulate any one of the three landing gears.

Figures 17 through 20 show progressively in block diagram form the method used for computing landing gear forces and moments in the NASA simulation.

The distance from the c.g. to the landing gear struts was computed as a function of c.g. location as shown in Figure 17. Oleo strut compression ($\Delta S_{Ti}$) was computed by knowing the height of the c.g. ($h$), the body pitch angle ($\theta_B$), the body bank angle ($\Phi_B$), and the coordinates of the landing
DISTANCE FROM CG TO LEFT MAIN GEAR
\[ X_{L2} = -7 + 0.01 (CG - 25) \]
\[ Y_{L2} = -12 \]
\[ Z_{L2} = 17 \]

DISTANCE FROM CG TO NOSE GEAR
\[ X_{L1} = 77 + 0.01 (CG - 25) \]
\[ Y_{L1} = 0 \]
\[ Z_{L1} = 17 \]

DISTANCE FROM CG TO RIGHT MAIN GEAR
\[ X_{L3} = -7 + 0.01 (CG - 25) \]
\[ Y_{L3} = 12 \]
\[ Z_{L3} = 17 \]

OLEO STRUT COMPRESSION AND COMPRESSION RATE

\[ \Delta S_{T_i} = h - h_R + X_{Li} \sin \theta_B - Y_{Li} \sin \phi_B \cos \theta_B - Z_{Li} \cos \phi_B \cos \theta_B \]
\[ \Delta \dot{S}_{T_i} = \dot{h} + X_{Li} \cos \theta_B \dot{\theta}_B + Y_{Li} \left( \sin \phi_B \sin \theta_B \dot{\phi}_B + \cos \phi_B \cos \theta_B \dot{\theta}_B \right) + Z_{Li} \left( \sin \theta_B \cos \phi_B \dot{\phi}_B + \sin \phi_B \cos \theta_B \dot{\theta}_B \right) \]

\[ \Delta S_{T2} \quad \Delta S_{T1} \quad \Delta S_{T3} \]
\[ \Delta \dot{S}_{T2} \quad \Delta \dot{S}_{T1} \quad \Delta \dot{S}_{T3} \]

NOTE: LANDING GEAR DOES NOT CONTACT THE RUNWAY UNLESS \( \Delta S_{T_i} < 0 \)

DETERMINATION OF OLEO STRUT COMPRESSION AND COMPRESSION RATE

FIGURE 17
gear struts. The oleo strut compression equation assumes small angle displacements. The equation for determining oleo strut compression (ΔS_T_i) is derived in its complete form in the Appendix. The rate of compression of the oleo strut (ΔS_T_i) was computed from the first derivative of the oleo strut compression equation.

The vertical oleo strut force (F_{GZ}) resulting from the oleo strut deflection and deflection rate were computed from the non-linear spring force and damping constant data as illustrated in the block diagram of Figure 18. Data for the spring force and damping constant parameter for the equivalent main landing gear and the nose gear are included in Section 13, Volume II, of this report.

The normal force of the tire on the runway generated by the oleo strut force (F_{GZ}) is

\[ F_{NG} = \frac{F_{GZ}}{\cos \theta_B \cos \phi_B} \]

With small angle approximations, runway normal force and oleo strut force were assumed equal. The axes system for the forces generated by the tires on the runway and the relation of these forces to aircraft body axes are shown and derived in the Appendix of this report.

Figure 19 is a block diagram illustrating the method of computing wheel side force. Tire deflection (δ_T) was computed from the runway normal force multiplied by the tire deflection constant (K_T). Wheel side force (F_{si}) was computed from:

\[ F_{si} = G T_i \delta_T_i - H T_i \delta_T_i^2 + (\delta_s - \beta_G) \]

Where:

- \(\beta_G\) = airplane side slip angle relative to ground velocity vector
- \(\delta_s\) = nose wheel steering angle (i = 1 only).

The nose wheel steering angle was calculated from rudder pedal deflection (δ_p) when the aircraft was on the runway. The constants GT_i and HT_i and the nose wheel steering gearing constant \(\partial \delta_s \partial \delta_p\) are included in Section 13, Volume II of this report.

Limiting side force on the wheels was computed from the wheel normal force and a coefficient of sliding friction of 0.6.

Wheel drag force was determined from the rolling coefficient of friction and brake application as shown in the block diagram of Figure 20. Braking was available on the main gear only. The rolling coefficient of friction of 0.015 was assumed in addition to a breakout coefficient of friction varying from 0.014 at zero ground speed to 0 at 10 knots ground speed. The resultant coefficient of friction was combined with the wheel normal force F_N to obtain the gear drag with no braking. Braking was obtained by multiplying brake pedal deflection (δ_B) by a braking constant K_B and the aircraft mass (W/g). The braking force was limited to:

\[ (F_{B_i})_{max} = \mu_B K_B M \frac{W}{g} \]

where the value of \(\mu_B\) depended on the condition of the runway. Braking constants and gearing are included in Section 13, Volume II of this report. Brake and rolling friction are added to obtain the total retarding force on the tire.
DETERMINATION OF OLEO STRUT FORCES

FIGURE 18
NOTE: STEERING ANGLE $\delta S$ IS USED FOR NOSE WHEEL EQUATIONS ONLY

DETERMINATION OF WHEEL SIDE FORCE

FIGURE 19
Determination of wheel drag force

Figure 20

\[ \mu_B = \frac{.4}{w} \text{ (dry runway)} \]
\[ \mu_B = \frac{.1}{w} \text{ (wet runway)} \]

Note: Brakes are used with main gear equations only.

\[ F\mu_i \]
is always negative.

\[ \nu = .015 \]

Rolling friction

\[ \nu = .014 \]

Breakout friction

\[ K_{BM} \]

\[ \frac{w}{g} \]

\[ (F_{Bi})_{MAX} \]

Limiting force

\[ F_{Bi} \]

\[ F_{Ni} \]

\[ V_G \]

\[ \delta_{Bi} \]
Tire normal force, side force, and drag force were computed for each individual oleo strut and resolved into body axes by:

\[ F_{RX_{pi}} = F \mu_i - F_{GZ_i} \theta_B \quad \text{(FOR MAIN GEAR, } i = 2,3) \]

\[ F_{RX_{pi}} = F \mu_i - F_{GZ_i} \theta_B - F_{si} \delta_S \quad \text{(FOR NOSE GEAR, } i = 1) \]

\[ F_{RY_{pi}} = F_{si} + F_{GZ_i} \phi_B \]

\[ F_{RZ_{pi}} = F \mu_i \theta_B - F_{si} \phi_B + F_{GZ_i} \cdot \]

The forces exerted on the aircraft through the oleo struts are obtained by summing the partial body axes forces.

\[ F_{RX} = \sum_{i=1}^{3} F_{RX_{pi}} \]

\[ F_{RY} = \sum_{i=1}^{3} F_{RY_{pi}} \]

\[ F_{RZ} = \sum_{i=1}^{3} F_{RZ_{pi}} \]

The vertical distance from the c.g. to the normal force, side force, and drag force vectors created by the tires in contact with the runway is

\[ h_{Bc.g.i} = 17 + \Delta ST_i \cdot \]

Body axes moments were computed from the partial body axes forces and the distance from the c.g. to the runway.

\[ M_{RX} = \sum_{i=1}^{3} (F_{RZ_{pi}} Y_{Li} - F_{RY_{pi}} h_{Bc.g.i}) \]

\[ M_{RY} = \sum_{i=1}^{3} (-F_{RZ_{pi}} X_{Li} + F_{RX_{pi}} h_{Bc.g.i}) \]

\[ M_{RZ} = \sum_{i=1}^{3} (F_{RY_{pi}} X_{Li} - F_{RX_{pi}} Y_{Li}) \]

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ATMOSPHERE MODEL

The ICAO standard atmosphere model was used to simulate the atmosphere's physical properties throughout the flight envelope of the 747. Table 2 shows the equation used to compute atmospheric temperature ratio ($\theta$) and pressure ratio ($\delta AM$).

### ICAO Standard Atmosphere

<table>
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<th>ALTITUDE &lt;36,089 FT</th>
<th>ALTITUDE &gt;36,089 FT</th>
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<td>$\theta = \frac{T}{T_o} = 1 - 6.875 \times 10^{-6} h$</td>
<td>$\theta = \frac{T}{T_o} = .7518$</td>
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<td>$\delta AM = \frac{P}{P_o} = \theta^{5.256}$</td>
<td>$\delta AM = \frac{P}{P_o} = .2234 e^{-4.806 \times 10^{-5} (h - 36,089)}$</td>
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**TABLE 2**

The sea level standard pressure, temperature, and speed of sound are

- $P_o = 2116.2$ LB/FT$^2$
- $T_o = 518.7^\circ R$
- $a_o = 1116.4$ FT/SEC.

### AIRSPEED EQUATIONS

The speed of sound (ft/sec) was determined from $a = 1116.4 \sqrt{\theta}$. Mach number was computed from the true airspeed obtained from the airframe equations of motion,

$$M = \frac{V}{a}.$$

The compressible dynamic pressure was computed by

$$q_c = 2116.2 \delta AM \left(1 + .2M^2\right)^{3.5-1} \text{ LB/FT}^2$$

from which calibrated airspeed was computed,

$$V_c = \left\{2.187,745 \left(\frac{q_c}{2116.2} + 1\right)^{2/7 - 1}\right\}^{1/2} \text{ KNOTS}.$$
The mathematical models and data of the 747 aircraft were qualitatively and quantitatively checked to substantiate the accuracy and validity of the NASA FSAA simulation. The checkout included verification of the aerodynamic forces and moments, the solution of the equations of motion, and the simulation of flight control system characteristics. Checkout consisted of non-pilot and piloted static and dynamic flight conditions as well as an overall piloted assessment of the simulation. The quantitative results were compared to Boeing simulator and flight test data.

Quantitative non-piloted simulation checkout included the following:

1. Cockpit instruments
2. Atmosphere model
3. Engine characteristics
   a. Forward thrust
   b. Reverse thrust
   c. Transient response
4. Longitudinal trim
5. Configuration changes (trim)
   a. Flaps
   b. Landing gear
   c. Speed brakes
   d. Ground effect
6. Elevator stabilizer trades
7. Dynamics
   a. Short period
   b. Phugoid
   c. Dutch roll
8. Flight controls
   a. Force and displacement
The piloted evaluation of the simulation was conducted by a Boeing and a NASA pilot. The Boeing pilot flew the motion simulator for a total of 5 hours in 3 sessions. The NASA pilot assisted in obtaining the quantitative data as well as becoming familiar with the 747 characteristics.

Piloted simulation checkout included:

A. General qualitative assessment of the following:
   1. Airplane handling characteristics
      a. Dutch roll mode
      b. Spiral mode
      c. Short period mode
      d. Phugoid mode
      e. Roll rate
      f. Climb performance
      g. Flap extension and retraction
      h. Speed brakes
   2. Engine response
   3. Ground effect
   4. Control forces
   5. Takeoff (3 and 4 engine)
   6. Landing
   7. Stall
   8. Air minimum control speed
   9. Buffet
   10. Stick shaker

B. Quantitative evaluation of the following:
1. Takeoff time
2. Climb performance
3. Acceleration and deceleration times in the air
4. Steady turns (elevator and stick force per g)
5. Longitudinal static stability
6. Steady sideslips
7. Roll rates
8. Air minimum control speed

The quantitative results are documented in Section 14, Volume II of this report. In addition to the test data, the pilot comments substantiated the accuracy of the simulation.
THE SIMULATION OF A JUMBO JET
TRANSPORT AIRCRAFT
VOLUME II: MODELING DATA

D6-30643

Prepared by

C. Rodney Hanke and Donald R. Nordwall

THE BOEING COMPANY
Wichita Division Wichita, Kansas

September 1970

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Ames Research Center Moffett Field, California
PREFACE

This report summarizes all work conducted by The Boeing Company under Task II of Contract NAS2-5524, "Design for the Simulation of Advanced Aircraft". The National Aeronautics and Space Administration Technical Monitor was John Dusterberry of the Simulation Sciences Division. The Boeing Company Project Leader was Mr. C. Rodney Hanke of the Wichita Division Stability, Control and Flying Qualities Organization. Technical assistance was provided by Mr. Robert A Curnutt of the 747 Aerodynamics Staff in Everett, Washington.
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<tr>
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<td>7.0-6</td>
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<tr>
<td>Effect of Roll Rate</td>
<td>7.0-7</td>
</tr>
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<td>7.0-8</td>
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1.0 INTRODUCTION

The Boeing Company provided NASA-Ames Research Center with mathematical models and data to simulate the flying qualities and characteristics of the Boeing 747 on the NASA Flight Simulator For Advanced Aircraft (FSAA).

The contractual report is divided into two volumes. Volume I includes a description of:

1. The work performed under the contract.
2. Generalized equations and approximations used in the simulation.
3. The form of the data furnished to NASA.
4. Nomenclature used for the report.

Volume II contains only limited rights data. These data are to be retained within the Government until the Boeing Company chooses to treat the data as non-proprietary or until September 15, 1971, whichever occurs first.

This document has been prepared as a summary of the 747 aerodynamic data for use in flight simulator design. This introductory section contains a description of the 747 including its flight envelope, a general description of the control systems, and a short discussion of the simulation. The following six sections of the document present the airplane aerodynamic characteristics: lift, drag, pitching moment, rolling moment, yawing moment, and side force coefficients. The next three sections describe the control characteristics for pitch, roll and yaw in the various operating modes.

Sections 11, 12 and 13 describe the characteristics of the high lift system, propulsion system and landing gear. The final section contains the results of the simulation checkout.
The appendices contain a portion of the 747 Flight Manual, buffet characteristics, autothrottle, autopilot and revised simulation data.
AIRPLANE DESCRIPTION

The Boeing 747 is a very large four-fanjet intercontinental transport designed to operate from existing international airports. To obtain the necessary low speed characteristics the wing has triple-slotted trailing flaps and Krueger type leading edge flaps. The Krueger flaps outboard of the inboard nacelle are variable cambered and slotted while the inboard Krueger flaps are standard unslotted. The main landing gear consists of a pair of wing mounted four-wheel trucks and a pair of body mounted four-wheel trucks which are slightly aft of the wing. A load equalizing system between the trucks on each side with limited travel allows the center of pitch rotation to be midway between the two pairs of trucks. Longitudinal control is obtained through four elevator segments and a movable stabilizer. The lateral control employs five spoiler panels, an inboard aileron between the inboard and outboard flaps, and an outboard aileron which operates with flaps down only on each wing. The five spoiler panels on each wing also operate symmetrically as speedbrakes in conjunction with the most inboard sixth spoiler panel. Directional control is obtained from two rudder segments. A general arrangement drawing showing these controls and pertinent dimensions is on page 1.1-2. A summary of areas and dimensions necessary for simulation is on page 1.1-3. The airplane operating limits and placards are shown on page 1.1-4.
### SUMMARY OF AREAS AND DIMENSIONS

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<thead>
<tr>
<th>ITEM</th>
<th>VALUE</th>
<th>DIMENSION</th>
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</thead>
<tbody>
<tr>
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<td>Wing Span (b)</td>
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<td>Wing Gear</td>
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<td>Body Gear</td>
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<td>Effective Engine Moment Arms</td>
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<td>Inboard</td>
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</tr>
<tr>
<td>YeI</td>
<td>39.6</td>
<td>Ft.</td>
</tr>
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<td>ZeI</td>
<td>[14.6 (air)]</td>
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<td>[9.2 (ground)]</td>
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<td>Ft.</td>
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<tr>
<td>Outboard</td>
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<tr>
<td>YeO</td>
<td>69.4</td>
<td>Ft.</td>
</tr>
<tr>
<td>ZeO</td>
<td>[5.4 (air)]</td>
<td>Ft.</td>
</tr>
<tr>
<td>[7.3 (ground)]</td>
<td></td>
<td>Ft.</td>
</tr>
</tbody>
</table>

**Note**

The transition between the ground and air values for the effective engine pitching arms, $Z_{EI}$ and $Z_{EO}$, is a function of the averaged main landing gear compression ratio, $\eta$.

For $0 \leq \eta \leq 1$,  
\[ Z_{EO} = Z_{E0\text{Air}} + \eta \Delta Z_{EO} \]
\[ Z_{EI} = Z_{E1\text{Air}} + \eta \Delta Z_{EI} \]

where  
\[ \Delta Z_{EO} = Z_{E0\text{Ground}} - Z_{E0\text{Air}} = 1.9 \text{ Ft.} \]
\[ \Delta Z_{EI} = Z_{E1\text{Ground}} - Z_{E1\text{Air}} = -5.4 \text{ Ft.} \]

and  
\[ \eta = \frac{1}{36} \sum_{n_{EI}} \text{Main Landing Gear Oleo Compression (inches)} \]

SEE SECTION 19 FOR REVISED DATA
1.2 FLIGHT CONTROL SYSTEMS

All primary flight control surfaces are powered by irreversible hydraulic actuators supplied by four independent hydraulic systems. Each segment of the elevators, rudders and ailerons is driven by a single "dual-tandem" type actuator powered by various hydraulic system combinations. The flight and ground spoilers are driven by conventional single cylinder actuators individually supplied by three of the four hydraulic systems. A hydraulic schematic showing the distribution of the four hydraulic systems to the control surface actuators is shown on page 1.2-7.

All control actuators have input overtravel capability to allow unrestricted control input motion to the remaining surfaces when a given actuator is inoperative. There is no manual reversion capability on any surface in the event total hydraulic power is lost.

1.2.1 Longitudinal Controls

The longitudinal control system consists of four elevator segments and a trimmable stabilizer.

Elevators

Each of the four elevator segments is independently powered by dual-tandem actuators. Each inboard elevator actuator is powered by two hydraulic systems and each outboard elevator actuator by one hydraulic system as shown on page 1.2-7. The inboard elevators are controlled from the aft quadrant while...
1.2 the outboard elevators are slaved by a control cable system to
the opposite inboard elevator surface. If hydraulic power to
any segment is lost the segment will trail at an angle where
the hinge moment is zero.

An artificial feel system is used to program the feel forces
which consists of a hydraulic "q" spring modulated with sta-
bilizer setting and a mechanical centering spring. The system
is powered by hydraulic system numbers 2 and 3 normally with
number 1 as a backup to number 3. Any one system will provide
normal feel forces.

Stabilizer

The trimmable stabilizer is actuated by two hydraulic motors
driving a single jackscrew. The power available with one or
both motors is the same, but the trim rate with one motor is one-
half that with both motors operating. Each motor has a rate
control which varies the rate for both motors operating from
0.5 deg./sec. at low speeds to 0.2 deg./sec. at high speed.
Simultaneous control of both motors is obtained either electric-
ally by the thumb switch on each control wheel or mechanically
by the control stand levers which override any electrical input
signal. The autopilot system will operate either hydraulic
motor.

1.2.2 Lateral Controls

The lateral control system comprises a combination of inboard
and outboard ailerons and spoilers which also can be used as
1.2 Speedbrakes. Two dual-tandem central control actuators located in the wheel wells drive independent cable systems to the left and right wing lateral control surfaces. Pilot input to the central control actuators is provided by a cable system from each of the pilot and co-pilot control wheels.

Ailerons

Both inboard and outboard ailerons are actuated by dual-tandem actuators. The inboard ailerons operate in all flight conditions, but the outboard ailerons operate with flaps down only. A lockout mechanism which is actuated electrically by a switch on the outboard flap follow-up linkage, positions the outboard aileron actuators to neutral with flaps up. When both hydraulic systems to any aileron surface are inoperative the surface will trail at an angle where the hinge moment is zero.

Spoilers

There are six spoiler panels on each wing, five which are modulated with lateral control and speedbrake inputs and one (the most inboard panel) which is an unmodulated speedbrake only. Each panel is actuated by a single hydraulic actuator which has a check valve to prevent the panel from floating up to a zero hinge moment angle when the hydraulic system is inoperative. The five modulated panels on each wing are controlled from two "mixer boxes" which sum the inputs from the pilot's speedbrake handle and the central
1.2 control actuators. Lateral control inputs will move the spoiler (Cont'd) panels up or down within the travel limits at any speedbrake setting.

The speedbrake operation of the spoilers is divided into two functions. Moving the speedbrake handle to the "inflight" detent will raise spoiler panels 3, 4, 5 and 8, 9, 10 which are controlled through one mixer box to full deflection or blowdown angle. Before the "inflight" detent is reached, moving the speedbrake handle will raise spoiler panels 6 and 7 which are controlled directly by a two position solenoid valve, to full deflection or blowdown angle. Further handle movement to the ground detent position is possible only on the ground and will raise the remaining panels through the other mixer box.

Wheel forces are provided with a simple spring loaded follower and cam arrangement. Trim is obtained by rotating this mechanism with an electric servo motor and shifting the zero wheel force datum to the wheel angle desired. The servo motor is operated by a switch on the control stand.

1.2.3 Directional Controls

The directional control system consists of two rudder segments, each being actuated by a dual-tandem actuator. Rudder limiting is provided by a "q" programmed gear ratio changer which limits the rudder available from full pedal travel. Each rudder has a ratio changer with a comparator circuit to monitor their operation. If the two ratio changers disagree beyond the system tolerance
limits, a warning light in the cockpit is activated. There are certain conditions such as one hydraulic system off operation where rudder available will be limited by actuator force capability to smaller angles than the ratio changer allows.

The rudder pedal forces are programmed by a spring loaded follower and cam arrangement similar to the lateral feel system. However, the ratio changer varies the pedal force required to obtain a given rudder angle with airspeed. This is done by installing the ratio changer between the feel system and the actuator input.

Rudder trim is obtained by rotating the feel unit with the trim knob on the control stand. The trim authority is also a function of airspeed due to the ratio changer function.

A series type yaw damper and turn coordinator system is incorporated into each rudder actuator. Rudder inputs from these systems will add to the inputs from the pedals or trim but will not feed back through the control system.

1.2.4 Flaps

The flap system consists of leading edge flaps and trailing edge flaps as shown on page 1.1-2.

The leading edge flaps comprise four sets per wing, each set being powered by a separate air motor or, as a backup, by a separate electric motor. The leading edge flaps are two-positioned (fully retracted or extended) and are programmed to operate in conjunction...
1.2 with the trailing edge flaps. Group A comprising leading edge flap sets (6, 7, 8), (11, 12, 13) / (14, 15, 16), (19, 20, 21) extend fully when the outboard trailing edge flaps extend to flaps 1. As the inboard trailing edge flaps extend to flaps 5, group B comprising the remaining leading edge flap sets (1, 2, 3, 4, 5), (9, 10) / (17, 18), (22, 23, 24, 25, 26) extend fully.

The two inboard trailing edge flaps are actuated by one power drive system, the two outboard by another. Each drive system consists of a hydraulic motor (see page 1.2-7) coupled through gears to a torque tube extending laterally along both wings. Each trailing edge flap is driven by two ball screw actuators powered by the turning of the appropriate torque tube. Each drive system has an electric motor for backup power. During trailing edge flap extension or retraction, an inboard trailing edge flaps asymmetry monitor automatically causes hydraulic shutoff to the inboard trailing edge flaps when the position difference between the right and left inboard trailing edge flaps exceeds a predetermined amount. An outboard trailing edge flaps asymmetry monitor operates similarly.
1.3 DISCUSSION OF THE SIMULATION

An aircraft in motion is acted upon by external forces and moments resulting from thrust and gravity effects, landing gear forces, and aerodynamic loads. These force and moment components comprise the coefficients of the airplane equations of motion, which are the key to a realistic description of the aircraft's flight characteristics.

Aircraft flight simulation, then, requires continuous, real-time solution of these equations of motion, as well as an accurate representation of those systems and characteristics necessary to allow the pilot to "fly" the simulator with sufficient realism.

This document contains data which describe the forces and moments created by aerodynamic loads on the airplane. They may be functions of several variables, including altitude, airspeed, Mach number, angle of attack, rotation rates, center of gravity, ground proximity and geometry changes, such as control deflections and gear and flap extensions.

The data contained in the following sections are computed about stability axes \((x_S, y_S, z_S)\) as shown on page 1.3-4. Stability axes differ from body axes in that the \(x\)- and \(z\)-axes are rotated about the \(y\)-axis through the angle of attack; that is, the \(x_S\)-axis lies in the plane determined by the relative wind and the body \(y\)-axis. This \(x_S\)-axis also lies in the \((x-z)\) plane of symmetry of the
1.3 airplane and is thus rotated about $z_s$ away from the relative wind (Cont'd) by sideslip angle, $\beta$. Forces and moments measured in this system are presented as dimensionless coefficients which are broken down into separate terms (stability derivatives) showing the effects of each important parameter.

While the use of stability axes simplifies the presentation of the aerodynamic functions, the forces and moments must be resolved to the appropriate axes systems for solution of the equations of motion. The schematic flow chart on Page 1.3-5 presents the method used to solve the dynamic equations in which the aerodynamic forces and moments are transformed into body axes. Note that the appropriate axis transformation here is to the fuselage reference line (FRL) body axes, or through the angle $\alpha_{F.R.L.}$. This allows the direct use of body axis inertias, $I_{xx}$, $I_{yy}$, $I_{zz}$, and $I_{xz}$ without any inertia transformations. Note also that the wing angle of attack ($\alpha_{W.D.P.} = \alpha_{F.R.L.} + 2^\circ$) is used only in conjunction with the aerodynamic data curves - it is not used for any axis transformations.

A second item of importance is the simulation of forces and moments due to thrust. Flight testing has shown that a simple effective engine moment-arm representation accounts for thrust.
1.3 At present, the thrust vector may be considered to be inclined up (Cont'd) 2.5° from the fuselage reference line, with each engine canted inward by 2°. The corresponding thrust effects are as follows:

\[
T_x = T_{\text{eng.} \#1} + T_{\text{eng.} \#2} + T_{\text{eng.} \#3} + T_{\text{eng.} \#4}
\]

\[
T_y = 0.0349 \left( T_{\#1} + T_{\#2} - T_{\#3} - T_{\#4} \right)
\]

\[
T_z = -0.0436 \, T_x
\]

\[
L_T = 0.0436 \, N_T
\]

\[
M_T = \left( T_{\#1} + T_{\#4} \right) \cdot Z_{E_0} + \left( T_{\#2} + T_{\#3} \right) \cdot Z_{E_1}
\]

\[
N_T = \left( T_{\#1} + T_{\#4} \right) \cdot Y_{E_0} + \left( T_{\#2} - T_{\#3} \right) \cdot Y_{E_1}
\]

The effective engine moment arm values are found on page 1.1-3.

The thrust reverser effects on the lift, drag and pitching moment coefficients are presented in Section 12.5. These increments are to be added to the equations for \( C_L \), \( C_D \) and \( C_{m_{c.g.}} \) on pages 2.0-1, 3.0-1 and 4.0-1 respectively.

Sign conventions for the controls and aerodynamic coefficients are shown on page 1.3-6, while maximum control deflections are listed on page 1.3-7.

To aid in scaling, a list of maximum values is given on page 1.3-8, and a summary of nomenclature begins on page 1.3-9.
x, y, z - BODY AXES (FRL)
x_w, y_w, z_w - WIND AXES
x_s, y_s, z_s - STABILITY AXES

AXES ORIGIN AT THE
CENTER OF GRAVITY
POSITIVE STABILIZER DEFLECTION - L.E. UP
FOR INDIVIDUAL AILERONS, + θA - T.E. DOWN
FOR AILERONS ON BOTH WINGS, + θA
R.H. WING T.E. DN.
R.H. WING T.E. UP

RIGHTWING SPOILER DEFLECTION IS POSITIVE.

SIGN CONVENTION
(STABILITY AXES)

THE BOEING COMPANY
RENTON, WASHINGTON
## MAXIMUM CONTROL SURFACE DEFLECTION AND RATES

<table>
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<tr>
<th>Control Surface</th>
<th>Symbol</th>
<th>Maximum Displacement (deg)</th>
<th>Normal Operation Rate (deg/sec)</th>
<th>One Hydraulic System Failure Rate (deg/sec)</th>
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<td>Elevators</td>
<td>θₑᵢ</td>
<td>+17, -23</td>
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<td>30 down</td>
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<td></td>
<td></td>
<td></td>
<td>37 up</td>
<td>26 up</td>
</tr>
<tr>
<td>Stabilizer</td>
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<td>0.25→0.1</td>
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<td>Pilot’s Thumb</td>
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<td>0.5→0.2</td>
<td>0.25→0.1</td>
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<td>27 down</td>
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<td></td>
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<td>45 up</td>
<td>35 up</td>
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<td>22 down</td>
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<td></td>
<td></td>
<td>55 up</td>
<td>45 up</td>
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<td>9,10,11,12</td>
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<td>Panels 6,7 (Speedbrakes only)</td>
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<td>θᵣᵤ</td>
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<tr>
<td>Lower</td>
<td>θᵣₗ</td>
<td>±25</td>
<td>50</td>
<td>40</td>
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</table>

**Note:** The deflection rates shown above are average values applicable to all flight conditions. The actual non-linear deflection rate response characteristics are presented in D6-13336, "Flight Control Systems Data For The 747 Flight Simulator".

1 NASA simulation used a control wheel rate limit of 100 deg/sec in place of individual control rate limits.
### MAXIMUM VALUES

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<th>Parameter</th>
<th>Maximum Value</th>
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<td>$\dot{\phi}$</td>
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<tr>
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**NOTE:** For complete center of gravity information consult D6-13585, "Mass Properties - 747 Flight Simulator".
# NOMENCLATURE

<table>
<thead>
<tr>
<th>SYMBOL</th>
<th>DEFINITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>$a_i$</td>
<td>Acceleration along the &quot;i&quot; axis (\text{ft/sec}^2)</td>
</tr>
<tr>
<td>$b$</td>
<td>Wing span (ft).</td>
</tr>
<tr>
<td>$C_D$</td>
<td>Airplane drag coefficient. $C_D = \frac{D}{qS}$, where a positive drag force acts along the negative $x_s$ axis.</td>
</tr>
<tr>
<td>$C_L$</td>
<td>Airplane lift coefficient. $C_L = \frac{L}{qS}$, where a positive lift force acts along the negative $z_s$ axis.</td>
</tr>
<tr>
<td>$C_Y$</td>
<td>Airplane side force coefficient. $C_Y = \frac{Y}{qS}$, where a positive side force acts along the positive $y_s$ axis.</td>
</tr>
<tr>
<td>$C_m$</td>
<td>Airplane rolling moment coefficient about the reference axis, $x_s$. $C_m = \frac{M}{qSb}$</td>
</tr>
<tr>
<td>$C_n$</td>
<td>Airplane pitching moment coefficient about the reference axis, $y_s$. $C_n = \frac{N}{qSb}$</td>
</tr>
<tr>
<td>$C_s$</td>
<td>Airplane yawing moment coefficient about the reference axis, $z_s$. $C_s = \frac{N}{qSb}$</td>
</tr>
<tr>
<td>C.G.</td>
<td>Airplane center of gravity position as a fraction of the wing mean aerodynamic chord.</td>
</tr>
<tr>
<td>$\varepsilon$, M.A.C.</td>
<td>Wing mean aerodynamic chord (ft).</td>
</tr>
<tr>
<td>$F_P$</td>
<td>Rudder pedal force, positive for a left rudder pedal force (lb).</td>
</tr>
<tr>
<td>$F_s$</td>
<td>Control column force, positive for a column pull force (lb).</td>
</tr>
<tr>
<td>$F_w$</td>
<td>Control wheel force, positive for a clockwise wheel moment (lb).</td>
</tr>
<tr>
<td>SYMBOL</td>
<td>DEFINITION</td>
</tr>
<tr>
<td>--------</td>
<td>------------</td>
</tr>
<tr>
<td>HM</td>
<td>Hinge moment (lb-ft).</td>
</tr>
<tr>
<td>h_p</td>
<td>Pressure altitude (ft).</td>
</tr>
<tr>
<td>i,j</td>
<td>Airplane mass moment of inertia about the reference axes i, j (slug-ft²).</td>
</tr>
<tr>
<td>L,M,N</td>
<td>Rolling, pitching and yawing moments about a reference axes system.</td>
</tr>
<tr>
<td>M</td>
<td>Mach number.</td>
</tr>
<tr>
<td>n_z</td>
<td>Airplane normal load factor along the z-axis.</td>
</tr>
<tr>
<td>P, q, r</td>
<td>Roll, pitch and yaw rates about a reference axes system (radians/sec).</td>
</tr>
<tr>
<td>q</td>
<td>Dynamic pressure (lb/ft²).</td>
</tr>
<tr>
<td>q_c</td>
<td>Impact pressure (lb/ft²).</td>
</tr>
<tr>
<td>S</td>
<td>Wing area (ft²).</td>
</tr>
<tr>
<td>T</td>
<td>Engine thrust (lb).</td>
</tr>
<tr>
<td>V</td>
<td>True airspeed (ft/sec).</td>
</tr>
<tr>
<td>V_c</td>
<td>Calibrated airspeed (knots).</td>
</tr>
<tr>
<td>V_e</td>
<td>Equivalent airspeed (knots).</td>
</tr>
</tbody>
</table>

\[
V_c = \frac{1479.1026 \sqrt{(q_c/2116.2166 + 1)^{(2/7)}} - 1}{0.002}
\]

\[
V_e = \sigma^{1/2} \cdot V / 1.689
\]

\[
n_z = \frac{Z_s \cos \alpha - X_s \sin \alpha + 0.0456T - \{Z_{due to landing gear reactions}\}}{mg}
\]
<table>
<thead>
<tr>
<th>SYMBOL</th>
<th>DEFINITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>W</td>
<td>Airplane gross weight (lb).</td>
</tr>
<tr>
<td>X, Y, Z</td>
<td>Axial force, side force, and normal force along a reference system.</td>
</tr>
<tr>
<td>x, y, z</td>
<td>Body axes system described in Section 1.3.</td>
</tr>
<tr>
<td>x_s, y_s, z_s</td>
<td>Stability axes system described in Section 1.3.</td>
</tr>
<tr>
<td>x_w, y_w, z_w</td>
<td>Wind axes system described in Section 1.3.</td>
</tr>
<tr>
<td>y_e_i</td>
<td>Effective inboard engine yawing arm (ft).</td>
</tr>
<tr>
<td>y_e_o</td>
<td>Effective outboard engine yawing arm (ft).</td>
</tr>
<tr>
<td>z_e_i</td>
<td>Effective inboard engine pitching arm (ft).</td>
</tr>
<tr>
<td>z_e_o</td>
<td>Effective outboard engine pitching arm (ft).</td>
</tr>
<tr>
<td>α, α_f_r_l</td>
<td>Airplane angle of attack relative to the fuselage reference line (degrees).</td>
</tr>
<tr>
<td>α_w_d_r</td>
<td>Airplane angle of attack relative to the wing design plane (degrees).</td>
</tr>
<tr>
<td>β</td>
<td>Airplane sideslip angle (degrees).</td>
</tr>
<tr>
<td>δ</td>
<td>Air pressure ratio.</td>
</tr>
<tr>
<td>δ_a</td>
<td>Aileron deflection angle (degrees).</td>
</tr>
<tr>
<td>δ_a_i</td>
<td>Inboard aileron deflection angle (degrees).</td>
</tr>
<tr>
<td>δ_a_o</td>
<td>Outboard aileron deflection angle (degrees).</td>
</tr>
<tr>
<td>δ_c</td>
<td>Control column deflection angle, positive for rearward column movement (degrees).</td>
</tr>
<tr>
<td>δ_e</td>
<td>Elevator deflection angle (degrees).</td>
</tr>
<tr>
<td>δ_e_i</td>
<td>Inboard elevator deflection angle (degrees).</td>
</tr>
<tr>
<td>δ_e_o</td>
<td>Outboard elevator deflection angle (degrees).</td>
</tr>
<tr>
<td>δ_f</td>
<td>Flap setting</td>
</tr>
<tr>
<td>SYMBOL</td>
<td>DEFINITION</td>
</tr>
<tr>
<td>--------</td>
<td>------------</td>
</tr>
<tr>
<td>$\delta_p$</td>
<td>Rudder pedal deflection angle, positive for positive rudder deflection (degrees).</td>
</tr>
<tr>
<td>$\delta_R$</td>
<td>Rudder deflection angle (degrees).</td>
</tr>
<tr>
<td>$\delta_{RL}$</td>
<td>Lower rudder deflection angle (degrees).</td>
</tr>
<tr>
<td>$\delta_{RU}$</td>
<td>Upper rudder deflection angle (degrees).</td>
</tr>
<tr>
<td>$\delta_{SP}$</td>
<td>Spoiler deflection angle (degrees).</td>
</tr>
<tr>
<td>$\delta_W$</td>
<td>Control wheel deflection angle, positive for a clockwise wheel movement (degrees).</td>
</tr>
<tr>
<td>$\alpha_{F.R.L.}$</td>
<td>Horizontal stabilizer angle relative to the fuselage reference line (degrees).</td>
</tr>
<tr>
<td>$\Theta$</td>
<td>Airplane Euler pitch angle.</td>
</tr>
<tr>
<td>$\Phi$</td>
<td>Airplane Euler roll angle.</td>
</tr>
<tr>
<td>$\Psi$</td>
<td>Airplane Euler yaw angle.</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Air mass density (slugs/ft$^3$).</td>
</tr>
<tr>
<td>$\sigma$</td>
<td>Air density ratio</td>
</tr>
</tbody>
</table>

**SUBSCRIPT**

| F.R.L. | Fuselage reference line (any body water line). |
| W.D.P. | Wing design plane. |

As used with $V_E$ and $M$ placards:—

- **FE** = Operational flaps extended placard.
- **LO** = Landing gear operating placard.
- **LE** = Landing gear extended placard. (Note: The landing gear cannot be extended above the $LO$ limit, but flight to $LE$ limits is possible).
SUBSCRIPT

MO = Maximum operating limit.
DF = Design dive flight placard.

The time derivative operation normally denoted by \( \frac{d}{dt} \) is replaced in this document by the dot derivative notation,

\[
\frac{d}{dt} = \dot{}(\cdot) \quad \text{AND} \quad \frac{d}{dt^2} = \ddot{}(\cdot).
\]
2.0 LIFT FORCE COEFFICIENT

The dimensionless aerodynamic lift force coefficient is given in terms of its significant components by the equation below.

At a given $\alpha_{W.D.P.},$

$$C_L = C_{L_{\text{Basic}}} + (AC_{L})_{\alpha_{W.D.P.}=0^\circ} + \Delta \left( \frac{dC_L}{d\alpha} \right) \cdot \alpha_{W.D.P.}$$

$$+ \frac{dC_L}{d\alpha} \cdot \left( \frac{\alpha E}{2V} \right) + \frac{dC_L}{d\alpha} \cdot \left( \frac{\alpha}{2V} \right) + \frac{dC_L}{dn_x} \cdot n_x$$

$$+ K_{\alpha} \frac{dC_L}{d\alpha} \cdot \delta_{F.R.L.} + K_{\alpha} \frac{dC_L}{d\delta_{x_1}} \cdot \delta_{E_1} + K_{\alpha} \frac{dC_L}{d\delta_{E_0}}$$

$$+ \Delta C_{L_{\text{Spoilers}}} + \Delta C_{L_{\text{Outboard Ailerons}}} + \Delta C_{L_{\text{Landing Gear}}}$$

$$+ \Delta C_{L_{\text{Ground Effect}}} + \left[ \Delta C_{L_{\text{Flap Failure}}} \right]^{(*)}$$

where,

$$C_{L_{\text{Basic}}}$$ = Basic lift coefficient for the rigid airplane at $\alpha_{F.R.L.}=0^\circ$, in free air and with the landing gear retracted. For low speed, $C_{L_{\text{Basic}}}$ is plotted on page 2.0-7. For flaps up, $C_{L_{\text{Basic}}}$ is plotted on page 2.0-8.

$$(AC_{L})_{\alpha_{W.D.P.}=0^\circ}$$ = Change in basic lift coefficient at $\alpha_{W.D.P.}=0^\circ$ due to aeroelasticity. For low speed, $$(AC_{L})_{\alpha_{W.D.P.}=0^\circ}$$

(*) NOT IN NASA SIMULATION
2.0

(Cont'd)

is plotted on page 2.0-9. For flaps up, \((\Delta C_L)_{\alpha_{W.D.R.}} = 0^\circ\)
is plotted on page 2.0-10.

\[ \Delta \left( \frac{dC_L}{d\alpha} \right)_{\alpha_{W.D.R.}} = \text{Change in basic lift coefficient due to the aero-elastic effect on the rigid airplane basic lift coefficient curve slope.} \]

For low speed, \(\Delta \left( \frac{dC_L}{d\alpha} \right)\) is plotted on page 2.0-11. For flaps up, \(\Delta \left( \frac{dC_L}{d\alpha} \right)\) is plotted on page 2.0-12.

\[ \frac{dC_L}{d\alpha} \left( \frac{g\bar{a}}{2V} \right) = \text{Change in basic lift coefficient due to rate of change of angle of attack.} \]

is plotted on page 2.0-13.

\[ \frac{dC_L}{d\varphi} \left( \frac{g\bar{a}}{2V} \right) = \text{Change in basic lift coefficient due to pitch rate.} \]

\[ \frac{dC_L}{d\varphi} = K_q \cdot \frac{dC_L}{d\varphi} \]

where \(\frac{dC_L}{d\varphi}\) and the center of gravity factor, \(K_q\) are plotted on page 2.0-14.

\[ \frac{dC_L}{dn_z} \cdot \eta_3 = \text{Change in basic lift coefficient due to aeroelastic inertia relief caused by normal load factor, } \eta_3. \]

For low speed, \(\frac{dC_L}{dn_z}\) is plotted on page 2.0-15.

For flaps up, \(\frac{dC_L}{dn_z}\) is plotted on page 2.0-16.

\[ K_\alpha \cdot \frac{dC_L}{d\phi} \cdot \Delta_{F.R.L.} = \text{Change in basic lift coefficient due to change in stabilizer angle from } \Delta_{F.R.L.} = 0^\circ. \]

is plotted on page 2.0-17. The effectiveness factor for the
stabilizer (and elevators), $K_{\alpha}$ is plotted on page 4.0-19.

$K_{\alpha} \frac{dC_L}{d\delta_{EI}} = \text{Change in basic lift coefficient due to change in inboard elevator angle from } \delta_{EI} = 0^\circ . \frac{dC_L}{d\delta_{EI}}$ is plotted on page 2.0-18.

$K_{\alpha} \frac{dC_L}{d\delta_{EO}} = \text{Change in basic lift coefficient due to change in outboard elevator angle from } \delta_{EO} = 0^\circ . \frac{dC_L}{d\delta_{EO}}$ is plotted on page 2.0-19.

The normal system stick free (rigged) elevator deflection is $+2^\circ$ from the faired position.

$\Delta C_L_{\text{SPOILERS}} = \text{Change in basic lift coefficient due to spoiler or speedbrake deflection. It should be noted that "spoilers" extended on one wing are used for lateral control, while symmetrically extended spoilers are used for "speed brakes."}$

$\Delta C_L_{\text{SPOILERS}} = \sum_{\text{OPERATING SPOLIER PANELS}} (K_{\delta_{SP}})(\Delta C_{L_{SP}})_{AS} \left(\frac{C_{L_{SP}}}{C_{L_{SP}}_{M=0}}\right)_R F_{\delta_{SP}}$

where $(\Delta C_{L_{\text{SP}}})_{45}$ is the change in basic lift coefficient due to deflecting the operating spoiler panels to $45^\circ$. The operating spoiler panels are determined from the hydraulic systems schematic on page 1.2-7. $(\Delta C_{L_{\text{SP}}})_{45}$ is plotted for spoilers and ground spoilers on page 2.0-21 and page 2.0-22.
respectively. The spoiler effectiveness factor, 
$$K_{SP}$$, is plotted on page 2.0-20. The Mach number 
effect, $$\frac{(C_{LSP})_M}{(C_{LSP})_{Ma=0}}$$, is plotted on page 
2.0-23. The aeroelastic effect, $$\frac{E}{L_{SP}}$$, is plotted 
on pages 2.0-24, 2.0-25, and 2.0-26. The ground 
effect factor, $$F_{GE}$$, is obtained from page 5.0-29.

$$\Delta C_{OUTBOARD AILERONS} =$$ Change in basic lift coefficient due to outboard 
aileron deflection.

$$\Delta C_{OUTBOARD AILERONS} = \sum \left( K_{SAO} \cdot \Delta C_{LAO} \cdot F_{GE} \right)$$

where $$\Delta C_{LAO}$$ is the change in basic lift coefficient 
due to deflecting one outboard aileron up to 25° or 
the opposite outboard aileron down to 15°. $$\Delta C_{LAO}$$ 
is plotted on page 2.0-27. The outboard aileron 
effectiveness factor, $$K_{SAO}$$, is plotted on page 
5.0-26. The ground effect factor, $$F_{GE}$$, is obtained 
from page 5.0-29.

$$\Delta C_{LANDING GEAR} =$$ Change in basic lift coefficient due to main and nose 
landing gear extension.

$$\Delta C_{LANDING GEAR} = K_{GEAR} \cdot \Delta C_{GEAR} \cdot \frac{(C_{LGEAR})_M}{(C_{LGEAR})_{Ma=0}}$$

where $$\Delta C_{GEAR}$$ is plotted on page 2.0-29. The Mach 
number effect, $$\frac{(C_{LGEAR})_M}{(C_{LGEAR})_{Ma=0}}$$, is plotted on page 
2.0-30. The landing gear effectiveness factor, $$K_{GEAR}$$ 
is plotted on page 2.0-28.
2.0 \[ \Delta C_L_{\text{GROUND EFFECT}} \] = Change in basic lift coefficient due to ground effect.

\[ \Delta C_L_{\text{GROUND EFFECT}} = K_{GE} B \cdot \Delta C_L_{\text{GE}} \]

where \[ \Delta C_L_{\text{GE}} \] is plotted on page 2.0-32. The ground effect height factor, \[ K_{GE} \], is plotted on page 2.0-31.

\[ \Delta C_L_{\text{FLAP FAILURE}} \] = Change in basic lift coefficient due to flap extension or retraction from the flap position at which symmetric failure of both inboard or both outboard flaps occurs.

For symmetric inboard or outboard flap failure,

\[ \Delta C_L_{\text{FLAP FAILURE}} = \left[ (\Delta C_L)_{\alpha_{W.D.P.} = 0^\circ} \right]_{\text{FLAP FAILURE}} \]

\[ + \Delta \left( \frac{dC_L}{d\alpha} \right)_{\alpha_{W.D.P.} = 0^\circ} \]

where \[ (\Delta C_L)_{\alpha_{W.D.P.} = 0^\circ} \] is the change in basic lift coefficient at \( \alpha_{W.D.P.} = 0^\circ \) due to symmetric inboard or outboard flap failure. \[ \Delta \left( \frac{dC_L}{d\alpha} \right)_{\alpha_{W.D.P.} = 0^\circ} \] is plotted on page 2.0-31. \[ \Delta \left( \frac{dC_L}{d\alpha} \right)_{\text{FLAP FAILURE}} \cdot \alpha_{W.D.P.} \] is the change in basic lift coefficient due to the effect of symmetric inboard or outboard flap failure on the rigid airplane basic lift coefficient curve slope.

\[ \Delta \left( \frac{dC_L}{d\alpha} \right)_{\text{FLAP FAILURE}} \] is plotted on page 2.0-34.

The above data is also applicable for asymmetric (monitor limited) inboard or outboard flap failure, e.g., one inboard flap failed and the opposite inboard flap at the monitor limited extension or retraction position.
\( \Delta C_L \) FLAP FAILURE is to be added to total \( C_L \) computed for the inboard flap position. Note that inboard flap position should be used for all functions of flap in this document.

The angle of attack for stick shaker actuation is plotted on page 2.0-35. The angle of attack for initial buffet is plotted on page 2.0-36. Certification stall speeds are plotted on page 2.0-37. The initial buffet boundary and trimmed \( C_{L_{\text{MAX}}} \) values are plotted on page 2.0-38.
LOW SPEED

NOTE: USE FOR ALL ALTITUDES

\( (\Delta C_l)_{A_{W,D,P}} \) vs. \( \alpha \)

EQUIVALENT AIRSPEED, \( V_{eq} \) ~ KT.

CALC: LOW 11/18/67
CHECK: FOSTER 124-67
REVISED DATE: LOW 123-70

LIFT COEFFICIENT
EFFECT OF FLAPS ON \( (\Delta C_l)_{A_{W,D,P}} = 0^\circ \)

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PAGE 2.0-9
NOTE: USE FOR ALL ALTITUDES.

2. \( AC_L = A \left( \frac{dC_L}{d\alpha} \right) \cdot \text{W.D.R.} \)

\[ \Delta \left( \frac{dC_L}{d\alpha} \right) \text{ PER DEGREE} \]

\[ \text{EQUIVALENT AIRSPEED, } V_e \text{ ~ KT} \]

FLAPS

0°
15°
30°
45°
60°

LIFT COEFFICIENT
EFFECT OF FLAPS ON \( \Delta (dC_L/d\alpha) \)

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Vol. II

PAGE 2.0-11

REV. D
LIFT COEFFICIENT EFFECT OF $\frac{m}{\rho}$

THE BOEING COMPANY
LOW SPEED

NOTE: USE FOR ALL ALTITUDES.

\[ \Delta C_L = \frac{dC_L}{dn_z} \]  
\( n_z = 1 \) FOR STEADY LEVEL FLIGHT

\[ \frac{dC_L}{dn_z} \]

EQUIVALENT AIRSPEED, \( V_e \), 0 to 600

LIFT COEFFICIENT
EFFECT OF FLAPS ON \( \frac{dC_L}{dn_z} \)

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CALC \|
LOW \|
11/18/67 \|
REVISED \|
DATE
CHECK \|
FOSTER \|
1-24-68 \|
LOW \|
1-27-70
INK \|
ODEGARD \|
11/18/67

747
DG-30643
Vol. II
PAGE
2.0-15
REV. D
LIFT COEFFICIENT
EFFECT OF STABILIZER

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USE: CLDS = -.2678 CMDS
LIFT COEFFICIENT

EFFECT OF OUTBOARD ELEVATORS

THE BOEING COMPANY

USE: $\text{CL} = -0.2538 \text{ CMDE$}^\phi$
DATA SHOWN FOR INDIVIDUAL PANEL 4 OR 5
FOR PANEL 12 OR 1, MULTIPLY BY 0.40
PANELS 4 & 5 LIMITED TO 20 DEG MAX DEFORMATION

NOTE

TOTAL EFFECT OF SPOILER GROUP 3.0, 1, OR 2.5.
WITH HYDRAULIC SYSTEM NO. 2 OFF, MULTIPLY BY 0.40
WITH HYDRAULIC SYSTEM NO. 2 ON, MULTIPLY BY 0.20
FOR SPOILER GROUP 9.10 (OR 5.4), MULTIPLY BY 0.20

LIFT COEFFICIENT
EFFECT OF SPOILERS

THE BOEING COMPANY
NOTE: DATA SHOWN FOR BOTH PANELS OPERATING.

SPOILER PANELS 6 AND 7

PANELS LIMITED TO 20 DEG MAX DEFLECTION

LIFT COEFFICIENT
EFFECT OF SPOILERS (6 AND 7)

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SPOILER PANEL 12 OR 1

NOTE: USE FOR ALL FLAP SETTINGS

LIFT COEFFICIENT DUE TO SPOILERS (12 OR 1)

THE BOEING COMPANY

LESPD

REV B

DET-39413

PAGE 2.O-26
NOTE 1  FREE AIR

2  FOR LANDING GEAR FAILURE, REPLACE $\Delta C_{L \text{gear}}$ BY
$\Delta C_{L \text{gear failure}} = K_{\text{gear}} \cdot \Delta C_{L \text{gear}}$

<table>
<thead>
<tr>
<th>GEAR SELECTION</th>
<th>GEAR FAILURE</th>
<th>$K_{\text{gear}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>DOWN</td>
<td>WING GEARS FAIL TO EXTEND</td>
<td>0.1</td>
</tr>
<tr>
<td>UP</td>
<td>WING GEARS FAIL TO RETRACT</td>
<td>0.9</td>
</tr>
<tr>
<td>✓</td>
<td>ONE WING GEAR FAILS TO RETRACT</td>
<td>0.45</td>
</tr>
<tr>
<td>✓</td>
<td>ONE BODY GEAR FAILS TO RETRACT</td>
<td>0.4</td>
</tr>
<tr>
<td>✓</td>
<td>ONE BODY GEAR DOOR FAILS TO RETRACT</td>
<td>0.3</td>
</tr>
<tr>
<td>✓</td>
<td>NOSE GEAR FAILS TO RETRACT</td>
<td>0.0</td>
</tr>
</tbody>
</table>

$LIFT COEFFICIENT$

$\Delta C_{L \text{gear}}$ vs $V_{\text{WASP}}$ DEGREES

---

LIFT COEFFICIENT EFFECT OF LANDING GEAR

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NOE 1. GEAR ON GROUND
2. \( k_{GE} = 1.0 \)

NORMAL TAXI ATTITUDE

TAIL SKID AND EXTENDED GEAR TOUCHING

\( \Delta C_{L,GE} \)

\( \Delta \theta_{WBR} \) DEGREES

SEE SECTION 19 FOR REVISED DATA

LIFT COEFFICIENT GROUND EFFECT

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CALC  CURNUTT  3-24-67
CHECK  FOSTER  1-24-66  CURNUTT  3-5-70
APR
APR
INK  KINSMAN  3-5-70

Page 2.0-32
NOTE: DATA APPLICABLE FOR SYMMETRIC OR ASYMMETRIC (MONITOR LIMITED) FLAP FAILURE

\[
\frac{\Delta C_L}{\Delta \alpha} \text{ FLAP FAILURE PER DEG.}
\]

INBOARD FLAP POSITION, \( \delta_{F_i} = 0.1 \)

OUTBOARD FLAP POSITION, \( \delta_{F_o} \)

LIFT COEFFICIENT EFFECT OF FLAPS ON

\[
\left[ \frac{\Delta C_L}{\Delta \alpha} \right] \text{ FLAP FAILURE}
\]

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CALC  LOW  5-3-69  REVISED  DATE
CHECK LOW  2-17-70
APR LOW  6-25-70
INK ODEGARD  5-3-69

REVISED DATE

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DE-3064-11

PAGE 2.0-34
SEE SECTION 19
FOR REVISED DATA

ANGLE OF ATTACK
FOR INITIAL BUFFET

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CALC  LOW  1-19-68  REVISED  DATE
CHECK  FOSTER  1-24-68  LOW  6-4-69
ABR  LOW  1-29-70
APR
INK  ODEGARD  1-19-68
SEE SECTION 19
FOR REVISED DATA

ANGL OF ATTA FOR STICK SHAKER ACTUATION

THE BOEING COMPANY
NOTE
1. ENGINEs AT IDLE THRUST
2. LOAD FACTOR LESS THAN ONE
3. FORWARD C.G. LOCATION
4. GEAR UP EXCEPT FOR FLAPS 25 AND FLAPS 30
3.0 DRAG FORCE COEFFICIENT

The dimensionless aeroodynamic drag force coefficient is given in terms of its significant components by the equation below:

At a given $\alpha_{W.D.P}$,

$$C_D = K \cdot \left[ C_{D_{BASIC}} + \frac{dC_D}{d\alpha} \cdot \Delta F.R.L. \right] + \left[ 1 - K \right] [C_D]_M$$

$$+ \Delta C_{D_{SPOILERS}} + \Delta C_{D_{LANDING GEAR}} + \Delta C_{D_{GROUND EFFECT}}$$

$$+ \Delta C_{D_{SIDESLIP}} + \Delta C_{D_{RUDDERS}} + \left[ \Delta C_{D_{FLAP FAILURE}} \right]$$

where,

$C_{D_{BASIC}}$ = Basic drag coefficient for the rigid airplane at $\Delta F.R.L. = 0^\circ$, in free air and with the landing gear retracted. $C_{D_{BASIC}}$ is plotted on page 3.0-5.

$K$ = 0 for flaps up, $K = 1$ for flaps 1, 5, 10, 20, 25 and 30.

$\frac{dC_D}{d\alpha} \cdot \Delta F.R.L.$ = Change in basic drag coefficient due to change in stabilizer angle from $\Delta F.R.L. = 0^\circ$. $\frac{dC_D}{d\alpha}$ is plotted on page 3.0-6 and page 3.0-7.

$[C_D]_M$ = Drag coefficient at Mach number, $M$. $[C_D]_M$ is plotted on page 3.0-8 and 3.0-9. $C_L^*$ is given by the first air terms of the lift force coefficient equation on page 2.0-1.

NOT IN NASA SIMULATION
3.0 $\Delta C_{D_{\text{spoilers}}} = \text{Change in drag coefficient due to spoiler or speedbrake deflection.}$

(Cont'd)

$$\Delta C_{D_{\text{spoilers}}} = \Delta C_{D_{\text{SP}}} \left[ 1 + F_D \cdot K_{GE}^B \right]$$

$$+ 0.05 \cdot F_D \cdot K_{GE}^B \cdot \frac{1}{45 n} \cdot \sum_{\text{operating spoiler panels}} \delta_{\text{SP}}$$

where $\Delta C_{D_{\text{SP}}}$ is the change in drag coefficient due to spoiler or speedbrake deflection in free air.

$$\Delta C_{D_{\text{SP}}} = \sum_{\text{operating spoiler panels}} \left[ \left( \Delta C_{D_{\text{SP}}} \right)_{\alpha_{\text{W.D.P.}}=0} + \frac{d \left( \Delta C_{D_{\text{SP}}} \right)}{d\alpha} \left( \alpha_{\text{W.D.P.}}=4^\circ \right) \right] \frac{(C_{D_{\text{SP}}})_M}{(C_{D_{\text{SP}}})_{M=0}}$$

where $\left( \Delta C_{D_{\text{SP}}} \right)_{\alpha_{\text{W.D.P.}}=4^\circ}$ is the change in drag coefficient at $\alpha_{\text{W.D.P.}}=4^\circ$ due to deflecting the operating spoiler panels. $\left( \Delta C_{D_{\text{SP}}} \right)_{\alpha_{\text{W.D.P.}}=4^\circ}$ is plotted for spoilers on page 3.0-10 and is tabulated for ground spoilers on page 3.0-12. $\frac{d \left( \Delta C_{D_{\text{SP}}} \right)}{d\alpha}$ is the rate of change with angle of attack of drag coefficient due to deflecting the operating spoiler panels. $\frac{d \left( \Delta C_{D_{\text{SP}}} \right)}{d\alpha}$ is plotted for spoilers on page 3.0-11 and is tabulated for ground spoilers on page 3.0-12. The Mach number effect, $\frac{(C_{D_{\text{SP}}})_M}{(C_{D_{\text{SP}}})_{M=0}}$, is plotted on page 3.0-13. The ground effect lateral control factor, $F_D$, is plotted on page 3.0-14. The ground effect height factor, $K_{GE}^B$, is plotted on page 2.0-31. $n$ is the number of operating spoiler panels. $\delta_{\text{SP}}$ is the spoiler panel deflection.
3.6 angle (degrees).

(Cont'd) 

\[ \Delta C_{D_{\text{LANDING GEAR}}} = \text{Change in drag coefficient due to main and nose landing gear extension.} \]

\[ \Delta C_{D_{\text{LANDING GEAR}}} = K_{\text{GEAR}} \cdot \Delta C_{D_{\text{GEAR}}} \cdot \frac{(C_{D_{\text{GEAR}}})_M}{(C_{D_{\text{GEAR}}})_{M=0}} \]

where \( \Delta C_{D_{\text{GEAR}}} \) is plotted on page 3.0-15. The Mach number effect, \( \frac{(C_{D_{\text{GEAR}}})_M}{(C_{D_{\text{GEAR}}})_{M=0}} \), is plotted on page 3.0-16. The landing gear effectiveness factor, \( K_{\text{GEAR}} \), is plotted on page 2.0-28.

\[ \Delta C_{D_{\text{GROUND EFFECT}}} = \text{Change in drag coefficient due to ground effect.} \]

\[ \Delta C_{D_{\text{GROUND EFFECT}}} = K_{\text{GE}} \cdot \Delta C_{D_{\text{GE}}} \]

where \( \Delta C_{D_{\text{GE}}} \) is plotted on page 3.0-16. The ground effect height factor, \( K_{\text{GE}} \), is plotted on page 3.0-17.

\[ \Delta C_{D_{\text{SIDESLIP}}} = \text{Change in drag coefficient due to angle of sideslip, } \beta. \]

\( \Delta C_{D_{\text{SIDESLIP}}} \) is plotted on page 3.0-19.

\[ \Delta C_{D_{\text{RUDDERS}}} = \text{Change in drag coefficient due to rudder deflection.} \]

\[ \Delta C_{D_{\text{RUDDERS}}} = \Delta C_{D_{\text{RU}}} + \Delta C_{D_{\text{RL}}} \]

where \( \Delta C_{D_{\text{RU}}} \) and \( \Delta C_{D_{\text{RL}}} \) are the changes in drag coefficient due to deflection of the upper rudder and the lower rudder respectively. \( \Delta C_{D_{\text{RU}}} \) and \( \Delta C_{D_{\text{RL}}} \) are obtained from page 3.0-19.
3.0  \( \Delta C_{D_{\text{FLAP FAILURE}}} \) (Cont'd)

\( \Delta C_{D_{\text{FLAP FAILURE}}} = \left[ (\Delta C_D)_{\alpha_{\text{W.D.P.}} = 0^\circ} \right]_{\text{FLAP FAILURE}} + \Delta \left( \frac{dC_D}{d\alpha} \right)_{\text{FLAP FAILURE}} \cdot \alpha_{\text{W.D.P.}} \)

where \( \left[ (\Delta C_D)_{\alpha_{\text{W.D.P.}} = 0^\circ} \right]_{\text{FLAP FAILURE}} \) is the change in basic drag coefficient at \( \alpha_{\text{W.D.P.}} = 0^\circ \) due to symmetric inlet or outlet flow failure. \( \Delta \left( \frac{dC_D}{d\alpha} \right)_{\text{FLAP FAILURE}} \cdot \alpha_{\text{W.D.P.}} \) is the change in basic drag coefficient due to the effect of symmetric inlet or outlet flow failure on the airplane basic drag coefficient curve slope. \( \Delta \left( \frac{dC_D}{d\alpha} \right)_{\text{FLAP FAILURE}} \) is listed on page 3.0-11.

The above data is also available for asymmetric (monitor limited) inlet or outlet flap failure.

\( \Delta C_{D_{\text{FLAP FAILURE}}} \) is the change in total \( C_D \) at the inlet or outlet position.
EFFECT OF MACH NUMBER

DRAG COEFFICIENT

EFFECT OF MACH NUMBER

M ≤ .7

THE BOEING COMPANY
NOTE: DATA SHOWN FOR INDIVIDUAL PANELS A OR B.

1. FOR PANEL 12 OR 3, MULTIPLY BY 0.60.
2. PANELS A OR B LIMITED TO 20 DEG. MAX DEFLECTION.

NOTE: TOTAL EFFECT OF SPOILER GROUP 9, 10, 11 (OR 2, 3, 4) SHOWN.

1. WITH HYDRAULIC SYSTEM NO. 2 OFF, MULTIPLY BY 0.60.
2. WITH HYDRAULIC SYSTEM NO. 3 OFF, MULTIPLY BY 0.30.
3. FOR SPOILER GROUP 9, 10 (OR 2, 3, 4), MULTIPLY BY 0.60.
**SPOILER PANELS 6 AND 7**

NOTE: DATA SHOWN FOR BOTH PANELS OPERATING AT 20°.

<table>
<thead>
<tr>
<th>FLAP SETTING</th>
<th>((\Delta C_{\text{D}<em>{\text{SP}}})</em>{\alpha_{W,D}})</th>
<th>(\frac{d(\Delta C_{\text{D}_{\text{SP}}})}{d\alpha})</th>
</tr>
</thead>
<tbody>
<tr>
<td>UP</td>
<td>0.0008</td>
<td>-0.00013</td>
</tr>
<tr>
<td>1</td>
<td>0.0008</td>
<td>-0.00013</td>
</tr>
<tr>
<td>5</td>
<td>0.0020</td>
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<tr>
<td>30</td>
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<td>-0.00104</td>
</tr>
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</table>

---

**DRAG COEFFICIENT EFFECT OF SPOILERS (6 AND 7)**

THE BOEING COMPANY

---

**REV. C**
### Note 1: Free Air

2. For landing gear failure, replace $\Delta C_D$ by $K_{\text{gear}} \cdot \Delta C_D$.

<table>
<thead>
<tr>
<th>Gear Selection</th>
<th>Gear Failure</th>
<th>$K_{\text{gear}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Down</td>
<td>Wing gears fail to extend</td>
<td>0.4</td>
</tr>
<tr>
<td>Up</td>
<td>Wing gears fail to retract</td>
<td>0.6</td>
</tr>
<tr>
<td>✓</td>
<td>One wing gear fails to retract</td>
<td>0.3</td>
</tr>
<tr>
<td>✓</td>
<td>One body gear fails to retract</td>
<td>0.1</td>
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<tr>
<td>✓</td>
<td>One body gear door fails to retract</td>
<td>0.1</td>
</tr>
<tr>
<td>✓</td>
<td>Nose gear fails to retract</td>
<td>0.1</td>
</tr>
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</table>

### Drag Coefficient

Effect of Landing Gear

*The Boeing Company*

<table>
<thead>
<tr>
<th>CALC</th>
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<th>REVISIONS</th>
<th>DATE</th>
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<tr>
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<td>BYSTRÖM</td>
<td>2-18-70</td>
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</table>
DRAG COEFFICIENT EFFECT OF MACH NUMBER ON $\Delta C_{D,per}$

THE BOEING COMPANY

REVISED DATE

747

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PAGE

3.0-16
NOTE
1. GEAR ON GROUND
2. $K_{GE} = 1.0$

$\Delta C_{D,GE}$ vs. W.D.P. ~ DEGREES

NORMAL TAXI ATTITUDE

TAIL SKID AND EXTENDED GEAR TOUCHING

FLAPS

SEE SECTION 19 FOR REVISED DATA

DRAG COEFFICIENT
GROUND EFFECT

THE BOEING COMPANY
NOTE:
1. \( \Delta C_{D_{F-L}} = (0.4) \Delta C_{D_{F-T}} \)
2. \( \Delta C_{D_{R-L}} = (0.6) \Delta C_{D_{R-T}} \)
3. Reverse sign on \( \delta \) for negative rudder deflection
4. Use for all angles of attack
5. Use for all flap positions

---

DRAG COEFFICIENT
EFFECT OF SIDESLIP
AND RUDDER

THE BOEING COMPANY

CALC: STIRLING 10/21/67
CHECK: HOLTZENER 12/1/67
APR
APR
INK: ODEGARD 10/21/67

REV E

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DE-30643
Vol. II
PAGE 3.0-19
NOTE: DATA APPLICABLE FOR SYMMETRIC OR ASYMMETRIC (MONITOR LIMITED) FLAP FAILURE.

THESE DATA NOT INCLUDED IN NASA SIMULATION.

[Graph showing effect of flaps on drag coefficient with flap failure.

<table>
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<td>APR</td>
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<td>6.25.70</td>
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<tr>
<td>INK</td>
<td>ODEGARD</td>
<td>5349</td>
<td></td>
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</tr>
</tbody>
</table>

DRAG COEFFICIENT EFFECT OF FLAPS ON

\[ (\Delta C_D)_{\text{W.D.R.} = 0^\circ} \] FLAP FAILURE

THE BOEING COMPANY
**NOTE**

DATA APPLICABLE FOR SYMMETRIC OR

**ASYMMETRIC (MONITOR LIMITED) FLAP FAILURE**

These data not included in NASA simulation.

$$\left[ \frac{\Delta (dC_D)}{dx} \right]_{FLAP\ FAILURE}$$

**INBOARD FLAP POSITION**

$$\delta_f = 5$$

**OUTBOARD FLAP POSITION, $$\delta_f$$**

<table>
<thead>
<tr>
<th>CALC</th>
<th>LOW 5.369</th>
<th>REVISED</th>
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<tr>
<td>APR</td>
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<tr>
<td>APR</td>
<td>ODEGARD 5.369</td>
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</tbody>
</table>

**THE BOEING COMPANY**

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Page 3.0-21
4.0 PITCHING MOMENT COEFFICIENT

The dimensionless aerodynamic pitching moment coefficient is given in terms of its significant components by the equation below.

At a given $\alpha_{\text{W.D.P.}}$,

$$C_{m_{\text{C.G.}}} = C_{m_{25\text{BASIC}}} + (\Delta C_{m_{25}})_{\alpha_{\text{W.D.P.}}=0^\circ} + \Delta \left( \frac{dC_{m_{25}}}{d\alpha} \right) \cdot \alpha_{\text{W.D.P.}}$$

$$+ C_L \left( \text{C.G.-25} \right) \cdot \frac{dC_{m_{25}}}{d\alpha} \cdot \left( \frac{c}{2V} \right)$$

$$+ \frac{dC_{m_{25}}}{d\alpha} \cdot \left( \frac{q}{2V} \right) + \frac{dC_{m_{25}}}{dN_2} \cdot n_z$$

$$+ K_{\alpha} \cdot \frac{dC_{m_{25}}}{d\alpha} \cdot \delta_{\text{FLAP}} + K_{\alpha} \cdot \frac{dC_{m_{25}}}{d\delta_{\text{EL.}}} \cdot \delta_{\text{EL.}} + K_{\alpha} \cdot \frac{dC_{m_{25}}}{d\delta_{\text{E.}}} \cdot \delta_{\text{E.}}$$

$$+ \Delta C_{m_{25\text{SPOILERS}}} + \Delta C_{m_{25\text{INBOARD AILERS}}} + \Delta C_{m_{25\text{OUTBOARD AILERS}}} + \Delta C_{m_{25\text{LANDING GEAR}}}$$

$$+ \Delta C_{m_{25\text{GROUNDEFF}}} + \Delta C_{m_{25\text{SIDESLIP}}} + \Delta C_{m_{25\text{RUDDER}}} \left[ +\Delta C_{m_{25\text{FLAP FAILURE}}} \right]$$

where,

$$C_{m_{25\text{BASIC}}} = \text{Basic pitching moment coefficient for the rigid airplane at } \alpha_{\text{F.R.L.}} = 0^\circ \text{ in free air, with the landing gear retracted, and with the C.G. = 25% M.A.C.}$$

For low speed, $C_{m_{25\text{BASIC}}}$ is plotted on pages 4.0-8 and 4.0-9. For flaps up, $C_{m_{25\text{BASIC}}}$ is plotted on page 4.0-10.

$$\left( \Delta C_{m_{25}} \right)_{\alpha_{\text{W.D.P.}}=0^\circ} = \text{Change in basic pitching moment coefficient at } \alpha_{\text{W.D.P.}} = 0^\circ \text{ due to aeroelasticity. For low speed,}$$

[ ]* NOT IN NASA SIMULATION
For flaps up, \( (\Delta C_{m_{25}})_{w.a.r.} = 0^\circ \) is plotted on page 4.0-11.

\( \Delta \left( \frac{dC_{m_{25}}}{d\alpha} \right) \) is change in basic pitching moment coefficient due to the aeroelastic effect on the rigid airplane basic pitching moment coefficient curve slope. For low speed, \( \Delta \left( \frac{dC_{m_{25}}}{d\alpha} \right) \) is plotted on page 4.0-13.

For flaps up, \( \Delta \left( \frac{dC_{m_{25}}}{d\alpha} \right) \) is plotted on page 4.0-14.

\( C_{L}(C.G.-25) = \) Change in pitching moment coefficient due to center of gravity variation from 25% M.A.C. The total lift coefficient, \( C_L \), is defined in Section 2.0.

\( \frac{dc_{m}}{d\alpha} \) = Change is basic pitching moment coefficient due to rate of change of angle of attack.

\[ \frac{dc_{m}}{d\alpha} = k_{\alpha} \cdot \frac{dc_{m_{25}}}{d\alpha} \]

where \( \frac{dc_{m_{25}}}{d\alpha} \) and the center of gravity factor, \( k_{\alpha} \) are plotted on page 4.0-15.

\( \frac{dc_{m_{25}}}{d\dot{\alpha}} \) = Change in basic pitching moment coefficient due to pitch rate. \( \frac{dc_{m_{25}}}{d\dot{\alpha}} \) is plotted on page 4.0-16.

\( \frac{dc_{m_{25}}}{dn_{E}} \) = Change in basic pitching moment coefficient due to aeroelastic inertia relief caused by normal load factor, \( n_{E} \). For low speed, \( \frac{dc_{m_{25}}}{dn_{E}} \) is plotted on page 4.0-17. For flaps up, \( \frac{dc_{m_{25}}}{dn_{E}} \) is plotted...
4.0 on page 4.0-18.

(Cont'd)

$K_{\alpha} \cdot \frac{dC_{m_{25}}}{d\alpha} \cdot A_{FRL} =$ Change in basic pitching moment coefficient due to change in stabilizer angle from $\alpha_{FRL} = 0^\circ$.

$\frac{dC_{m_{25}}}{d\alpha}$ is plotted on page 4.0-20. The effectiveness factor for the stabilizer (and elevators), $K_{\alpha}$, is plotted on page 4.0-19.

$K_{\alpha} \cdot \frac{dC_{m_{25}}}{d\delta_{EI}} \cdot \delta_{EI} =$ Change is basic pitching moment coefficient due to change in inboard elevator angle from $\delta_{EI} = 0^\circ$.

$\frac{dC_{m_{25}}}{d\delta_{EI}}$ is plotted on page 4.0-21.

$K_{\alpha} \cdot \frac{dC_{m_{25}}}{d\delta_{EO}} \cdot \delta_{EO} =$ Change in basic pitching moment coefficient due to change in outboard elevator angle from $\delta_{EO} = 0^\circ$.

$\frac{dC_{m_{25}}}{d\delta_{EO}}$ is plotted on page 4.0-22.

The normal system stick free (rigged) elevator deflection is $+2^\circ$ from the faired position.

$\Delta C_{m_{25}}_{spoilers} =$ Change in basic pitching moment coefficient due to spoiler or speed brake deflection.

$$\Delta C_{m_{25}}_{spoilers} = \sum (K_{SP} \cdot (\Delta C_{m_{25}}_{SP})_{45} \cdot \frac{(C_{m_{25}}_{SP})_{SP}}{(C_{m_{25}}_{SP})_{MP}} \cdot \frac{(M_E)}{(M_{RP})_{SP}} \cdot F_{mg} \cdot GE$$

where $(\Delta C_{m_{25}}_{SP})_{45}$ is the change in basic pitching moment coefficient due to deflecting the operating spoiler panels to $45^\circ$. $(\Delta C_{m_{25}}_{SP})_{45}$ is plotted for spoilers and ground spoilers on page 4.0-24 and...
4.0-25 respectively. The spoiler effectiveness factor, \((K_{sSP})_m\) is plotted on page 4.0-23. The Mach number effect, \((C_{m.25SP})_M/(C_{m.25SP})_{M=0}\) is plotted on page 4.0-26. The aeroelastic effect, \((M_e)/(M_{SP})_s\) is plotted on pages 4.0-27, 4.0-28, and 4.0-29. The ground effect factor, \(F_{mGE}\) is obtained as follows:

\[
F_{mGE} = [1 + (F_m \cdot K_{GE})]
\]

where \(F_m\) is plotted on page 4.0-32. The ground effect height factor, \(K_{GE}\) is plotted on page 2.0-31.

\[
\Delta C_{m.25}^{INBOARD AILERONS} = \text{Change in basic pitching moment coefficient due to inboard aileron deflection.}
\]

\[
\Delta C_{m.25}^{INBOARD AILERONS} = K_{SAI} \cdot (\Delta C_{m.25}^{AI})_{20} \cdot \left(\frac{C_{m.25}^{AI}}{C_{m.25}^{AI}_{M=0}}\right) \cdot F_{mGE}
\]

where \((\Delta C_{m.25}^{AI})_{20}\) is the change in basic pitching moment coefficient due to deflecting one inboard aileron up to 20°. \((\Delta C_{m.25}^{AI})_{20}\) is plotted on page 4.0-30. The inboard aileron effectiveness factor, \(K_{SAI}\) is to be obtained for the up inboard aileron deflection and is plotted on page 5.0-22. The Mach number effect, \((C_{m.25}^{AI})_M/(C_{m.25}^{AI})_{M=0}\) is plotted on page 4.0-30. The ground effect factor, \(F_{mGE}\) is obtained from page 4.0-32.

\[
\Delta C_{m.25}^{OUTBOARD AILERONS} = \text{Change in basic pitching moment coefficient due to}
\]
4.0

outboard aileron deflection.

\[ \Delta C_{m,25,\text{OUTBOARD AILERONS}} = \sum_{\text{UP AND DOWN OUTBOARD AILERONS}} K_{\delta_{AO}} \cdot \Delta C_{m,25A_O} \cdot F_{mGE} \]

where \( \Delta C_{m,25A_O} \) is the change in basic pitching moment coefficient due to deflecting one outboard aileron up to 25° or the opposite outboard aileron down to 15°. \( \Delta C_{m,25A_O} \) is plotted on page 4.0-31.

The outboard aileron effectiveness factor, \( K_{\delta_{AO}} \), is plotted on page 5.0-26. The ground effect factor, \( F_{mGE} \), is obtained from page 4.0-32.

\[ \Delta C_{m,25,\text{LANDING GEAR}} \]

= Change is basic pitching moment coefficient due to main and nose landing gear extension.

\[ \Delta C_{m,25,\text{LANDING GEAR}} = K_{\text{GEAR}} \cdot \Delta C_{m,25,\text{GEAR}} \cdot \frac{(C_{m,25,\text{GEAR}})_{M=0}}{(C_{m,25,\text{GEAR}})_{M=\infty}} \]

where \( \Delta C_{m,25,\text{GEAR}} \) is plotted on page 4.0-35.

The Mach number effect, \( \frac{(C_{m,25,\text{GEAR}})_{M=0}}{(C_{m,25,\text{GEAR}})_{M=\infty}} \), is plotted on page 4.0-34. The landing gear effectiveness factor, \( K_{\text{GEAR}} \), is plotted on page 2.0-28.

\[ \Delta C_{m,25,\text{GROUND EFFECT}} \]

= Change in basic pitching moment coefficient due to ground effect.

\[ \Delta C_{m,25,\text{GROUND EFFECT}} = k_{GE} \cdot \Delta C_{m,25,\text{GE}} \]

where \( \Delta C_{m,25,\text{GE}} \) is plotted on page 4.0-35.
The ground effect weight factor, $K_{GE}$, is plotted on page 6.0-31.

$$\Delta C_{m_{25\text{SIDESLIP}}} = \text{Change in basic pitching moment coefficient due to angle of sideslip} \beta. \quad \Delta C_{m_{25\text{SIDESLIP}}}$$ is plotted on page 6.0-36.

$$\Delta C_{m_{25\text{RUDDERS}}} = \Delta C_{m_{25\text{RU}}} + \Delta C_{m_{25\text{RL}}}$$

$\Delta C_{m_{25\text{RU}}}$ and $\Delta C_{m_{25\text{RL}}}$ are the changes in pitching moment coefficient due to deflection of upper and lower rudders respectively.

$\Delta C_{m_{25\text{RU}}}$ and $\Delta C_{m_{25\text{RL}}}$ are obtained from page 4.0-36.

$\Delta C_{m_{25\text{FLAP FAILURE}}} = \text{Change in basic pitching moment coefficient due to flap deflection or removal from the flap position at which stall failure occurs. Flap deflection is either inward or both outboard flaps inward.}$

$\Delta C_{m_{25\text{FLAP FAILURE}}} = \left[(\Delta C_{m_{25}})_{\alpha_{\text{W.D.P.}}=0^\circ}\right]_{\text{FLAP FAILURE}}$

$$+ \Delta \left(\frac{dC_{m_{25}}}{d\alpha}\right)_{\text{FLAP FAILURE}} \cdot \alpha_{\text{W.D.P.}}$$

where $\left[(\Delta C_{m_{25}})_{\alpha_{\text{W.D.P.}}=0^\circ}\right]_{\text{FLAP FAILURE}}$ is the change in basic pitching moment coefficient at $\alpha_{\text{W.D.P.}} = 0^\circ$. 
due to symmetric inboard or outboard flap failure.

\[ \left( \Delta C_{m,25} \right)_{\alpha_{W.D.P.} = 0^\circ} \] FLAP FAILURE is plotted on page 4.0-31. \( \Delta \left( \frac{dC_{m,25}}{d\alpha} \right) \) FLAP FAILURE is the change in basic pitching moment coefficient due to the effect of symmetric inboard or outboard flap failure on the rigid airplane basic pitching moment coefficient curve slope.

\( \Delta \left( \frac{dC_{m,25}}{d\alpha} \right) \) FLAP FAILURE is plotted on page 4.0-38. The above data is also applicable for asymmetric (monitor limited) inboard or outboard flap failure.

\( \Delta C_{m,25} \) FLAP FAILURE is to be added to total \( C_{m_{c.q.}} \) computed for the inboard flap position.
NOTE: NO THRUST EFFECT

\[ \alpha_{flbr} = \theta - \delta_{f} = \delta_{f} \cdot 0.1 \]

GEAR UP, FREE AIR

\[ C_{m,2b basics} \]

LOW SPEED

PITCHING MOMENT COEFFICIENT EFFECT OF ANGLE OF ATTACK ON BASIC \( C_{m,2b} \) (FLAPS UP, 1, 5, 10)

THE BOEING COMPANY

CALC   CHECK   APR   APR   INK
LOW    LOW    LOW    BYSTRA  KINSMA
11/21/67  1.24/68  2/1/68  2.9/70  3.2/70
NOTE
NO THRUST EFFECT

1. \( \Delta \beta_{RL} + \theta^* \), \( \beta_{I} = \beta_{EO} + \theta^* \)
2. GEAR UP, FREE AIR

\( \alpha_{WDR} = \text{DEGREES} \)

C_{m_{25}}^BASIC

LOW SPEED

CALC  | LOW  | 11/21/67  | REVISED  | DATE  | 747
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INK   | KINSMAN | 3-2-70   |        |        | THE BOEING COMPANY
LOW SPEED

NOTE: USE FOR ALL ALTITUDES

\[ \Delta C_{m_{25}} = \Delta \left( \frac{dC_{m_{25}}}{d\alpha} \right) \times \alpha_{MSP} \]

\[ \Delta \left( \frac{dC_{m_{25}}}{d\alpha} \right) \]

PER DEG.

EQUIVALENT AIRSPEED, \( V_2 \) IN KNOTS

0 50 100 150 200 250

10, 20, 25, 30

PITCHING MOMENT COEFFICIENT EFFECT OF FLAPS ON \( \Delta (dC_{m_{25}}/d\alpha) \)

THE BOEING COMPANY
LOW SPEED

NOTE: USE FOR ALL ALTITUDES

\[ \Delta C_{m_{25}} = \frac{dC_{m_{25}}}{dn_z} \cdot n_z \quad (n_z = 1 \text{ FOR STEADY LEVEL FLIGHT}) \]

\[ dC_{m_{25}} \]
\[ \frac{dn_z}{dn_z} \]

FLAPS
30
20, 25
15, 10

EQUIVALENT AIRSPEED, \( V_e \) - KNOTS

0 50 100 150 200 250

PITCHING MOMENT COEFFICIENT EFFECT OF FLAPS ON \( dC_{m_{25}} / dn_z \)
AEROELASTIC EFFECT ON K.E AL.ANLWC l*6L Q..YIU REV. O AD 461 C-66

NOTE
1 USE FOR ALL FLAP SETTINGS
2 AC, m, K., dCm, dA, dK. dA

PITCHING MOMENT COEFFICIENT
AEROELASTIC EFFECT ON
STABILIZER EFFECTIVENESS
THE BOEING COMPANY

CALC LOW 10-11-67 REVISED DATE
CHECK FOSTER 1-24-68 BRYANT 8-17-69
APR
APR
INK ODEGARD 10-11-67

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dc

REV. 06-30643
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AD 461 C-66

THE BOEING COMPANY
NOTE

1. USE FOR ALL FLAP SETTINGS

2. BOTH OUTBOARD ELEVATORS DEFLECTED

3. FOR ONE OUTBOARD ELEVATOR DEFLECTED, USE HALF THE VALUE SHOWN

4. \[ \frac{\Delta C_{m_{25}}}{\Delta S_{E_0}} = K_e \cdot \frac{dC_{m_{25}}}{dS_{E_0}} \]
NOTE: LINK FOR ALL SPOILER PANELS.

Panel 5, 7, and 8 limited to 20° max deflection.

\[ (K_{c_{ip}})_{m} \]

\[ \theta_{m} \text{ DEGREES} \]

PITCHING MOMENT COEFFICIENT EFFECTIVENESS FACTOR SPOILERS

KUPCIS 12/14/67 REVISED 2/14/70 THE BOEING COMPANY

CHECK

FOSTER 1/24/69 KUPCIS 6/2/69
APR
KUPCIS 8/27/69
APR
INK ODEGARD 12/14/67

LOW 2/14/70

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REV. D
NOTE

1. DATA SHOWN FOR INDIVIDUAL PANELS 6 OR 5
2. FOR PANEL 12 OR 13 REVERSE SIGN
3. PANELS 6 OR 5 LIMITER TO 20 DEG MAX DEFLECTION

SPOILER PANEL GROUP 9, 10, 11 OR 1, 2, 3, 4

NOTE

1. TOTAL EFFECT OF SPOILER GROUP 9, 10, 11 (OR 1, 2, 3, 4) SHOWN
2. FOR SPOILER GROUP 9, 10, 11 (OR 1, 2, 3, 4), MULTIPLY BY 0.7
3. WITH HYDRAULIC SYSTEM NO 2 OFF, MULTIPLY BY 0.35
4. WITH HYDRAULIC SYSTEM NO 3 OFF, MULTIPLY BY 0.47

PITCHING MOMENT COEFFICIENT
EFFECT OF SPOILERS

THE BOEING COMPANY

K-5 ALBANESE 7/91
REV D

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NOTE: DATA SHOWN FOR BOTH PANELS OPERATING.

PANELS LIMITED TO 30° MAX DEVIATION.
SPOILER PANEL GROUP 9,10,11 OR 2,3,4.

NOTE: USE FOR ALL FLAP SETTINGS

---

[Graph showing pitching moment coefficient due to spoilers as a function of Mach number.]

---

THE BOEING COMPANY

MESSB

REVD. B

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MACH NUMBER, M

0.0 1.0 2.0 3.0 4.0 5.0 6.0 7.0 8.0 9.0 10.0

1.0

0.5

0.0

-0.5

-1.0

-1.5

-2.0
PITCHING MOMENT COEFFICIENT
AERODYNAMIC EFFECT ON PITCHING MOMENT
COEFFICIENT DUE TO SPOILERS (120 or 1)

THE BOEING COMPANY

REV. B
Moto: 1 \((C_{M_{x_{a}}})_{M_{w}}\) is shown for up aileron only
2 \((C_{M_{x_{a}}})_{M_{w}}\) for down aileron is zero

\(C_{M_{x_{a}}}\) vs.

\(M_{w}\)

Pitching Moment Coefficient
Effect of Inboard Aileron

The Boeing Company
PITCHING MOMENT COEFFICIENT EFFECT OF OUTBOARDAILERON

THE BOEING COMPANY

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PAGE 4.0-31

REV. D
NOTE: \[ F_{mg} = \frac{1}{1 - F_m \cdot K_{og}^2} \]

WHERE \( K_{og}^2 \) is shown on page 20-51.
LOW SPEED

NOTE

1. FREE AIR

2. FOR LANDING GEAR FAILURE, REPLACE $\Delta C_{m, \text{GEAR}}$ BY $\Delta C_{m, \text{GEAR FAILURE}} = K_{\text{m,GEAR}} \cdot \Delta C_{m, \text{GEAR}}$

| GEAR SELECTION | GEAR FAILURE | $K_{\text{m,GEAR}}$
<table>
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<tr>
<td>DOWN</td>
<td>WING GEARS FAIL TO EXTEND</td>
<td>0.6</td>
</tr>
<tr>
<td>UP</td>
<td>WING GEARS FAIL TO RETRACT</td>
<td>0.4</td>
</tr>
<tr>
<td>✓</td>
<td>ONE WING GEAR FAILS TO RETRACT</td>
<td>0.2</td>
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<td>ONE BODY GEAR FAILS TO RETRACT</td>
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<tr>
<td>✓</td>
<td>ONE BODY GEAR DOOR FAILS TO RETRACT</td>
<td>-0.4</td>
</tr>
<tr>
<td>✓</td>
<td>NOSE GEAR FAILS TO RETRACT</td>
<td>0.35</td>
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$\Delta C_{m, \text{GEAR}}$

PITCHING MOMENT COEFFICIENT EFFECT OF LANDING GEAR

THE BOEING COMPANY

CALC LOW 9-20-67 REVISED DATE 1-24-68 LOW 6-9-69
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AC 40-1 C-44

K-E ALBANY, NY 12203-2205

REV. D

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FOSTER 3.6.70
CRNUTT 3.6.70
CHECK

NOTE

GEAR ON GROUND

GROUND EFFECT

THE BOEING COMPANY

PITCHING MOMENT COEFFICIENT

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SEE SECTION 19
FOR REVISED DATA

KINGMAN VOL. 3094
4.0.36-35
PITCHING MOMENT COEFFICIENT
EFFECT OF SIDESLIP
AND RUDDER
THE BOEING COMPANY

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REV. D
NOTE DATA APPLICABLE FOR SYMMETRIC OR
ASYMMETRIC (MONITOR LIMITED) FLAP FAILURE

CLOSED DATA NOT INCLUDED
IN HIGH SIMULATION

\[
\frac{\Delta \left( \frac{dC_{m_{25}}}{d\alpha} \right)}{\text{PER DEG.}} = \text{FLAP FAILURE}
\]

INBOARD FLAP POSITION, \( \delta_{in} \)

OUTBOARD FLAP POSITION, \( \delta_{out} \)

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PITCHING MOMENT COEFFICIENT
EFFECT OF FLAPS ON

\[
\Delta \left( \frac{dC_{m_{25}}}{d\alpha} \right) = \text{FLAP FAILURE}
\]

THE BOEING COMPANY
ROLLING MOMENT COEFFICIENT

The dimensionless aerodynamic rolling moment coefficient is given in terms of its significant components by the equation below.

At a given $\alpha_{N.D.P.}$,

$$C_1 = \frac{dC_1}{d\beta} \cdot \beta + \frac{dC_1}{d\beta} \cdot \frac{P_{sb}}{zV} + \frac{dC_1}{d\beta} \cdot \frac{r_{sb}}{zV} + \Delta C_1_{\text{SPOILERS}} + \Delta C_1_{\text{INBOARD AILERONS}} + \Delta C_1_{\text{OUTBOARD AILERONS}} + \Delta C_1_{\text{RUDDERS}} + \left[ \Delta C_1_{\text{FLAP FAILURE}} + \Delta C_1_{\text{L.E. FAILURE}} \right]^*$$

where,

$$\frac{dC_1}{d\beta}$$

is the basic rate of change of rolling moment coefficient due to angle of sideslip, $\beta$.

The complete expression for $\frac{dC_1}{d\beta}$ is given as follows:

$$\frac{dC_1}{d\beta} = \left[ \frac{dC_1}{d\beta} \cdot \left( \frac{C_{1A}}{C_{1A}} \right) - \frac{C_{1A}}{C_{1A}} \right] + K_{\text{GEAR}} \cdot \Delta \left( \frac{dC_1}{d\beta} \right)_{\text{LANDING GEAR}}$$

where $\left( \frac{dC_1}{d\beta} \right)$ is the basic rate of change of rolling moment coefficient due to angle of sideslip. For low speed, $\left( \frac{dC_1}{d\beta} \right)$ is plotted on page 5.0-7. For flaps up, $\left( \frac{dC_1}{d\beta} \right)$ is plotted on page 5.0-8. $\frac{C_{1A}}{C_{1A}}$ is plotted on page 5.0-9.

[ ]* NOT IN NASA SIMULATION
The aeroelastic effect, \( \frac{C_{10}}{C_{100}} \), is plotted on page 5.0-10. The effect of main and nose gear extension, \( \Delta \left( \frac{dC_A}{d\beta} \right) \), is given by:

\[
\Delta \left( \frac{dC_A}{d\beta} \right)_{\text{LANDING GEAR}} = -.0003 \text{ PER DEGREE}
\]

The landing gear effectiveness factor, \( K_{GEAR} \), is plotted on page 2.0-28. The ground effect factor, \( F_{PGE} \), is obtained as follows:

\[
F_{PGE} = \left[ 1 + F_{P} \cdot K_{GE} \right]
\]

where the ground effect sideslip factor, \( F_{P} \), is plotted on page 5.0-11. The ground effect height factor, \( K_{GE} \), is plotted on page 2.0-31.

\[
\frac{dC_A}{d\beta} \cdot \frac{r_{sb}}{2V} = \text{Rolling moment coefficient due to roll rate about the stability axis, } x_s.
\]

The complete expression for \( \frac{dC_A}{d\beta} \) is given as follows:

\[
\frac{dC_A}{d\beta} = \left( \frac{dC_A}{d\beta} \right) \cdot \frac{(C_{2\beta})_M}{(C_{2\beta})_{M=0}}
\]

where \( \left( \frac{dC_A}{d\beta} \right) \) is plotted on page 5.0-12. The aeroelastic effect, \( \frac{(C_{2\beta})_M}{(C_{2\beta})_{M=0}} \), is plotted on page 5.0-13.

\[
\frac{dC_A}{df} \cdot \frac{r_{sb}}{2V} = \text{Rolling moment coefficient due to yaw rate about the stability axis, } z_s.
\]

The complete expression for \( \frac{dC_A}{df} \) is given as follows:
5.0

(Cont'd)

\[
\frac{dC_\alpha}{d\alpha} = \left(\frac{dC_\alpha}{d\alpha}\right)_{M=\alpha} \cdot \frac{(C_{\alpha \rho})_{M}}{(C_{\alpha \rho})_{M=0}}
\]

where \(\left(\frac{dC_\alpha}{d\alpha}\right)\) is plotted on page 5.0-14. The aeroelastic effect, \(\frac{(C_{\alpha \rho})_{M}}{(C_{\alpha \rho})_{M=0}}\), is plotted on page 5.0-15.

\[\Delta C_{\alpha \text{spoilers}}\]

Rolling moment coefficient due to spoiler deflection.

\[
\Delta C_{\alpha \text{spoilers}} = \sum_{\text{OPERATING \ Spoiler \ PANELS}} (K_{sp})_{\alpha} \cdot (\Delta C_{\alpha \text{spoilers}})_{45} \cdot \frac{(C_{\alpha \text{sp}})_{M}}{(C_{\alpha \text{sp}})_{M=0}} \cdot \frac{(R_E)}{(R_R)} \cdot F_{e GE}
\]

where \((\Delta C_{\alpha \text{sp}})_{45}\) is the rolling moment coefficient due to deflecting the operating spoiler panels to 45°. \((\Delta C_{\alpha \text{sp}})_{45}\) is plotted on page 5.0-17. The spoiler effectiveness factor, \((K_{sp})_{\alpha}\), is plotted on page 5.0-16. The Mach number effect, \(\frac{(C_{\alpha \text{sp}})_{M}}{(C_{\alpha \text{sp}})_{M=0}}\), is plotted on page 5.0-18. The aeroelastic effect, \(\frac{(R_E)}{(R_R/\text{sp})}\), is plotted on pages 5.0-19, 5.0-20, and 5.0-21. The ground effect factor, \(F_{e GE}\), is obtained as follows:

\[
F_{e GE} = \left[ 1 + F_L \cdot K_{e \text{GE}} \right]
\]

where the ground effect lateral control factor, \(F_L\), is plotted on page 5.0-29. The ground effect height factor, \(K_{e \text{GE}}\), is plotted on page 2.0-31.

\[\Delta C_{\alpha \text{inboard \ ailerons}}\]

Rolling moment coefficient due to inboard aileron deflection.
5.0

(Cont'd)

\[ \Delta C_{IAI} = \sum K_{SIA} \cdot (\Delta C_{LAI})_{20} \cdot \left( \frac{C_{LAI}}{C_{LAI}} \right)_{M=O} \cdot (R_e) \cdot F_{GE} \]

where \((\Delta C_{LAI})_{20}\) is the rolling moment coefficient due to deflecting one inboard aileron up to 20° or the opposite inboard aileron down to 20°. \((\Delta C_{LAI})_{20}\) is plotted on page 5.0-23. The inboard aileron effectiveness factor, \(K_{SIA}\), is plotted on page 5.0-22.

The Mach number effect, \(M_{\ldots} (\ldots 4x_{\ldots 4} M_{\ldots})\), is plotted on page 5.0-24. The aeroelastic effect, \(R_{L} \cdot C_{LAI}\), is plotted on page 5.0-25. The ground effect factor, \(F_{GE}\), is obtained from page 5.0-29.

\[ \Delta C_{IAO} = \sum K_{SIO} \cdot \Delta C_{IAO} \cdot (R_e) \cdot F_{GE} \]

where \(\Delta C_{IAO}\) is the rolling moment coefficient due to deflecting one outboard aileron up to 25° or the opposite outboard aileron down to 15°. \(\Delta C_{IAO}\) is plotted on page 5.0-27. The outboard aileron effectiveness factor, \(K_{SIO}\), is plotted on page 5.0-26. The aeroelastic effect, \(\left( \frac{R_e}{R_{R}} \right) \cdot C_{IAO}\), is plotted on page 5.0-28. The ground effect factor, \(F_{GE}\), is obtained from page 5.0-29.
5.0 $\Delta C_{\text{RUDDERS}}$ = Rolling moment coefficient due to rudder deflection.

(Cont'd)

$$\Delta C_{\text{RUDDERS}} = K_{\text{RU}} \cdot \Delta C_{\text{RU}}^{M=0} \cdot \frac{(C_{\text{RU}})_M}{(C_{\text{RU}})_M=0} + K_{\text{RL}} \cdot \Delta C_{\text{RL}}^{M=0} \cdot \frac{(C_{\text{RL}})_M}{(C_{\text{RL}})_M=0}$$

where $(\Delta C_{\text{RU}})_M$ and $(\Delta C_{\text{RL}})_M$ are the rolling moment coefficients due to full deflection of the upper rudder and the lower rudder respectively.

$(\Delta C_{\text{RU}})_M$ and $(\Delta C_{\text{RL}})_M$ are plotted on page 5.0-30. The upper rudder effectiveness factor, $K_{\text{RU}}$, and the lower rudder effectiveness factor, $K_{\text{RL}}$, are plotted on pages 6.0-17 and 6.0-18 respectively. The aeroelastic effects, $\frac{(C_{\text{RU}})_M}{(C_{\text{RU}})_M=0}$ and $\frac{(C_{\text{RL}})_M}{(C_{\text{RL}})_M=0}$, for the upper rudder and the lower rudder, are plotted on page 5.0-31 and page 5.0-32 respectively.

$\Delta C_{\text{FLAP FAILURE}}$ = Rolling moment coefficient due to an asymmetric (monitor limited) inboard or outboard flap failure.

$$\Delta C_{\text{FLAP FAILURE}} = \Delta C_{\text{iNBD FAILURE}} \text{ OR } \Delta C_{\text{OUTBD FAILURE}}$$

where $\Delta C_{\text{iNBD FAILURE}}$ is the rolling moment coefficient due to an asymmetric (monitor limited) inboard flap failure. $\Delta C_{\text{iNBD FAILURE}}$ for flap extension or retraction is plotted on pages 5.0-33 and 5.0-34 respectively.

$\Delta C_{\text{OUTBD FAILURE}}$ for flap extension or retraction is plotted on pages 5.0-35 and 5.0-36 respectively.
5.0 \( \Delta C_{l.e. \, failure} \) Rolling moment coefficient due to asymmetric leading
(Cont'd) edge flap failure. \( \Delta C_{l.e. \, failure} \) is plotted on
pages 5.0-37 and 5.0-38.
LOW SPEED

NOTE 1 GEAR UP, FREE AIR

ROLLING MOMENT COEFFICIENT
EFFECT OF SIDESLIP

THE BOEING COMPANY

C.R4 K.E ALDAN

REV. 0

ROLLING MOMENT COEFFICIENT
EFFECT OF SIDESLIP

THE BOEING COMPANY

CALC STIRLING 11-7-67
CHECK KUPCIS 11-7-67
APR APR
INK ODEGARD 2-9-70

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LOW 6-4-69 BECK 1-30-70

PAGE 5.0-7

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ROLLING MOMENT COEFFICIENT
EFFECT OF MACH NUMBER ON $dC_\alpha/d\beta$

THE BOEING COMPANY

CALC
STIRLING 11-29-67
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REV. D
\[ F_{r,ge} = \left[ 1 + F_{r,ge}k^B \right] \]

Where \( k^B \) is shown on page 2 of 51.
NOTE: \( \dot{\phi} = \frac{p_b \beta}{2v} \); \( p_b \sim \text{rad/sec} \), \( \beta \sim \text{ft/sec} \) (TRUE AIRSPEED)

\( \frac{dC_1}{d\dot{\phi}} \) per radian

FLAPS:
- 30
- 25
- 20
- 15
- 10
- 5
- 0
- 5

\( \delta_{\text{WBR}} \) ~ DEGREES

ROLLING MOMENT COEFFICIENT

EFFECT OF ROLL RATE

THE BOEING COMPANY
ROLLING MOMENT COEFFICIENT
AEROElastic EFFECT ON ROLLING MOMENT
COEFFICIENT DUE TO ROLL RATE

THE BOEING COMPANY
**SPOILER PANEL B OR 12**

**NOTE**
- Data shown for individual panels B or 12.
- For panel 1 or 5, reverse sign.
- Panels 1 & 5 limited to 20 deg. max. deflection.

**SPOILER PANEL GROUP 9, 10, 11**

**NOTE**
- Total effect of spoiler group 9, 10, 11 shown.
- For spoiler group 2 & 4, reverse sign.
- With hydraulic system no. 2 off, multiply by 0.32.
- With hydraulic system no. 3 off, multiply by 0.68.

**ROLLING MOMENT COEFFICIENT**

**EFFECT OF SPOILERS**

THE BOEING COMPANY

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NOTE: USE FOR ALL SPOILER PANELS.

2. USE FOR ALL FLAP SETTINGS.
NOTE: SYMMETRIC FOR UP AND DOWN AILERON

EFFECTIVENESS FACTOR
INBOARD AILERONS

THE BOEING COMPANY

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CHECK  FOSTER  1-24-68  LOW  6-4-69
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APR
INK  KINSMAN  1-27-70

EFFECTIVENESS FACTOR
INBOARD AILERONS

THE BOEING COMPANY
NOTE

\[(\Delta C_{l, a})_{10}\]

is shown for right aileron up only.

For full lateral control (right aileron up and left aileron down), use \[2 \times (\Delta C_{l, a})_{20}\]

---

**Rolling Moment Coefficient**

**Effect of Inboard Aileron**

**The Boeing Company**

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**Page 5-0-23**

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NOTE: USE FOR ALL FLAP SETTINGS

EFFECTIVENESS FACTOR
OUTBOARDAILERONS

THE BOEING COMPANY

CALC: KUPCIS 11-16-67
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APR: BECK 1-27-70
INK: KINSMAN 1-27-70

REVISED DATE
LOW 4-4-69

DEGREE

0 5 10 15 20 25

K_{\text{KA0}}
ROLLING MOMENT COEFFICIENT
AERODYNAMIC EFFECT ON ROLLING MOMENT
COEFFICIENT DUE TO INBOARD AILERONS
THE BOEING COMPANY

CALC  KUPCIC  10/31/67  REVISED  DATE
CHECK  FOSTER  1-29-67  LOW  6-4-69

INK  ODEGARD  10/31/67

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REV 0
\[ F_{\text{GLE}} = \frac{s^2}{1 + k s} \]

NOTE: \( F_{\text{GLE}} \) is shown on page 5-0-11.
ROLLING MOMENT COEFFICIENT

AEROELASTIC EFFECT ON ROLLING MOMENT

COEFFICIENT DUE TO UPPER RUDDER

THE BOEING COMPANY

PAGE 50-31
NOTE

1. RIGHT INBOARD FLAP AT MONITOR LIMITED EXTENSION POSITION
   CORRESPONDING TO LEFT INBOARD FLAP FAILURE POSITION

2. CHANGE SIGN FOR RIGHT INBOARD FLAP FAILURE

THESE DATA NOT INCLUDED
IN NASA SIMULATION

ROLLING MOMENT COEFFICIENT
EFFECT OFASYMMETRIC INBOARD FLAP FAILURE
FOR FLAP EXTENSION

THE BOEING COMPANY
NOTE

RIGHT INBOARD FLAP AT MONITOR LIMITED RETRACTION POSITION
CORRESPONDING TO LEFT INBOARD FLAP FAILURE POSITION

1. CHANGE SIGN FOR LEFT INBOARD FLAP FAILURE

2. EFFECT OF ASYMMETRIC INBOARD FLAP FAILURE

THESE DATA NOT INCLUDED IN NASA SIMULATION

ROLLING MOMENT COEFFICIENT

EFFECT OF ASYMMETRIC INBOARD FLAP FAILURE FOR FLAP RETRACTION

THE BOEING COMPANY
RIGHT OUTBOARD FLAP AT MONITOR LIMITED EXTENSION POSITION

CORRESPONDING TO LEFT OUTBOARD FLAP FAILURE POSITION.

CHANGE SIGN FOR RIGHT OUTBOARD FLAP FAILURE.

THESE DATA NOT INCLUDED IN NASA SIMULATION.

CALC  LOW   5-24-69  REVIS D  DATE
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ROLLING MOMENT COEFFICIENT
EFFECT OF ASYMMETRIC OUTBOARD FLAP FAILURE
FOR FLAP EXTENSION

THE BOEING COMPANY

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PAGE
6.0-35
NOTE: RIGHT OUTBOARD FLAP AT MONITOR LIMITED RETRACTION POSITION

CORRESPONDING TO LEFT OUTBOARD FLAP FAILURE POSITION

2. CHANGE SIGN FOR RIGHT OUTBOARD FLAP FAILURE

THESE DATA NOT INCLUDED IN NASA SIMULATION.

ΔC 2 OUTBO FAILURE

0.012

LEFT OUTBO FLAP FAILED AT 30

0.010

0.008

0.006

0.004

0.002

-0.002

-5 0 5 10 15 20 25

X W.B.F. DEGREES

ROLLING MOMENT COEFFICIENT

EFFECT OF ASYMMETRIC OUTBOARD FLAP FAILURE

FOR FLAP RETRACTION

THE BOEING COMPANY

| CALC | LOW  | 5-24-69 | REVISED | DATE
|------|------|---------|---------|------
| CHECK | LOW | 2-17-70 |
| APR | LOW | 6-25-70 |
| INK ODEGARD | 5-24-69 |
NOTE: CHANGE ASYMM. FOR FAILURE OF L.E. FLAP SEGMENTS 7, 8, 9, 14, 16.

These data not included in NASA simulation.

LE FLAP SEGMENTS 1, 2, 3, 4, 5 failed to extend.
LE FLAP SEGMENTS 17, 23, 24, 25, 26 failed to retract.

ROLLING MOMENT COEFFICIENT
EFFECT OF ASYMMETRIC L.E. FLAP SEGMENTS 1, 2, 3, 4, 5 OR 22, 23, 24, 25, 26

THE BOEING COMPANY
NOTE: CHANGE 3°C FOR FAILURE OF L.E. FLAP SEGMENTS 19, 20, 21

THIRD DATA NOT INCLUDED IN NASA SIMULATION

ROLLING MOMENT COEFFICIENT EFFECT OF ASYMMETRIC L.E. FLAP SEGMENTS 4, 7, 8 OR 19, 20, 21

THE BOEING COMPANY
6.0  YAWING MOMENT COEFFICIENT

The dimensionless aerodynamic yawing moment coefficient is given in terms of its significant components by the equation below.

At a given $\alpha_{w.d.r.}$,

$$C_{n_{c.o.}} = \frac{dC_n}{d\beta} \beta + \frac{dC_n}{d\beta} \frac{\beta b}{2V} + \frac{dC_n}{d\beta} \frac{R_b}{2V} + \frac{dC_n}{d\beta} \frac{R_b b}{2V}$$

$$+ \Delta C_n_{spoilers} + \Delta C_n_{inboard ailerons} + \Delta C_n_{outboard ailerons}$$

$$+ \Delta C_n_{rudders} + \left[ \Delta C_n_{flap failure} + \Delta C_n_{l.e. failure} \right]$$

where,

$\frac{dC_n}{d\beta} \beta$ = Yawing moment coefficient due to angle of sideslip, $\beta$.

The complete expression for $\frac{dC_n}{d\beta}$ is given as follows:

$$\frac{dC_n}{d\beta} = \left( \frac{dC_n}{d\beta} \frac{(C_n\beta)_m}{(C_n\beta)_{m=0}} \right) \cdot F_{n_{pe}} + \frac{dC_v}{d\beta} (c.g. - .25) \frac{E}{b}$$

where $\left( \frac{dC_n}{d\beta} \right)$ is the basic rate of change of yawing moment coefficient due to angle of sideslip. For low speed, $\left( \frac{dC_n}{d\beta} \right)$ is plotted on page 6.0-6. The aeroelastic effect, $\left( \frac{(C_n\beta)_{m}}{(C_n\beta)_{m=0}} \right)$, is plotted on page 6.0-6. The ground effect factor, $F_{n_{pe}}$, is obtained as follows:

[ ]* NOT IN NASA SIMULATION
6.0

(Cont'd)

\[ F_{n,E} = \left[ 1 + F_{n, \theta} \cdot K_{GE} \right] \]

where the ground effect sideslip factor, \( F_{n, \theta} \), is plotted on page 6.0-7. The ground effect height factor, \( K_{GE} \), is plotted on page 2.0-31. \( \frac{dC_{v}}{d\beta} \) is obtained from page 7.0-1.

\[ \frac{dC_{n}}{d\beta} \cdot \frac{\beta}{2V} \]

is Yawing moment coefficient due to rate of change of sideslip angle. The complete expression for \( \frac{dC_{n}}{d\beta} \) is given as follows:

\[ \frac{dC_{n}}{d\beta} = \left( \frac{dC_{n}}{d\beta} \right) \cdot \frac{(Cn\beta)_{M}}{(Cn\beta)_{M=0}} \]

where \( \left( \frac{dC_{n}}{d\beta} \right) \) is plotted on page 6.0-8. The aeroelastic effect, \( \left( \frac{Cn\beta}{(Cn\beta)_{M=0}} \right) \), is plotted on page 6.0-8.

\[ \frac{dC_{n}}{d\beta} \cdot \frac{P_{\theta} \beta}{2V} \]

is Yawing moment coefficient due to roll rate about the stability axis, \( \chi_{s} \). \( \frac{dC_{n}}{d\beta} \) is plotted on page 6.0-9.

\[ \frac{dC_{n}}{\Omega} \cdot \frac{r_{\phi}}{2V} \]

is Yawing moment coefficient due to yaw rate about the stability axis, \( \zeta_{s} \). The complete expression for \( \frac{dC_{n}}{\Omega} \) is given as follows:

\[ \frac{dC_{n}}{\Omega} = K_{\phi} \cdot \left( \frac{dC_{n, \phi}}{d\Omega} \right) \cdot \frac{(Cn\theta)_{M}}{(Cn\theta)_{M=0}} \]

where \( \left( \frac{dC_{n, \phi}}{d\Omega} \right) \) and the center of gravity factor, \( K_{\phi} \),
6.0

are plotted on page 6.0-10. The aeroelastic effect, 

\[
\frac{(Cn_p)_M}{(Cn_p)_{M=0}}
\]

is plotted on page 6.0-10.

\[\Delta C_{n_{spoilers}}\] Yawing moment coefficient due to spoiler deflection.

\[\Delta C_{n_{spoilers}} = \sum_{\text{operating spoiler panels}} (K_{\delta_{sp}})_{n} \cdot (\Delta C_{n_{sp}})_{45} \frac{(Cn_p)_M}{(Cn_p)_{M=0}} \cdot F_{n_{GE}}\]

where \((\Delta C_{n_{sp}})_{45}\) is the yawing moment coefficient due to deflecting the operating spoiler panels to 45°. \((\Delta C_{n_{sp}})_{45}\) is plotted on page 6.0-12.

The spoiler effectiveness factor, \((K_{\delta_{sp}})_{n}\), is plotted on page 6.0-11. The Mach number effect, \(\frac{(Cn_p)_M}{(Cn_p)_{M=0}}\), is plotted on page 6.0-13. The ground effect factor, \(F_{n_{GE}}\), is obtained as follows:

\[F_{n_{GE}} = \left[1 + F_{n} \cdot K_{GE}^B\right]\]

where the ground effect lateral control factor, \(F_{n}\), is plotted on page 6.0-14. The ground effect height factor, \(K_{GE}^B\), is plotted on page 2.0-31.

\[\Delta C_{n_{inboard ailerons}}\] Yawing moment coefficient due to inboard aileron deflection.

\[\Delta C_{n_{inboard ailerons}} = \sum_{\text{left and right inboard ailerons}} K_{\delta_{A_1}} \cdot (\Delta C_{n_{A_1}})_{20} \frac{(Cn_{A_1})_{M=0}}{(Cn_{A_1})_{M=0}} \cdot F_{n_{GE}}\]

where \((\Delta C_{n_{A_1}})_{20}\) is the yawing moment coefficient due to deflecting one inboard aileron up to 20° or
the opposite inboard aileron down to 20°. \( \Delta C_{n_{\text{A}1}} \) is plotted on page 6.0-15. The inboard aileron effectiveness factor, \( K_{5_{\text{A}1}} \), is plotted on page 5.0-22. The Mach number effect, \( \frac{\Delta C_{n_{\text{A}1}}}{\Delta C_{n_{\text{A}1}}} \), is plotted on page 6.0-15. The ground effect factor, \( F_{n_{\text{GE}}} \), is obtained from page 6.0-14.

\[ \Delta C_{n_{\text{OUTBOARD AILERONS}}} = \sum K_{5_{\text{A}O}} \cdot \Delta C_{n_{\text{AO}}} \cdot F_{n_{\text{GE}}} \]

where \( \Delta C_{n_{\text{AO}}} \) is the yawing moment coefficient due to deflecting one outboard aileron up to 25° or the opposite outboard aileron down to 15°. \( \Delta C_{n_{\text{AO}}} \) is plotted on page 6.0-16. The outboard aileron effectiveness factor, \( K_{5_{\text{A}O}} \), is plotted on page 5.0-26. The ground effect factor, \( F_{n_{\text{GE}}} \), is obtained from page 6.0-14.

\[ \Delta C_{n_{\text{RUDDERS}}} = K_{5_{\text{RU}}} \cdot \Delta C_{n_{\text{RU}}} \cdot \frac{C_{n_{\text{RU}}}}{C_{n_{\text{RU}}}} + K_{5_{\text{RL}}} \cdot \Delta C_{n_{\text{RL}}} \cdot \frac{C_{n_{\text{RL}}}}{C_{n_{\text{RL}}}} + \Delta C_{n_{\text{RU}}} \cdot (c.g.-25) \frac{E_b}{d} \]

where \( \Delta C_{n_{\text{RU}}} \) and \( \Delta C_{n_{\text{RL}}} \) are the yawing moment coefficients due to full deflection of the upper rudder and the lower rudder respectively.


(Cont'd)  

\[ (\Delta C_{n_{ RU}})_{25} \] and \[ (\Delta C_{n_{ RL}})_{25} \] are plotted on page 6.0-19. The upper rudder effectiveness factor, \( K_{RU} \), and the lower rudder effectiveness factor, \( K_{RL} \), are plotted on page 6.0-17 and on page 6.0-18 respectively. The aeroelastic effects, \( \frac{(C_{n_{RU}})_{M}}{(C_{n_{RU}})_{M=0}} \) and \( \frac{(C_{n_{RL}})_{M}}{(C_{n_{RL}})_{M=0}} \), for the upper rudder and the lower rudder, are plotted on page 6.0-20 and page 6.0-21 respectively. It should be noted that for rudder deflection with one rudder inoperative, the appropriate rudder contribution should be multiplied by 1.12.

\[ \Delta C_{Y \text{ RUDDERS}} \] is obtained from page 7.0-3.

\[ \Delta C_{n_{\text{FLAP FAILURE}}} = \text{Yawing moment coefficient due to an asymmetric (monitor limited) inboard or outboard flap failure.} \]

\[ \Delta C_{n_{\text{FLAP FAILURE}}} = \Delta C_{n_{\text{INBD FAILURE}}} \text{ OR } \Delta C_{n_{\text{OUTBD FAILURE}}} \]

where \( \Delta C_{n_{\text{INBD FAILURE}}} \) is the yawing moment coefficient due to an asymmetric (monitor limited) inboard flap failure. \( \Delta C_{n_{\text{INBD FAILURE}}} \) for flap extension or retraction is plotted on pages 6.0-22 and 6.0-23 respectively.

\( \Delta C_{n_{\text{OUTBD FAILURE}}} \) for flap extension or retraction is plotted on pages 6.0-24 and 6.0-25 respectively.

\[ \Delta C_{n_{\text{L.E. FAILURE}}} = \text{Yawing moment coefficient due to asymmetric leading edge flap failure.} \]

\( \Delta C_{n_{\text{L.E. FAILURE}}} \) is plotted on pages 6.0-26 and 6.0-27.
NOTE: \[ F_{n,p_{G E}} = [1 + F_{n,p} K_{G E}^2] \]

WHERE \( K_{G E}^2 \) IS SHOWN ON PAGE 2.0-31.

YAWING MOMENT COEFFICIENT
GROUND EFFECT SIDESLIP FACTOR, \( F_{n,p} \)

THE BOEING COMPANY

DATE
12/29/71
12/26/71
2/14/70

STIRLING
KUPCIS
LOW

CHECK
APR
APR

INK
ODEGARD
12/24/71
12/29/71
12/23/71

REvised
1
1
1

PAGe
6.0-7
6.0-7
6.0-7

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REV. D
REV. D

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De-30643
De-30643

747
747
747
NOTE: $\dot{\alpha} = \frac{\dot{\alpha}}{V}$, $\alpha = \text{deg}$, $V = \text{ft/sec}$ (true airspeed)

1. Use for all flap settings.

2. $dC_n_{125}/d\dot{\alpha}$ per radian

YAWING MOMENT COEFFICIENT $\dot{\alpha}$

EFFECT OF $\dot{\alpha}$

THE BOEING COMPANY

CALC RICHARDSON 11-13-67
CHECK CURNUTT 11-14-67
APR LUDWIG 11-3-69
APR CURNUTT 2-25-70
INK ODEGARD 3-3-70

MACH NUMBER $M$
NOTE 1: USE FOR ALL SPOILER PANELS

2. PANELS 645 LIMITED TO 20 DEG MAX. DEFLECTION

NOTE: 1. USE FOR ALL SPOILER PANELS

2. PANELS 645 LIMITED TO 20 DEG MAX. DEFLECTION

Diagram showing a graph with the title "YAWING MOMENT COEFFICIENT EFFECTIVENESS FACTOR SPOILERS"

THE BOEING COMPANY

747
SPOILER PANEL B OR 12

NOTE 1. DATA SHOWN FOR INDIVIDUAL PANELS B OR 12
2. FOR PANEL 1 OR 5, REVERSE SIGN
3. PANELS B & 5 LIMITED TO 20 DEG. MAX. DEFLECTION

SPOILER PANEL GROUP 9, 10, 11

NOTE 1. TOTAL EFFECT OF SPOILER GROUP 9, 10, 11 SHOWN
2. FOR SPOILER GROUP 2, 3, 4, REVERSE SIGN
3. WITH HYDRAULIC SYSTEM NO. 2 OFF, MULTIPLY BY 0.67
4. WITH HYDRAULIC SYSTEM NO. 3 OFF, MULTIPLY BY 0.67

YAWING MOMENT COEFFICIENT EFFECT OF SPOILERS

THE BOEING COMPANY
NOTE: \[ F_{n_{CE}} = \left[ 1 - F_{n} - K_{s_{CE}}^2 \right] \]

WHERE \( K_{s_{CE}} \) is shown on page 2.0-31.
NOTE 1  
\[(\Delta C_{n_m})_A\] AS SHOWN FOR RIGHT AILERON UP ONLY

\[(\Delta C_{n_m})_L\] AS SHOWN FOR FULL LATERAL CONTROL (RIGHT AILERON UP AND
LEFT AILERON DOWN), USE \[2 \times (\Delta C_{n_{m_A}})_A\]

NOTE 2  
\[(\Delta C_{n_A})_L\] AS SHOWN FOR LEFT AILERON DOWN ONLY

WING MOVEMENT

WING MOVEMENT

CALC: KUPCIS 11-4-67  REVISION: 2-14-70
CHECK: FOSTER 1-24-68  LOW 2-14-70
APR
APR
INK: W. ODEGARD 11-4-67

THE BOEING COMPANY

747

EFFECT OF INBOARD AILERON

REV. D
EFFECTIVENESS FACTOR
LOWER RUDDER

THE BOEING COMPANY
EFFECT OF RUDDERS

NOTE: FOR LOWER RUDDER INOPERATIVE, MULTIPLY BY 1.12

\[(\Delta C_{pR,L})_{25}^{.04}\]

\[(\Delta C_{pR,L})_{25}^{.04}\]

X-WAP - DEGREES

NOTE: FOR UPPER RUDDER INOPERATIVE, MULTIPLY BY 1.12

YAWING MOMENT COEFFICIENT
EFFECT OF RUDDERS

THE BOEING COMPANY
NOTE: FOR ALL FLAP SETTINGS

YAWING MOMENT COEFFICIENT
AEROLESTIC EFFECT ON YAWING MOMENT
COEFFICIENT DUE TO UPPER RUDDER

THE BOEING COMPANY
NOTE
RIGHT INBOARD FLAP AT MONITOR LIMITED RETRACTION POSITION CORRESPONDING TO LEFT INBOARD FLAP FAILURE POSITION

CHANGE SIGN FOR RIGHT INBOARD FLAP FAILURE

These data not included in NASA simulation.

---

EFFECT OF ASYMMETRIC INBOARD FLAP FAILURE FOR FLAP RETRACTION

THE BOEING COMPANY

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PAGE 6.0-23
NOTE

RIGHT OUTBOARD FLAP AT MONITOR LIMITED EXTENSION POSITION

CORRESPONDING TO LEFT OUTBOARD FLAP FAILURE POSITION

2. CHANGE SIGN FOR RIGHT OUTBOARD FLAP FAILURE

THESE DATA NOT INCLUDED IN NASA SIMULATION.

\[ \Delta C_n \text{ OUTBD FAILURE} \]

\[ 0.005 \]

\[ 0.004 \]

\[ 0.003 \]

\[ 0.002 \]

\[ 0.001 \]

\[ \text{LEFT OUTBD FLAPS FAILED AT 25} \]

\[ \text{X-WAB DEGREES} \]

\[ -5 \]

\[ 0 \]

\[ 5 \]

\[ 10 \]

\[ 15 \]

\[ 20 \]

\[ 25 \]

YAWING MOMENT COEFFICIENT EFFECT OF ASYMMETRIC OUTBOARD FLAP FAILURE FOR FLAP EXTENSION

THE BOEING COMPANY

LOW 5.2649

REvised DATE

APR 2.17.70

APR 6.25.70

INK ODEGARD 5.2649

PAGE 6.0-24
NOTE: 1. RIGHT OUTBOARD FLAP AT MONITOR LIMITED RETRACTION POSITION
   CORRESPONDING TO LEFT OUTBOARD FLAP FAILURE POSITION
   2. CHANGE SIGN FOR RIGHT OUTBOARD FLAP FAILURE

THESE DATA NOT INCLUDED
IN NASA SIMULATION.

\[ AC_{f1} \] OUTBOARD FAILURE

\[ X_{wbr} = \text{DEGREES} \]

CALC  LOW 5-26-69  REVISIONS  DATE
CHECK LOW 2-17-70
APR LOW 6-25-70
INK ODEGARD 5-26-69

YAWING MOMENT COEFFICIENT
EFFECT OF ASYMMETRIC OUTBOARD FLAP FAILURE
FOR FLAP RETRACTION

THE BOEING COMPANY

PAGE 6.0-25

THESE DATA NOT INCLUDED IN NASA SIMULATION.

L.E. FLAP SEGMENTS 1, 2, 3, 4, 5 FAILED TO EXTEND

L.E. FLAP SEGMENTS 1, 2, 3, 4, 5 FAILED TO RETRACT

Yawing Moment Coefficient Effect of Asymmetric L.E. Flap Segments 1, 2, 3, 4, 5 or 22, 23, 24, 25, 26

The Boeing Company
NOTE 1. CHANGE SIGN FOR FAILURE OF L.E. FLAP SEGMENTS 19,20,21

FLAP DATA NOT INCLUDED
IN THIS SIMULATION.

LE FLAP SEGMENTS 6,7,8 FAILED TO EXTEND
LE FLAP SEGMENTS 4,7,8 FAILED TO RETRACT

\( \Delta C_n \)
L.E. FAILURE

\(-0.024\)
\(-0.020\)
\(-0.016\)
\(-0.012\)
\(-0.008\)
\(-0.004\)
0
\(0.004\)

\( \alpha_{W.D.P.} \) - DEGREES

0 5 10 15 20 25

YAWING MOMENT COEFFICIENT
EFFECT OF ASYMMETRIC L.E. FLAP
SEGMENTS 6,7,8 OR 19,20,21

THE BOEING COMPANY
7.0 SIDE FORCE COEFFICIENT

The dimensionless aerodynamic side force coefficient is given in terms of its significant components by the equation below.

At a given $\alpha_{w.d.p.}$,

$$C_Y = \frac{dC_Y}{d\beta} \cdot \beta + \frac{dC_Y}{d\beta} \cdot \frac{\rho_{sb}}{2V} + \frac{dC_Y}{d\beta} \cdot \frac{r_{sb}}{2V} + \Delta C_Y_{\text{sloiers}} + \Delta C_Y_{\text{rudders}} + \left[ \Delta C_Y_{\text{flap failure}} \right] + \Delta C_Y_{\text{L.E. failure}}$$

where,

$$\frac{dC_Y}{d\beta} \cdot \beta = \text{Side force coefficient due to angle of sideslip, } \beta.$$

The complete expression for $\frac{dC_Y}{d\beta}$ is given as follows:

$$\frac{dC_Y}{d\beta} = \left[ \left( \frac{dC_Y}{d\beta} \right) \cdot \frac{(C_Y)_{M}}{(C_Y)_{M=0}} + K_{\text{gear}} \cdot \Delta \left( \frac{dC_Y}{d\beta} \right)_{\text{landing gear}} \right] F_Y_{\text{gear}}$$

where $\left( \frac{dC_Y}{d\beta} \right)$ is the basic rate of change of side force coefficient due to angle of sideslip. For low speed, $\left( \frac{dC_Y}{d\beta} \right)$ is plotted on page 7.0-5. The aeroelastic effect, $\frac{(C_Y)_{M}}{(C_Y)_{M=0}}$, is plotted on page 7.0-5. The effect of main and nose gear extension, $\Delta \left( \frac{dC_Y}{d\beta} \right)_{\text{landing gear}}$ is given by:

$$\Delta \left( \frac{dC_Y}{d\beta} \right)_{\text{landing gear}} = -0.002 \text{ per degree}$$

[ ]* NOT IN NASA SIMULATION
The landing gear effectiveness factor, $K_{GEAR}$ is plotted on page 2.0-28.

The ground effect factor, $F_{Y_{GE}}$ is obtained as follows:

\[ F_{Y_{GE}} = \left[ 1 + F_{Y\beta} \cdot K_{GE}^{B} \right] \]

where the ground effect sideslip factor, $F_{Y\beta}$, is plotted on page 7.0-6. The ground effect height factor, $K_{GE}^{B}$ is plotted on page 2.0-31.

\[
\frac{dC_{Y}}{d\beta} = \frac{P_{sb}}{2V} = \text{Side force coefficient due to roll rate about the stability axis, } x_{E}. \quad \frac{dC_{Y}}{d\beta} \text{ is plotted on page 7.0-7.}
\]

\[
\frac{dC_{Y}}{d\phi} = \frac{r_{sb}}{2V} = \text{Side force coefficient due to yaw rate about the stability axis, } z_{E}. \text{ The complete expression for } \frac{dC_{Y}}{d\phi}, \text{ is given as follows:}
\]

\[
\frac{dC_{Y}}{d\phi} = \left( \frac{dC_{Y}}{d\phi} \right) \cdot \left( \frac{C_{Yf}}{C_{Yf}} \right)_{M=0}
\]

where $\left( \frac{dC_{Y}}{d\phi} \right)$ is plotted on page 7.0-8. The aeroelastic effect, $\left( \frac{C_{Yf}}{C_{Yf}} \right)_{M=0}$ is plotted on page 7.0-9.

$\Delta C_{YSPOILERS}$ = Side force coefficient due to spoiler deflection.

\[
\Delta C_{YSPOILERS} = \sum_{\text{OPERATING Spoiler PANELS}} \left( K_{SBSP} \right) \cdot (\Delta C_{YS})_{45} \cdot \left( \frac{C_{YS}}{C_{YS}} \right)_{M=0} \cdot F_{N_{GE}}
\]

where $(\Delta C_{YS})_{45}$ is the side force coefficient due to deflecting the operating spoiler panels to $45^\circ$.

$(\Delta C_{YS})_{45}$ is plotted on page 7.0-11. The spoiler
effectiveness factor, \((K_{spg})_Y\), is plotted on page 7.0-10. The Mach number effect, \(\frac{C_{ysp}}{C_{ysp}}_{M=0}\), is plotted on page 7.0-12. The ground effect factor, \(F_{nge}\), is obtained from page 6.0-14.

\[ \Delta C_{yrudders} = \text{Side force coefficient due to rudder deflection.} \]

\[
\Delta C_{yrudders} = K_{sru} \cdot (\Delta C_{yru})_{25} \frac{C_{yru}}{C_{yru} \cdot M=0} + K_{srl} \cdot (\Delta C_{yrl})_{25} \frac{C_{yrl}}{C_{yrl} \cdot M=0}
\]

where \((\Delta C_{yru})_{25}\) and \((\Delta C_{yrl})_{25}\) are the side force coefficients due to full deflection of the upper rudder and the lower rudder respectively. \((\Delta C_{yru})_{25}\) and \((\Delta C_{yrl})_{25}\) are plotted on page 7.0-13. The upper rudder effectiveness factor, \(K_{sru}\), and the lower rudder effectiveness factor, \(K_{srl}\), are plotted on pages 6.0-17 and 6.0-18 respectively. The aero-elastic effects, \(\frac{C_{yru}}{C_{yru} \cdot M=0}\) and \(\frac{C_{yrl}}{C_{yrl} \cdot M=0}\), for the upper rudder and the lower rudder, are plotted on pages 7.0-14 and 7.0-15 respectively.

\[ \Delta C_{yflap failure} = \text{Side force coefficient due to an asymmetric (monitor limited) inboard or outboard flap failure.} \]

\[ \Delta C_{yflap failure} = \Delta C_{y inbd failure} \text{ OR } \Delta C_{y outbd failure} \]

where \(\Delta C_{y inbd failure}\) is the side force coefficient due to an asymmetric (monitor limited) in-board flap failure. \(\Delta C_{y inbd failure}\) for flap extension or retraction is plotted on pages 7.0-16 and 7.0-17.
respectively.

(Cont'd) \( \Delta C_{Y_{\text{OUTED FAILURE}}} \) for flap extension or retraction is plotted on pages 7.0-18 and 7.0-19 respectively.

\( \Delta C_{Y_{\text{L.E. FAILURE}}} \) = Side force coefficient due to asymmetric leading edge flap failure. \( \Delta C_{Y_{\text{L.E. FAILURE}}} \) is plotted on pages 7.0-20 and 7.0-21.
NOTE: 

$$F_{y\phi_{ge}} = k_{ge}^2 F_{y\phi}$$

where \( k_{ge} \) is shown on page 2.0-2.
Note: 

1. Use for all spoiler panels.

2. Panels 6 & 5 limited to 20 deg. max deflection.

**Diagram:**

The Boeing Company

**Table:**

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**Side Force Coefficient**

Effectiveness Factor

Spoilers

**THE BOEING COMPANY**

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Page 70-10
**Spoiler Panel 6 or 12**

Note:
1. Data shown for individual panels 6 or 12.
2. For Panel 10 or 5, reverse sign.
3. Panels 6 & 5 limited to 20 deg. max. deflection.

** Spoiler Panel Group 5, 10, 11**

Note:
1. Total effect of spoiler group 5, 10, 11 shown.
2. For spoiler group 2, 3, 4, reverse sign.
3. With hydraulic system No. 2 off, multiply by 0.84.
4. With hydraulic system No. 3 off, multiply by 0.60.

---

**Side Force Coefficient Effect of Spoilers**

The Boeing Company
SIDE FORCE COEFFICIENT EFFECT OF RUDDERS

THE BOEING COMPANY
SIDE FORCE COEFFICIENT
AEROELASTIC EFFECT ON SIDE FORCE
COEFFICIENT DUE TO LOWER RUDDER

THE BOEING COMPANY
NOTE
RIGHT INBOARD FLAP AT MONITOR LIMITED EXTENSION POSITION
CORRESPONDING TO LEFT INBOARD FLAP FAILURE POSITION

2. CHANGE SIGN FOR RIGHT INBOARD FLAP FAILURE

THOSE DATA NOT INCLUDED
IN NASA SIMULATION.

SIDE FORCE COEFFICIENT
EFFECT OF ASYMMETRIC INBOARD FLAP FAILURE
FOR FLAP EXTENSION

THE BOEING COMPANY

CALC  LOW  52469  REVISION  DATE
CHECK   LOW  21770
APR  LOW  62570
APR
INK  ODEGARD  52469

PAGE 7.0-16
NOTE

RIGHT INBOARD FLAP AT MONITOR LIMITED RETRACTION POSITION

CORRESPONDING TO LEFT INBOARD FLAP FAILURE POSITION

1. CHANGE SIGN FOR LEFT INBOARD FLAP FAILURE

2. CHANGE SIGN FOR RIGHT INBOARD FLAP FAILURE

SIDE FORCE COEFFICIENT

EFFECT OF ASYMMETRIC INBOARD FLAP FAILURE

FOR FLAP RETRACTION

THE BOEING COMPANY

CALC LOW 5/26/69 REVISIONS DATE
CHECK LOW 2/17/70
APR LOW 6/25/70
INK ODEGARD 5/26/69

PAGE 7.0-17

Dr. 30643, 191117
NOTE

1. RIGHT OUTBOARD FLAP AT MONITOR LIMITED EXTENSION POSITION
   CORRESPONDING TO LEFT OUTBOARD FLAP FAILURE POSITION

2. CHANGE SIGN FOR RIGHT OUTBOARD FLAP FAILURE

SIDE FORCE COEFFICIENT

EFFECT OF ASYMMETRIC OUTBOARD FLAP FAILURE
FOR FLAP EXTENSION

THE BOEING COMPANY

CALC: LOW 5-26-69  REVISION: DATE
CHECK: LOW 2-17-70
APR: LOW 6-25-70
INK ODEGARD 5-26-69

747

DE-30643
Vol. II
70-18
NOTE

RIGHT OUTBOARD FLAP AT MONITOR LIMITED RETRACTION POSITION

CORRESPONDING TO LEFT OUTBOARD FLAP FAILURE POSITION

2. CHANGE SIGN FOR RIGHT OUTBOARD FLAP FAILURE

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---

Diagram showing the effect of asymmetric outboard flap failure on side force coefficient.

These data not included in NASA simulation.

L.E. FLAP SEGMENTS 1, 2, 3, 4, 5 failed to extend.

L.E. FLAP SEGMENTS 1, 2, 3, 4, 5 failed to retract.

SIDE FORCE COEFFICIENT

EFFECT OF ASYMMETRIC L.E. FLAP SEGMENTS 1, 2, 3, 4, 5 OR 22, 23, 24, 25, 26

THE BOEING COMPANY
NOTE 1. CHANGE SIGN FOR FAILURE OF L.E. FLAP SEGMENTS 19, 20, 21.

SIDE FORCE COEFFICIENT
EFFECT OF ASYMMETRIC L.E. FLAP
SEGMENTS 6, 7, 8 OR 19, 20, 21

THE BOEING COMPANY
8.0 LONGITUDINAL CONTROL SYSTEM

A general description of the longitudinal system is presented in the Introduction on Pages 1.2-1 and 1.2-2 and in Volume I. A block diagram of the simulated elevator control system and stick force program is shown on Page 8.1-3. The data for each particular block can be found on the page numbers adjacent to the block.

8.1 Control Column Force and Elevator Deflection

The column travel in the 747 is 12.67° pull and -12.5° push. The column travel in the FSAA is +11°. The column deflection of the FSAA was scaled up by 1.15 (= 12.67/11) so that maximum column in the FSAA resulted in maximum column in the simulated 747.

The feel unit pressure, used in determining the feel unit torque, is

\[ P_f = \frac{dF_s}{d\gamma} \cdot (-120.9) \text{ lb/in}^2 \]

where

\[ \frac{dF_s}{d\gamma} = \left( -0.0025 \cdot q_c + "Fs" \right) \Rightarrow "FQC" \]

\[ P_f = \text{Feel unit pressure, lb/in}^2 \]

\[ \frac{dF_s}{d\gamma} = \text{Control column force gradient, lb/deg} \]

"Fs" = Column force gradient at \( q_c = 0 \). This is plotted on Page 8.1-7

"FQC" = Column force gradient \( q_c \) limit. This is plotted on Page 8.1-8
The stick force due to the column mass unbalance is:

\[ F_{S_{\text{mass}}} = n_z \left[ \theta_B + \delta_{\text{column}} - 5.4 \right] \cdot (-.275) \text{ lb} \]

The FSAA control loader was programmed with a breakout force plus a constant gradient for zero computed stick force. The control loader preload characteristics recorded by an x-y plotter are shown on Page 8.1-9. The preload characteristics were subtracted from the computed stick force on the NASA digital computer. The control loader was commanded with an incremental stick force (computed stick force minus preload stick force).
<table>
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<tr>
<th>A.Q.T.</th>
<th>F.U.T. $P_f=0$</th>
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**Diagram:**
- X-axis: A.Q.T. ~ IN.
- Y-axis: F.U.T. ~ IN.-L.B.
- Two curves: $P_f=0$ and $P_f=2000$ PSI.

**Text:**
- Feel Unit Torque vs. Aft Quadrant Travel
- The Boeing Company
- Page 8.1-5
- Document: 747
- Date: 11-15-69
- Revised: APR
- Check: APR
- TD 441 C-M
"FGC" FORC E GRADIENT $q_c$ LIMIT ~ LB/DEG

IMPACT PRESSURE, $q_c$ ~ LB/FT^2

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<td>NORDWALL</td>
<td>7-22-70</td>
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CONTROL COLUMN FORCE GRADIENT $q_c$ LIMIT

THE BOEING COMPANY
8.2 Elevator Control System

8.2.1 Elevator Limits - Boost On and Off

The maximum inboard and outboard elevator limits are shown on pages 8.2-2 and 8.2-3 for full and half boost operation, respectively.

An inboard or outboard elevator surface with both hydraulic systems off will trail at the float angles shown on pages 8.2-4 and 8.2-5, respectively.

8.2.2 Elevator Rigging

The elevator is downrigged at +2° from the faired position.

The outboard elevator angle is equal to the inboard elevator angle up to the blowdown angle of the outboard elevator.
NOTE: $\kappa_{\text{E, float}} = \left(\frac{\kappa_{\text{E, float}}}{\Delta_{\text{E, float}}}\right)^{-1}$

FLAPS UP

[Graph showing relationship between $\kappa_{\text{E, float}}$ and Mach number $M$.]

FLAPS DOWN

EFFECT OF STABILIZER

LONGITUDINAL CONTROL
INBOARD ELEVATOR FLOAT ANGLES

THE BOEING COMPANY

CALC: FOSTER 12-9-67
CHECK: HOLTZNER 1-15-68
INK: ODEGARD 2-6-70

REVISED DATE: LOW 2-6-70

PAGE 8.2-4

AD 481 C-R6
8.3 Stabilizer Trim System

8.3.1 Trim Rate

A block diagram of the simulated trim system is shown on Page 8.3-1. The stabilizer trim rate is programmed by impact pressure, $q_c$.

Page 8.3-2 shows the pilot and autopilot trim rates.

---

**Diagram:**

- **CAB**
- **THUMBSWITCH**
- **NASA DIGITAL COMPUTER**
- **|q_c||q_c||q_c||q_c|
- **+ΔΔ PILOT**
- **-ΔΔ PILOT**
- **+Δ Δ PILOT**
- **INITIAL**
- **POSITION LIMIT**
- **SET BY COMPUTER TRIM PROGRAM**

**STABILIZER TRIM**

---

**BOEING**

**No. D6-30643**

**SECT**

**Page 8.3-1**
9.0 LATERAL CONTROL SYSTEM

A general description of the system is presented in the Introduction on Pages 1.2-2, 1.2-3 and 1.2-4 and in Volume I. A block diagram of the NASA simulated lateral control system is shown on Page 9.1-2a. The data for each particular block can be found on the page number adjacent to the block. The computer requirements for the simulation were reduced by applying an equivalent rate limit to the wheel rather than to each spoiler panel. The outboard aileron lockout program utilized in the simulation was a function of flap screw travel, Page 9.2-8, rather than a function of time, as shown on Page 9.2-4.

9.1 Control Wheel Force and Angle

9.1.1 Control Wheel Force

\[ F_W = F_{WS} \pm F_{WFR} \]

where,

- \( F_W \) = Control wheel force, positive for a clockwise wheel moment (lb).
- \( F_{WS} \) = Control wheel force due to the spring and cam mechanism (lb). This is plotted on page 9.1-3.
- \( F_{WFR} \) = Friction force opposing control wheel motion (= 2.0 lb).
9.1.2 Control Wheel Angle

\[ \delta_W = \delta_{W_{\text{REF}}} + 0.35 F_W \]

where,

\[ \delta_W = \text{Control wheel angle (degrees). The control wheel limits are } \delta_W = \pm 88 \text{ degrees.} \]

\[ \delta_{W_{\text{REF}}} = \text{Reference control wheel angle (degrees). This does not include the effect of cable stretch and is the input to the aileron and spoiler programs plotted on pages 9.2-3 and 9.2-5 respectively.} \]

\[ 0.35 = \text{Cable stretch factor (deg/lb).} \]

9.1.3 Lateral Trim

\[ \delta_{W_{\text{TRIM}}} = \text{Zero force datum wheel angle due to trim (degrees).} \]

This is plotted on page 9.1-4. The trim shifts the wheel force datum but does not change the wheel limits. The trim limits are \( \pm 6.27 \) units. The nominal trim rate is 2.5 degrees of control wheel per second.
NOTE
1. $F_w = F_{w_s} + F_{w_{fr}}$
2. $F_{w_{fr}} = 2.0$ lb
3. SYMMETRIC FOR OPPOSITE WHEEL
4. CONTROL WHEEL LIMITS ARE $S_w = \pm 8\text{°}$

THESE DATA NOT INCLUDED IN NASA SIMULATION SEE P. (4.1-52)
9.2 Aileron-Spoiler-Wheel Program

9.2.1 Lateral Control Only

The inboard aileron wheel program is plotted on page 9.2-3.

For fully unlocked outboard ailerons, the outboard aileron wheel program is plotted on page 9.2-3. Full outboard aileron authority is available at all times for flaps 5, 10, 20, 25 and 30.

Outboard aileron unlocking is started at outboard flap jackscrew extension to 0% of jackscrew travel and full unlocking takes about 15 seconds. After this time, full outboard aileron authority is available. Outboard aileron locking is started at outboard flap jackscrew retraction to 0% of jackscrew travel and full locking takes about 15 seconds. After this time, no outboard aileron authority is available. For intermediate outboard aileron authority

$$\delta_{AO} = k_{\delta_{AO}} \cdot \delta_{AO,REF}$$

where,

$$\delta_{AO} = \text{Outboard aileron angle (degrees). The outboard aileron mechanical limits are 25° T.E. up and 15° T.E. down.}$$

$$k_{\delta_{AO}} = \text{Intermediate outboard aileron gain factor. This is a function of flap screw travel and is plotted on Page 9.2-4.}$$
9.2.1 \( \theta_{A_{0,\text{REF}}} \) = Reference outboard aileron angle (degrees) commanded from the aileron - wheel program, Page 9.2-3.

In the NASA simulation outboard ailerons are locked out when flaps are fully retracted.

The spoiler-wheel program is plotted on page 9.2-5.

9.2.2 Lateral Control With Speed Brake Operation

The aileron wheel program remains the same for all speed brake handle positions.

For intermediate and normal inflight speed brake handle positions, the spoiler wheel program is plotted on page 9.2-6.

9.2.3 Speed Brake Operation

The speed brake program with no lateral control inputs is plotted on page 9.2-7. Panels 3, 4, 5, 6, 7, 8, 9, 10 are used as inflight speed brakes (intermediate and inflight speed brake handle positions). The remaining panels 1, 2, 11, 12 operate only as ground speed brakes (ground speed brake handle position). Spoiler panels 6 and 7, which operate symmetrically as speed brakes only, cannot be modulated. These surfaces are either fully extended or fully retracted depending on the position of the speed brake handle.
NOTE
OUTBOARD AILERON PROGRAM IS FOR OUTBOARD
AILERONS FULLY UNLOCKED

LATERAL CONTROL
AILERON-WHEEL PROGRAM

THE BOEING COMPANY
OUTBOARD AILERON = OUTBOARD AILERON COMMANDED BY WHEEL \times \frac{k_{s_{A_0}}}{S_{A_0}}

\left( S_{A_0} = S_{A_0}^{\text{REF}} \cdot \frac{k_{s_{A_0}}}{S_{A_0}} \right)

REFER TO P. 9.2-3 FOR THE AILERON - WHEEL PROGRAM

\begin{align*}
\text{USE FOR UP AND DOWN AILERON}
\end{align*}

\begin{align*}
\% \text{ PST (FLAP SCREW TRAVEL)}
\end{align*}
NOTE: FOR SPEED BRAKES IN THE GROUND DETENT, USE THE IN-FLIGHT DETENT CURVES FOR PANELS 3, 4, 5, 6, 8, 9, 10. FOR PANELS 11 & 12 USE THE CURVE FOR PANELS 3, 4. FOR PANELS 11 & 12 USE THE CURVE FOR PANELS 9, 10. PANELS 4 & 7 REMAIN AT 20° FOR ALL WHEEL ANGLES.

SEE SECTION 19 FOR REVISED DATA.

LATERAL CONTROL
SPOILER PROGRAM AT COMBINED LATERAL CONTROL - SPEED BRAKES

THE BOEING COMPANY

747

DE-30643, Vol. II
PAGE 9.2-6
NOTE: SPEED BRAKE HANDLE FRICTION FORCE = 20 lb. PULL, 0 lb. PULL

2. MAXIMUM AVAILABLE IN-FLIGHT SPEED BRAKE HANDLE
   Position = 34 deg (IN-FLIGHT DETENT)

3. SPEED BRAKES BEYOND THE IN-FLIGHT DETENT
   ARE AVAILABLE ONLY ON THE GROUND.

SEE SECTION 19
FOR REVISED DATA
9.3 Control Surface Limits and Float Angles

The maximum inboard aileron and maximum outboard aileron angles for full boost and half boost are plotted on pages 9.3-2 and 9.3-3 respectively. The maximum spoiler angles are plotted on pages 9.3-4, 9.3-5 and 9.3-6.

Any inboard or outboard aileron surface with both hydraulic systems off, will trail at the float angles plotted on pages 9.3-7 and 9.3-8 respectively. The spoilers are held in a faired position, boost off, due to the hold down check valves.
NOTE 1  HYDRAULIC SYSTEMS 1 AND 5  
ON LEFT AILERON  

NOTE 2  HYDRAULIC SYSTEMS 2 AND 4  
ON RIGHT AILERON  

NOTE 3  FULL BOOST  
HALF BOOST  

\[ V_D, M_D \]  

MACH NUMBER, M  

MECHANICAL LIMIT  

T.E. DOWN  

LATERAL CONTROL  
INBOARD AILERON BLOWDOWN  

THE BOEING COMPANY  

PAGE 9.3-2
LATERAL CONTROL

OUTBOARD AILERON BLOWDOWN

THE BOEING COMPANY

NOTE 1 HYDRAULIC SYSTEMS 1 AND 2 ON LEFT AILERON

NOTE 2 HYDRAULIC SYSTEMS 3 AND 4 ON RIGHT AILERON

NOTE 3 FULL BOOST

NOTE 4 HALF BOOST

EQUIVALENT AIRSPEED, $V_E \sim$ KNOTS

120 140 160 180 200 220 240 260

CALC HOLTZNER 9-14-67 REVISED DATE
CHECK FOSTER 12-14-67 LAGREE 1-22-70
APR
APR
INK ODEGARD 9-14-67

PAGE 9.3-3
NOTE

2. INTERPOLATE LINEARLY FOR INTERMEDIATE FLAP SETTINGS.

THOSE DATA NOT INCLUDED.

FLAPS UP, 5
FLAPS 30
FLAPS 0
FLAPS UP, 7

MACH NUMBER, M

CALC: HOLTZNER 11-14-67
CHECK: FOSTER 1-15-68
APR: LOW 1-29-70
APR: INK ODEGARD 11-14-67

LATERAL CONTROL
OUTBOARD AILERON FLOAT ANGLES

THE BOEING COMPANY

747
06-30643
Vol. II
PAGE 9.3-8
10.0 DIRECTIONAL CONTROL SYSTEM

A general description of the system is presented in the Introduction on Pages 1.2-4 and 1.2-5 and in Volume I. A block diagram of the simulated directional control system is shown on Page 10.1-28. The data for each particular block can be found on the page number adjacent to the block.

10.1 Rudder Pedal Force and Angle

10.1.1 Rudder Pedal Force

\[ F_P = F_{PS} \pm F_{FR} \]

where,

\[ F_P \] = Rudder pedal force, positive for a left rudder pedal force (lb).

\[ F_{PS} \] = Rudder pedal force due to the spring and cam mechanism (lb). This is plotted on page 10.1-3.

\[ F_{FR} \] = Friction force opposing rudder pedal motion (= 6.0 lb).

10.1.2 Rudder Pedal Angle

\[ \delta_P = \delta_{PRE} + 0.023 F_P \]

where,
\( \delta_p \) = Rudder pedal angle (degrees). The rudder pedal limits are \( \delta_p = \pm 14 \) degrees.

\( \delta_{p_{\text{REF}}} \) = Reference rudder pedal angle (degrees). This does not include the effect of cable stretch.

\( .023 \) = Cable stretch factor (deg/lb).

10.1.3 Rudder Trim

\( \delta_{p_{\text{TRIM}}} \) = Zero force datum rudder pedal angle due to trim (degrees). This is plotted on page 10.1-4. The trim shifts the rudder pedal force datum but does not change the rudder limits. The trim limits are \( \pm 10 \) units.

10.1.4 Rudder Limiter

\( \delta_R = \frac{\delta_R}{\delta_{R_{\text{MAX}}}} \cdot \delta_{R_{\text{MAX}}} \)

where,

\( \frac{\delta_R}{\delta_{R_{\text{MAX}}}} \) = Rudder angle ratio. This is plotted on page 10.1-5.

\( \delta_{R_{\text{MAX}}} \) = Maximum available rudder angle (degrees). This is plotted on page 10.1-6.
NOTE
1. $F_p = F_{p0} + F_{pDR}$
2. $F_{pDR} = 6.0$ lb
3. SYMMETRIC FOR OPPOSITE RUDDER
4. RUDDER PEDAL LIMITS ARE $S_p = \pm 14^\circ$

THESE DATA NOT INCLUDED
IN NASA SIMULATION  SEE P.14.1-53

INCREMENTAL PEDAL ANGLE: $(S_p - S_p^{\text{trim}})$ - DEG

DIRECTIONAL CONTROL
FORCE DUE TO SPRING AND CAM MECHANISM

THE BOEING COMPANY
NOTE
1. SYMMETRIC FOR OPPOSITE TRIM
2. RUDDER PEDAL FORCE DATUM SHIFTS WITH TRIM
3. TRIM LIMITS ± 10 UNITS (± 0.36 TRIM WHEEL TURNS)
4. CLOCKWISE TRIM WHEEL ROTATION GIVES RIGHT Rudder
5. TRIM WHEEL OPERATING TORQUE ≈ 24 IN-MA
   [TRIM WHEEL RADIUS ≈ 2.07 IN]

---

DIRECTIONAL CONTROL
RUddER TRIM

THE BOEING COMPANY
DIRECTIONAL CONTROL
RUDDER TRAVEL LIMITS

THE BOEING COMPANY

CALC  FOSTER  12-9-67
CHECK  HOLTZNER  1-5-68
APR
INK  ODEGARD  12-9-67

REVISED  DATE
KUPCIS  6-29-68
LOW  1-29-70

747
D6-30643
Vol. II
PAGE 10.1-6
10.2 Rudder Blowdown

Rudder blowdown is not included in NASA Simulation.

The rudders are actuator force limited below the mechanically available rudder under some flight conditions. To determine these limits use the following equations:

\[ HM \cos \delta_{RU} = 855.2 \cdot C_{HR} \cdot q \]
\[ HM \cos \delta_{RL} = 833.2 \cdot C_{HR} \cdot q \]

where:

\[ HM = \pm 13,800 \text{ lb-ft. for each rudder with all hydraulic systems operative.} \]

This value is reduced by one-half if one of the two hydraulic systems on either rudder becomes inoperative.

The hinge moment coefficient can be calculated for any flight condition using the data on pages 10.2-2, 10.2-3, and 10.2-4 with the following equation:

\[ C_{HR} = \frac{(C_{HR})_M}{(C_{HR})_{M=0}} \cdot (C_{HR})_{M=0} \]

Note that the hinge moments for rudders deflected separately are different from those with rudders deflected together. The rudders deflected separately curve is used only when one of the rudders is operating.
NOTE: REVERSE SIGNS FOR NEGATIVE RUDDER USE FOR ALL FLAP SETTINGS

UPPER RUDDER

\[ \left( \frac{C_{HR}}{C_{M0}} \right) M=0 \]

SIDE SLIP ANGLE, \( \alpha \sim \text{DEG} \)

25°
20°
15°
10°
5°
0°
-5°
-10°
-15°

LOWER RUDDER

\[ \left( \frac{C_{HR}}{C_{M0}} \right) M=0 \]

SIDE SLIP ANGLE, \( \alpha \sim \text{DEG} \)

25°
20°
15°
10°
5°
0°
-5°
-10°
-15°

DIRECTIONAL CONTROL RUDDER HINGE MOMENTS SEGMENTS DEFLECTED TOGETHER

THE BOEING COMPANY

CHECK
FOSTER 10-21-67
HOLTZNER 9-24-68
CURNUTT 10-29-68

APR
LAGREE 2-10-70
BECK 6-29-70

INK
KINSMA 2-16-70

747
DB-301343
Vol. II
PAGE 10-2-2
Reverse signs for negative rudder.

Use for all flap settings.

These data not included in NASA simulation.

Upper Rudder

$$C_{HR(j)} M = 0$$

Lower Rudder

$$C_{HR(e)} M = 0$$
10.3 Yaw Damper/Turn Coordinator

Dutch roll oscillations on the 747 are attenuated by a series yaw damper which commands rudder proportional to yaw rate and bank angle. The yaw damper output signals are fed through shaping networks into the rudder actuator package, which drives the rudder. These rudder deflections do not result in any movement of the rudder pedals, nor do they affect normal operation of the rudder. The yaw damper is dualized in that both the upper and lower rudders have independent yaw damping systems. Deflection is limited to 3.6 degrees and the rate of deflection (for yaw damping) cannot exceed 15 degrees per second. A complete description of the yaw damper system is shown in the block diagram on page 10.3-2.

Turn coordination is achieved by deflecting the rudder, through the yaw damper actuator, proportional to roll rate. The input roll rate signal is actually a derived rate, as shown in the block diagram. The turn coordinator operates only when the flaps are down, having a gain of .69 degrees/degree per second.

An "easy-on" circuit has been incorporated with the flap switch to smooth transients in the bank attitude signals when the flap switch activates. (A warning light in the cockpit is provided to indicate improper operation of the flap switch).
HIGH LIFT SYSTEM

A general description of the system is presented in the Introduction on Pages 1.2-5 and 1.2-6 and in Volume I. A block diagram of the simulated high lift model is shown on Page 11.0-2. The data for each particular block can be found on the page number adjacent to the block.

A flap auto-retraction system is designed as part of the flap system. With the flap lever in the 30 detent and with aircraft speeds exceeding 169 knots, the flaps automatically drive back to the 25 position. The flaps automatically return to the original 30° setting when the airspeed is decreased to 164 knots.
FLAP SCREW TRAVEL RATE $= \frac{dFST}{dt} = \pm 2.2 \text{ deg/sec}$

HIGH LIFT MODEL
12.0 PROPULSION SYSTEM

A general description of the system is presented in the Introduction on Pages 1.3-2 and 1.3-3 and in Volume I.

12.1 Engine Pressure Ratio

A block diagram of the engine pressure ratio simulation is shown on Page 12.1-2. The flight idle limit, occurring when the flap position is 25 or 30 and gear is down but not on the ground, was simulated by adding an incremental EPR of 0.015.

The NASA engine transient program, consisting of a transport delay and a lag, was modified to approximate the 747 engine dynamics shown on Pages 12.1-6 through 12.1-17.

The engine operating limitations are contained in Appendix A - Flight Manual.
ENGINE PRESSURE RATIO SIMULATION

FLIGHT IDLE

\[ \Delta EPR = 0.015 \]  
WHEN \( \delta_F \geq 25 \),  
LANDING GEAR DOWN,  
& \( h_G > 0 \)

ACTUAL ENGINE PRESSURE RATIO EPR

NASA DIGITAL COMPUTER

ENGINE POWER LEVER ANGLE

REV. THRUST LEVER  
FORWARD THRUST LEVER

THROTTLE ANGLE  
P. 12.1-3

STATIC ENGINE PRESSURE RATIO EPR\(_{\text{STATIC}}\)

ENGINE POWER LEVER ANGLE  
P. 12.1-4

EPR\(_{\text{SPEED DECAY}}\)

\[ \Delta EPR \]

M  
P. 12.1-5

\( \delta_F \)

\( h_G \)
SNAP ACCELERATIONS
SEA LEVEL STATIC  JT9D-3
SNAP ACCELERATION - PARTIAL THROTTLE
SEA LEVEL STATIC  JT9D-3

THROTTLE POSITION ~ DEG

Δ TIME ~ SEC

REF: D6-13302
P. 6.2-3

ENGINE TRANSIENT CHARACTERISTICS

THE BOEING COMPANY
SNAP ACCELERATION - PARTIAL THROTTLE
SEA LEVEL STATIC   JT9D-3

THROTTLE POSITION ~DEG

\[ \Delta \text{TIME ~ SEC} \]

\[ \pm 50 \]

\[ \pm 40 \]

\[ \pm 30 \]

\[ \pm 20 \]

\[ \pm 10 \]

\[ \pm 0 \]

\[ 0 \]

\[ 10 \]

\[ 20 \]

\[ 30 \]

\[ 40 \]

\[ 60 \]

\[ 1.5 \]

\[ 1.4 \]

\[ 1.3 \]

\[ 1.2 \]

\[ 1.1 \]

\[ 1.0 \]

REF: D6-13302
P 6.2-4

ENGINE TRANSIENT CHARACTERISTICS
THE BOEING COMPANY

CALC
CHECK
APR
APR
DRN 1/10/69
SLOW DECELERATION - PARTIAL THROTTLE
SEA LEVEL STATIC JT9D-3
SLOW DECEL-ACCEL
30,000 FT  M=.55  JT9D-3

POWER LEVER
ANG. DISP.
~ DEG.

THRUXT LEVER = 28°

THRUXT LEVER = 1°

THRUXT LEVER = 35°

\[ \frac{P_{r7}}{P_{r1}} \]

\( \Delta \text{TIME} \sim \text{SEC} \)

REF: DG-13302
P. 6.2-11
MEDIUM DECEL - SNAP ACCEL
30,000 FT  M = .55  JT9D-3

POWER LEVER ANG. DISP. ~ DEG.

THRUST LEVER = 27.5°

THRUST LEVER = 0°

THRUST LEVER = 28°

\[ \frac{P_{t7}}{P_{t1}} \]

\[ \Delta \text{TIME} \approx \text{SEC} \]

REF: D6-13302
P. 6.2-13
SNAP ACCEL - DECEL
30,000 FT  M = .85  JT9D-3

POWER LEVER
ANG. DISP.  ~ DEG.

THRUSt LEVER = 29°

THRUSt LEVER = 0

\( \frac{P_{T7}}{P_{T1}} \)

\( \Delta \text{TIME} \sim \text{SEC} \)

REF: DG-13302
P. 6.2-15
SLOW DECEL - MEDIUM ACCEL
30,000 FT M=.85 JT9D-3

POWER LEVER ANG. DISP.
~ DEG

THrust LEVER = 30°

THrust LEVER = 0°

PT7/PT1

Δ TIME ~ SEC

REF: DG-13302
P. 6.2-17
ENGINE TRANSIENT CHARACTERISTICS

THE BOEING COMPANY
12.2 Engine Thrust

A block diagram of the engine thrust simulation is shown on Page 12.2-2.

The reverse thrust characteristics used in the simulation were a function of ambient pressure and EPR, Page 12.2-5. This data was based on a Mach number of .2. Reverse thrust data as a function of Mach number is shown on Pages 12.2-6, 12.2-7 and 12.2-8.
REVERSE NET THRUST  
JT9D-3 ENGINES  
PRIMARLY REVERSER  

MACH 6
4
2
0

ENGINE PRESSURE RATIO ~ Pt/Pt,
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0

THESE DATA NOT INCLUDED  
IN NASA SIMULATION  

CORRECTED NET THRUST ~ (F'N/ft²)*Rev ~ 1000 LBS

CALC CHECK APR
JVS EDK 12/24/69

747
DX-30643 Vol. II
THE BOEING COMPANY
12.2-8
PARAMETERS FOR 747 ENGINE GAUGE DISPLAY
12.3 Engine Cockpit Display Parameters

A block diagram for engine display parameters is shown on Page 12.3-2 and the block diagram for the NASA engine gauge simulation is shown on Page 12.3-3. The compressor RPM was scaled up by a factor of 8.8 to obtain the exhaust gas temperature in the simulation. The table on Page 12.3-4 shows the accuracy and the data points used in determining the scale factor.
APPROXIMATIONS FOR NASA ENGINE GAUGE DISPLAY

\[ K_{\text{EGT}} = 8.8 \]

\[ \frac{\%N_1}{\sqrt{\theta t_2}} \]

\[ \%N_1 \text{ GAUGE} \]

\[ \sqrt{\theta t_2} \]

\[ \% \text{COMPRESSOR RPM} \]

\[ M \]

\[ \text{EPR GAUGE} \]

\[ \text{EGT GAUGE} \]

\[ \text{EPR} \]

\[ \text{NASA DIGITAL COMPUTER} \]

\[ \text{CAB} \]
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<th>EPR</th>
<th>M</th>
<th>θ₁₂</th>
<th>(\sqrt{\theta_{12}})</th>
<th>(\frac{N_1}{\sqrt{\theta_{12}}}) P.123-6</th>
<th>(\frac{\theta_{12}}{\theta_{11}}) P.123-5</th>
<th>(\frac{T_{12}^0}{\theta_{11}})</th>
<th>(\frac{T_{12}^0}{\theta_{11}}) C</th>
<th>(\frac{\text{SF}}{\theta_{12}^0\text{C}})</th>
<th>(\frac{\theta_{12}^0\text{C}}{N_1})</th>
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<td>792</td>
<td>8.6</td>
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</table>

\(\theta_{12} = 1 + 2M^2\)

\(\theta_{11} \approx \theta_{12}\)

\(\text{SF} = \frac{T_{12}^0\text{C}}{N_1} \approx 8.8 \frac{\text{C}}{N_1}\)
12.4 Windmilling Drag

The windmilling drag characteristics on Page 12.4-2 were included in the simulation as a term in the engine equations.
For Flight Simulator only

JT9D-3 TURBOFAN ENGINE
WINDMILLING DRAG
REF: P&WA CURVE NO. 35819

CORRECTED WINDMILLING DRAG ~ Fwd
5 km ~ LBS.
4000
3000
2000
1000
0
0
0.2
0.4
0.6
0.8
1.0
1.2
MACH NO.

CALC
JMM 2/10/7
CHECK
APR 9/30/7
REVISIONS
APR

WINDMILLING DRAG
JT9D 3 TURBOFAN ENGINE
THE BOEING COMPANY

PAGE 12.4-2
Thrust Reverser Effects on Aerodynamic Coefficients

The thrust reverser effects on the lift, drag and pitching moment coefficients are presented on Page 12.5-2. The approximations used in the simulation were made to conserve computer storage and justified because the thrust reversers are recommended to be in the idle reverse position by approximately 60 knots.
DETERMINATION OF OLEO STRUT COMPRESSION
AND COMPRESSION RATE

\[ \Delta S_{Ti} = h - h_R + X_{Li} \sin \theta_B - Y_{Li} \sin \phi_B \cos \theta_B - Z_{Li} \cos \phi_B \cos \theta_B \]
\[ \Delta \dot{S}_{Ti} = \dot{h} + X_{Li} \cos \theta_B \dot{\theta}_B + Y_{Li} \left( \sin \phi_B \sin \phi_B \dot{\theta}_B - \cos \phi_B \cos \theta_B \dot{\phi}_B \right) + Z_{Li} \left( \sin \theta_B \cos \phi_B \dot{\phi}_B \right) + \sin \phi_B \cos \theta_B \dot{\phi}_B \]

\[ \Delta S_{T2} \quad \Delta S_{T1} \quad \Delta S_{T3} \]
\[ \Delta \dot{S}_{T2} \quad \Delta \dot{S}_{T1} \quad \Delta \dot{S}_{T3} \]

NOTE: LANDING GEAR DOES NOT CONTACT THE RUNWAY UNLESS \( \Delta S_{Ti} < 0 \)
\( \Delta S_{Ti} \) IS NEGATIVE FOR STRUT COMPRESSION

DETERMINATION OF OLEO STRUT COMPRESSION AND COMPRESSION RATE
LANDING GEAR

A general description of system and landing gear equations is included in Volume I of this report. A detailed derivation of the landing gear equations is contained in the Appendix to Volume I.

Block diagrams showing the method of calculating strut compression and force and wheel side and drag forces in the NASA simulation are shown on Pages 13.0-2 through 13.0-5. Refer to Volume I for the landing gear nomenclature.
DETERMINATION OF OLEO STRUT FORCES

\[2 \times c_i \times \Delta s_{Ti} \times |\Delta s_{Ti}|\]

\(\Delta s_{Ti}\) and \(F_{Gz_i}\) are negative for compression.
NOTE: STEERING ANGLE δs IS USED FOR NOSE WHEEL EQUATIONS ONLY

10° STEERING, δs, FOR FULL PEDAL DEFLECTION, δp, 12.2°.

\[ \frac{\partial \delta s}{\partial \delta p} = 0.82 \]

\[ (\delta T_i)^2 \rightarrow H_{T_i} \]

\[ K_{T_i} = \frac{0.093}{2(1000)} \text{ IN/LB} \]

\[ K_{T_{2,3}} = \frac{0.077}{8(1000)} \text{ IN/LB} \]

\[ H_{T_i} = (2)(73730)(0.0042) = 619 \]

\[ H_{T_{2,3}} = 8(73730)(0.0042) = 2477 \]

\[ G_{T_i} = 2(73730)(0.026) = 3834 \]

\[ G_{T_{2,3}} = 8(73730)(0.026) = 15336 \]

DETERMINATION OF WHEEL SIDE FORCE
NASA DIGITAL COMPUTER

\[ F_{B_i} = 2 \left[ \frac{.263 \cdot W \cdot \delta_{B_i}}{g} \right] \]

\[ (F_{B_i})_{MAX} = 2 \left( .834 + 4.167 \cdot \mu \right) \cdot \frac{W}{g} \]

Maximum deceleration = 5 ft/sec^2 for maximum braking on each main gear
NOTE: FORCE DUE TO DAMPING FOR LEFT OR RIGHT MAIN GEAR = FORCE BASED ON MAIN GEAR CURVE TIMES 2.

\[
\text{FORCE DUE TO DAMPING} = 2 \times C \times (\text{GEAR}) \times (\text{VELO})
\]

LEAF OR RIGHT MAIN GEAR

NASA SIMULATION COMBINES WING, AND BODY GEAR INTO AN EQUIVALENT MAIN GEAR

MAIN GEAR

NOSE GEAR

TIRE + OLEO DEFLECTION / INCHES (COMPRESSION)

REF: D6-30437 P. 294

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REV LTR:

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<td>13.0 - 6</td>
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</table>
NOTE: FORCE DUE TO STRUT DEFLECTION FOR LEFT OR RIGHT MAIN GEAR = FORCE BASED ON MAIN GEAR CURVE TIMES 2.

NASA SIMULATION COMBINES WING AND BODY GEAR INTO AN EQUIVALENT MAIN GEAR.

REFER TO: DG-30437

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| LANDING GEAR AIR CURVE | 747 |

- | - | - | - |

BOEING | NO. DG-30437 |
| SECT | PAGE 13.0-7 |
14.1.1 COCKPIT INSTRUMENTS

The cockpit instruments were statically checked to assure that the piloted checkout could be conducted with confidence. Rudimentary checks were made of the following instruments:

1. Altimeter
2. Rate-of-climb-indicator
3. Airspeed indicator
4. Attitude indicator
5. Turn and bank indicator
6. Mach meter
7. Flap position indicator
8. Stabilizer position indicator

The checks were conducted by putting the computer in "hold" and comparing the instrument readings with the computer values. These comparisons were made a number of times throughout the simulation checkout and all of the instruments were in good agreement with the computed values. The instruments operated smoothly during the dynamic checks and piloted evaluation.

A comparison of digital values and cockpit instrument readings were tabulated during the four engine climb performance test. The comparison between the computed and indicated pitch attitude shows the largest discrepancy. However, the pitch attitude is difficult to read to a fraction of a degree. The differences are within the tolerances specified in the table on Page 14.2-10.
The stabilizer position indicator in the FSAA was programmed using stabilizer referenced to the fuselage reference line, $\Delta_{\text{FRL}}$. All of the aerodynamic data are in terms of $\Delta_{\text{FRL}}$. The 747 stabilizer indicator and stabilizer information used in the flight manual and D6-30833-1 is in stabilizer "pilot's units", $\Delta_p$. The conversion is given by:

$$\Delta_p = 3^\circ - \Delta_{\text{FRL}}.$$
14.1.2 ATMOSPHERE MODEL

The atmosphere model, presented in Volume I of this report, was checked for a number of altitude and airspeed conditions. Values of $V_e$ for a number of altitudes were input to the computer. Output values of $M$, $V_c(V_T)$, $V$, $q$, and $q_c$ were checked with the data in the table on Page 14.1-5.
### ATMOSPHERE CHECK

1. Standard Day, No Wind
2. Pilot's Airspeed \( V_I \) is equal to \( V_C \)

<table>
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<th>ALT. FT</th>
<th>( V_I ) KTS</th>
<th>( V_e ) KTS</th>
<th>( V_{TRUE \ FT/SEC} )</th>
<th>MACH NO.</th>
<th>( q ) LB/FT(^2)</th>
<th>( q_C ) LB/FT(^2)</th>
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<td>100</td>
<td>100</td>
<td>169</td>
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14.1.3 ENGINE CHARACTERISTICS

Static forward and reverse thrust characteristics were checked by comparing the computer output data with a manual calculation of the same condition. The comparisons are shown in the examples on Pages 14.1-7 and -8.

The transient engine characteristics are shown on Pages 14.1-9 and -10. A sea level static condition was set up and throttle changes were commanded. The resulting transient characteristics for EPR approximate the engine transient data shown in Section 12.
# Static Thrust

**Altitude:** 200 ft  
**M:** 0.2  
**Standard Day**

## Maximum Forward Thrust

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<th>Manual Calculation</th>
<th>Digital Output</th>
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<td>60.80°</td>
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<td>( \Delta \theta )</td>
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<td>-0.0361</td>
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<tr>
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<td>( F_{n}/s )</td>
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<tr>
<td>( s )</td>
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<tr>
<td>( F_{n} )</td>
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<td>90.3%</td>
</tr>
<tr>
<td>( N_{i} / \sqrt{\theta_{i}^{2}} )</td>
<td>93.6%</td>
<td></td>
</tr>
<tr>
<td>( N_{i} )</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

## Idle Forward Thrust

<table>
<thead>
<tr>
<th>Forward Thrust Lever Angle (Input)</th>
<th>Manual Calculation</th>
<th>Digital Output</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power Lever Angle</td>
<td></td>
<td></td>
</tr>
<tr>
<td>( \theta )</td>
<td>0°</td>
<td>19°</td>
</tr>
<tr>
<td>57.5°</td>
<td>57.75°</td>
<td></td>
</tr>
<tr>
<td>( \Delta \theta )</td>
<td></td>
<td></td>
</tr>
<tr>
<td>EPR (( f(\text{Power Lever Angle}) ))</td>
<td>1.02</td>
<td></td>
</tr>
<tr>
<td>( \Delta \text{EPR} )</td>
<td>-0.036</td>
<td>-0.0361</td>
</tr>
<tr>
<td>EPR (Final)</td>
<td>0.984</td>
<td>0.9839</td>
</tr>
<tr>
<td>( F_{n}/s )</td>
<td>1250 lb</td>
<td>1264 lb</td>
</tr>
<tr>
<td>( F_{n} )</td>
<td>1241 lb</td>
<td></td>
</tr>
<tr>
<td>( N_{i} / \sqrt{\theta_{i}^{2}} )</td>
<td>31%</td>
<td>43.7%</td>
</tr>
<tr>
<td>( N_{i} )</td>
<td>31.1%</td>
<td></td>
</tr>
</tbody>
</table>
## Idle Reverse Thrust

<table>
<thead>
<tr>
<th>REVERSE THRUST LEVER ANGLE (INPUT)</th>
<th>MANUAL CALCULATION</th>
<th>DIGITAL OUTPUT</th>
</tr>
</thead>
<tbody>
<tr>
<td>POWER LEVER ANGLE</td>
<td>-34°</td>
<td>-31.07°</td>
</tr>
<tr>
<td>EPR (f(Power Lever Angle))</td>
<td>45°</td>
<td>46.46°</td>
</tr>
<tr>
<td>ΔEPR</td>
<td>1.02</td>
<td>1.036</td>
</tr>
<tr>
<td>EPR (FINAL)</td>
<td>-0.036</td>
<td>-0.0361</td>
</tr>
<tr>
<td>(F_N / S)_REVERSE</td>
<td>.984</td>
<td>.9839</td>
</tr>
<tr>
<td>F_N REVERSE</td>
<td>4400 lb</td>
<td>4450 lb</td>
</tr>
<tr>
<td>N_1 / (GT²)</td>
<td>-4369 lb</td>
<td>-4422 lb</td>
</tr>
<tr>
<td>N_1</td>
<td>31.9%</td>
<td>42.7%</td>
</tr>
</tbody>
</table>

## Maximum Reverse Thrust

<table>
<thead>
<tr>
<th>REVERSE THRUST LEVER ANGLE (INPUT)</th>
<th>MANUAL CALCULATION</th>
<th>DIGITAL OUTPUT</th>
</tr>
</thead>
<tbody>
<tr>
<td>POWER LEVER ANGLE</td>
<td>-107°</td>
<td>-108.40°</td>
</tr>
<tr>
<td>EPR (f(Power Lever Angle))</td>
<td>3°</td>
<td>3°</td>
</tr>
<tr>
<td>ΔEPR</td>
<td>1.615</td>
<td>1.636</td>
</tr>
<tr>
<td>EPR (FINAL)</td>
<td>-0.036</td>
<td>-0.0361</td>
</tr>
<tr>
<td>(F_N / S)_REVERSE</td>
<td>1.579</td>
<td>1.5826</td>
</tr>
<tr>
<td>F_N REVERSE</td>
<td>34700 lb</td>
<td>34646 lb</td>
</tr>
<tr>
<td>N_1 / (GT²)</td>
<td>-34454 lb</td>
<td>-34428 lb</td>
</tr>
<tr>
<td>N_1</td>
<td>97.4%</td>
<td>95.06%</td>
</tr>
</tbody>
</table>

---

## Static Thrust

### Boeing

**DG-30643**

**Vol. II**

**SECT**

**PAGE** 14.1-8
SEA LEVEL, STANDARD DAY, STATIC
IDLE TO MAXIMUM THRUST
SNAP ACCELERATION - DECELERATION

ENGINE TRANSIENTS
SEA LEVEL, STANDAD DAY, STATIC
EPR AT T=0 = 1.08
SNAP ACCELERATION - DECELERATION

ENGINE TRANSIENTS
14.1.4  LONGITUDINAL TRIM

The simulated airplane was statically trimmed at several variations of weight, c.g., altitude, speed, and flap positions. Computed values of \( A \), \( B \), and \( EPR/F_n \) were compared to Boeing simulator and flight test results. Comparisons to simulator results are tabulated on Pages 14.1-12 and -13 and plotted on Pages 14.1-14 thru -18. Comparisons to flight test and simulator results are plotted on Pages 14.1-19 thru -21.

The effects of configuration changes, landing gear up and down and speed brakes up and down, were computed and the results are tabulated on Page 14.1-22.

The effect of ground effect on trim is tabulated on Page 14.1-12. The trim data are based on the ground effect curves on Pages 2.0-31, -32, 3.0-17, -18 and 4.0-35.
### GROUND EFFECT

<table>
<thead>
<tr>
<th>GEAR</th>
<th>ALT. ~ FT</th>
<th>$\Delta \rho$ UNITS</th>
<th>$\theta_b$ ~ DEG</th>
<th>EPR</th>
<th>THRUST ~ LB</th>
</tr>
</thead>
<tbody>
<tr>
<td>BOEING NASA</td>
<td>100</td>
<td>4.89</td>
<td>5.0</td>
<td>1.190</td>
<td>76570</td>
</tr>
<tr>
<td>BOEING NASA</td>
<td>30</td>
<td>6.55</td>
<td>3.7</td>
<td>1.139</td>
<td>60750</td>
</tr>
<tr>
<td>BOEING NASA</td>
<td>10</td>
<td>7.11</td>
<td>3.3</td>
<td>1.106</td>
<td>50650</td>
</tr>
</tbody>
</table>

564,000 LB
C.G. = 33% $V_2$ = 142 KT
FLAPS 30 GEAR DOWN

### Longitudinal Trim

<table>
<thead>
<tr>
<th>CALC</th>
<th>DRN</th>
<th>MAY</th>
<th>REVISED DATE</th>
</tr>
</thead>
<tbody>
<tr>
<td>CHECK</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>APPD</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>APPD</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>COND. NO</td>
<td>FLAP POSITION</td>
<td>GEAR</td>
<td>G.M. 2000 LB</td>
</tr>
<tr>
<td>----------</td>
<td>---------------</td>
<td>------</td>
<td>-------------</td>
</tr>
<tr>
<td>4.0.1</td>
<td>UP</td>
<td>UP</td>
<td>650</td>
</tr>
<tr>
<td>4.0.2</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>4.0.3</td>
<td>UP</td>
<td>UP</td>
<td>500</td>
</tr>
<tr>
<td>4.0.4</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>4.0.5</td>
<td>UP</td>
<td>UP</td>
<td>500</td>
</tr>
<tr>
<td>4.0.6</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>4.0.7</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>4.0.8</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>4.0.9</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

**LONGITUDINAL TRIM**

**COMPUTATION TOLERANCE**

- NASA VALUES ARE FROM COMPUTER OUTPUT

<table>
<thead>
<tr>
<th></th>
<th>± .5 UNITS</th>
<th>± .5 DEG.</th>
<th>± .01</th>
<th>± 3%</th>
<th>± .25°</th>
</tr>
</thead>
<tbody>
<tr>
<td>Boeing</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>NASA</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
LONGITUDINAL TRIM
ALT. = 35000 FT.
THE BOEING COMPANY
<table>
<thead>
<tr>
<th>SYM</th>
<th>A/P</th>
<th>FLT</th>
<th>C.G.-%MAC</th>
<th>G.W.-1000LB</th>
<th>ALT.-1000FT</th>
</tr>
</thead>
<tbody>
<tr>
<td>☐</td>
<td>ROMD</td>
<td>.30-2</td>
<td>14</td>
<td>.537-.539</td>
<td>34; 35</td>
</tr>
<tr>
<td>☑</td>
<td>31-6</td>
<td>32</td>
<td>.542-.541</td>
<td>34; 35</td>
<td></td>
</tr>
<tr>
<td>☐</td>
<td>18-1</td>
<td>22</td>
<td>.506-.492</td>
<td>35</td>
<td></td>
</tr>
</tbody>
</table>

**SIMULATOR**

(06-20423, REV.D)

**NASA**

**PILOT UNITS**

---

**CALC BY STROM 3-18-70**

**REvised DATE**

**CORNUT 5-20-70**

**THE BOEING COMPANY**

**14.1-20**
# Table 6.5 Configuration Changes

<table>
<thead>
<tr>
<th>Test</th>
<th>Flap Position</th>
<th>Gear</th>
<th>G. M. 1000 Lb.</th>
<th>C. G. 1000 Lb.</th>
<th>Altitude 1000 Ft.</th>
<th>V_I/M</th>
<th>Speed</th>
<th>Δ P Units</th>
<th>F_s ~ Lb.</th>
<th>Δ e ~ Deg.</th>
<th>Δ B ~ Deg.</th>
<th>Y ~ Deg.</th>
<th>R/C ~ Ft/Min</th>
<th>IEPR</th>
<th>F_N Total ~ Lb.</th>
</tr>
</thead>
<tbody>
<tr>
<td>Boeing</td>
<td>Up</td>
<td>Up</td>
<td>564</td>
<td>25</td>
<td>5</td>
<td>150</td>
<td>Zero</td>
<td>6.3</td>
<td>6.38</td>
<td>0</td>
<td>2.0</td>
<td>3.5</td>
<td>0</td>
<td>0</td>
<td>1.225</td>
</tr>
<tr>
<td>Boeing // NASA</td>
<td>Dn</td>
<td>Up</td>
<td>564</td>
<td>25</td>
<td>5</td>
<td>150</td>
<td>Zero</td>
<td>6.6</td>
<td>6.72</td>
<td>0</td>
<td>2.0</td>
<td>3.8</td>
<td>0</td>
<td>0</td>
<td>1.250</td>
</tr>
<tr>
<td>Boeing // NASA</td>
<td>Up</td>
<td>Up</td>
<td>564</td>
<td>32</td>
<td>5</td>
<td>250</td>
<td>Zero</td>
<td>3.1</td>
<td>3.1</td>
<td>0</td>
<td>2.0</td>
<td>3.9</td>
<td>0</td>
<td>0</td>
<td>1.035</td>
</tr>
<tr>
<td>Boeing // NASA</td>
<td>Up</td>
<td>Dn</td>
<td>564</td>
<td>32</td>
<td>5</td>
<td>250</td>
<td>Zero</td>
<td>2.5</td>
<td>2.49</td>
<td>0</td>
<td>2.0</td>
<td>3.2</td>
<td>0</td>
<td>0</td>
<td>1.162</td>
</tr>
<tr>
<td>Boeing // NASA</td>
<td>Up</td>
<td>Up</td>
<td>564</td>
<td>32</td>
<td>5</td>
<td>270</td>
<td>Zero</td>
<td>2.8</td>
<td>2.82</td>
<td>0</td>
<td>2.0</td>
<td>3.0</td>
<td>0</td>
<td>0</td>
<td>1.028</td>
</tr>
<tr>
<td>Boeing // NASA</td>
<td>Up</td>
<td>Up</td>
<td>564</td>
<td>32</td>
<td>5</td>
<td>270</td>
<td>Infl. Detent</td>
<td>2.5</td>
<td>2.49</td>
<td>0</td>
<td>2.0</td>
<td>4.2</td>
<td>0</td>
<td>0</td>
<td>1.093</td>
</tr>
</tbody>
</table>

**Computation Tolerance**

<table>
<thead>
<tr>
<th></th>
<th>±</th>
<th>±10%</th>
<th>±5</th>
<th>±3</th>
<th>±0.1</th>
<th>±3%</th>
</tr>
</thead>
<tbody>
<tr>
<td>Boeing // NASA</td>
<td>1</td>
<td>1.25</td>
<td>0.5</td>
<td>0.3</td>
<td>0.01</td>
<td>0.3</td>
</tr>
</tbody>
</table>
14.1.5 ELEVATOR-STABILIZER TRADES

The simulated airplane was initially trimmed for 1 g straight and level flight with zero column deflection. Values of $S_{col}$ were input and the airplane was retrimmed with stabilizer. The results are tabulated and plotted with Boeing simulation and flight test results on Page 14.1-24. This test provided a check on the column to elevator gearing, outboard elevator blowdown, and elevator and stabilizer effectiveness.
**NOTE**

1. FLAGGED SYMBOLS INDICATE OUTBD. ELEV. BLOWN DOWN
2. $\Delta \alpha_p$ FROM TRIM COUNTER

**FLT. 73-5**
- $M = .74 -.76$
- G.W. = 628000 LB.
- C.G. = 11.7% MAC
- $\Delta \alpha_{trim} = 5.25$ UNITS

**FLT. 73-11**
- $M = .74 -.76$
- G.W. = 604000 LB.
- C.G. = 12.1% MAC
- $\Delta \alpha_{trim} = 5.12$ UNITS

**SIMULATOR (D6-20423, REV.D DATA)**
- $M = .75$
- G.W. = 628000 LB.
- C.G. = 11.7% MAC
- $\Delta \alpha_{trim} = 5.4$ UNITS

<table>
<thead>
<tr>
<th>$\Delta \alpha_p$</th>
<th>$\delta_e$</th>
<th>$\delta_{col}$</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>UNITS</strong></td>
<td><strong>DEG.</strong></td>
<td><strong>DEG.</strong></td>
</tr>
<tr>
<td>Boeing Target</td>
<td>NASA Test</td>
<td>Boeing Target</td>
</tr>
<tr>
<td>5.4</td>
<td>5.35</td>
<td>2.0</td>
</tr>
<tr>
<td>4.4</td>
<td>4.30</td>
<td>-.1</td>
</tr>
<tr>
<td>3.4</td>
<td>3.36</td>
<td>-2.3</td>
</tr>
<tr>
<td>7.4</td>
<td>7.24</td>
<td>6.2</td>
</tr>
<tr>
<td>9.4</td>
<td>9.32</td>
<td>10.8</td>
</tr>
</tbody>
</table>

**TOLERANCE**
- $\pm 5^\circ$ | $\pm 1^\circ$ | 747

---

**CALC**

**CHECK**

**APPD**

**APPD**

**REV LTR:**

**STABILIZER - ELEVATOR TRADES**

**SIMULATOR - FLIGHT TEST COMPARISON**

**BOEING**

**NO.**

**VOL II**

**SECT**

**PAGE** 14.1-24
14.1.6 AIRPLANE DYNAMICS

Dynamic responses were run to check the Dutch roll, short period, and phugoid modes of the simulated airplane.

14.1.6.1 Dutch Roll

The Dutch roll was initiated by releasing the airplane from an initial condition of sideslip. Responses were made with yaw damper on and off. Comparisons of the time histories between the NASA and Boeing simulations are shown on Pages 14.1-26 thru -36.

14.1.6.2 Short Period and Phugoid

The short period and phugoid modes were exited by an elevator input after the airplane had been trimmed for straight and level flight. An incremental elevator was input for a prescribed time and then removed. Comparisons of the time histories between the NASA and Boeing simulations are shown on Pages 14.1-37 thru -49.
**NASA**

**YAW DAMPER OFF**

\[ \beta \]

\[ \phi \]

- \( \beta_{16} = 5^\circ \)
- (NASA DAC LIMITED TO)
- (\( \beta = 4^\circ \) FOR TEST)

**BOEING**

**GW = 564,000 LB**

- \( h = 20,000 \text{ FT} \)
- \( V_e = 325 \text{ KT} \)
- C.G. = 25%

**NASA-BOEING DUTCH ROLL COMPARISON**
NASA

YAW DAMPER OFF

β

5°

0

-5°

10°

0

-10°

Φ

GW: 564,000 LB
h = 20,000 FT
Ve = 225 KT
C.G. = 25%

NASA DAC LIMITED TO
β = 4° FOR TEST

BOEING

5°

0

-5°

10°

0

-10°

Φ

NASA-BOEING DUTCH ROLL COMPARISON
GW = 564,000 lb
h = 5,000 ft  Ve = 250 KT
C.g. = 25%

β₁C = 5°
NASA DAC LIMITED TO
β = 4° FOR TEST
\textbf{NASA}

\textit{YAW DAMPER OFF}

$\theta$ vs. Time

$\theta$: 4°

$-4°$ to $4°$

$0°$ to $10°$

$-10°$

\textbf{BOEING}

$GW = 564,000 \text{ LB}$

$h = 200 \text{ FT}$

$V_e = 142 \text{ KT}$

C.G.: 25%

$\delta_r = 30$

\textbf{NASA-BOEING DUTCH ROLL COMPARISON}
$g_w = 564,000 \text{ LB}$
$h = 35,000 \text{ FT}$
$V_e = 275 \text{ KT}$
$c.g. = 25\%$

**NASA - BOEING DUTCH ROLL COMPARISON**
NASA

**NASA-BOEING DUTCH ROLL COMPARISON**

GW = 710,000 LB
h = 200 FT  Ve = 180 KT
c.g. = 25%  δF = 10
NASA

YAW DAMPER OFF

$6W = 564,000 \text{ LB}$
$h = 200 \text{ FT}$
$V_e = 180 \text{ KT}$
$c.g. = 25\%$
$\delta_F = 30$

BOEING

NASA-BOEING DUTCH ROLL COMPARISON
NASA

YAW DAMPER ON

4°
3

0
-3
-4°

10

0
-10

4

0

5°

h

4°

0

-4°

10

0
-10

5°

h

4°

0

-4°

NASA—BOEING DUTCH ROLL COMPARISON

GW = 564,000 lb
h = 200 ft  Ve = 180 kt
cg. = 25%  δ F = 30
NASA

YAW DAMPER ON

5 sec

\[ \begin{align*}
\beta &= 4^\circ \\
\phi &= 0 \\
\theta &= 0 \\
\delta_e &= 0 \\
\end{align*} \]

BOEING

\[ \begin{align*}
\beta &= 4^\circ \\
\phi &= 0 \\
\theta &= 0 \\
\delta_e &= 0 \\
\end{align*} \]

GW = 564,000 LB
h = 200 FT  Ve = 142 KT
C.g. = 25%  S.E. = 30

NASA-BOEING DUTCH ROLL COMPARISON
NASA

YAW DAMPER OFF

\[ \beta \]

\[ \phi \]

GW = 564,000 LB  
\( h = 35,000 \) FT  
\( V_e = 225 \) KT  
C.g. = 25%  

NASA-BOEING DUTCH ROLL COMPARISON

BOEING

YAW DAMPER ON
NASA

**NASA-BOEING DUTCH ROLL COMPARISON**

**BOEING**

GW = 564 000 Lb
h = 200 FT Ve = 142 KT
c.g. = 25% $\delta_f = 30$
GEAR DOWN
\( G/W = 710,000 \text{ lb} \quad h = 5000 \text{ ft} \)
\( V_I = 180 \text{ KT} \quad c.g. = 25\% \)
\( \delta_F = 10 \quad \text{GEAR UP} \)

LONGITUDINAL DYNAMICS
NASA

LONGITUDINAL DYNAMICS

GW = 710,000 LB  h = 5000 FT
V1 = 210 KT  C.G. = 25%
δF = 10  GEAR UP

BOEING NO. 86-30613
Vol. II
SECT  PAGE 14.1-39
\[ \begin{align*}
\alpha_x & \quad 10^\circ \\
\theta_x & \quad 20^\circ \\
\eta_z & \quad 2.0 \\
\delta_E & \quad 0 \quad \text{ΔControl = C° for 2 sec} \\
\dot{z} & \quad 0 \quad 10^\circ/\text{sec} \\
\dot{g} & \quad 0 \quad 10^\circ/\text{sec}
\end{align*} \]

\[ \begin{align*}
\text{GW} = 710,000 \text{ LB} & \quad h = 5000 \text{ FT} \\
V_x = 180 \text{ KT} & \quad c.g. = 25\% \\
\delta_F = 10 & \quad \text{GEAR UP}
\end{align*} \]

**LONGITUDINAL DYNAMICS**
GW = 710,000 lb  h = 5000 ft
V_I = 210 KT  c.g. = 25%
δ_f = 10  GEAR UP
LONGITUDINAL DYNAMICS

GW = 564,000 lb  h = 5000 ft
V_T = 153 KT  C.G. = 33 %
\delta_F = 3.0  GEAR DOWN
NASA

\[ 2.0 \]
\[ 20' \]
\[ 0' \]
\[ 20' \]
\[ 4.0 \]
\[ 200 \]
\[ -20' \]
\[ 20' \]
\[ -10'/sec \]
\[ 10'/sec \]
\[ \Delta SE = -17.05^\circ \text{ FOR 2 SEC} \]

\[ \text{GW} = 564,000 \text{ LB} \]
\[ V_1 = 153 \text{ KT} \]
\[ c.g. = 33\% \]
\[ \delta_F = 30 \text{ GEAR DOWN} \]

LONGITUDINAL DYNAMICS

BOEING

NO. 30643

SECT PAGE 14.1-42
GW = 564,000 LB  h = 5000 FT
\( V_L = 153 \text{ KT} \)  c.g. = 15%
\( \delta_F = 30 \) GEAR DOWN

\( \Delta \delta_E = -17.05^\circ \) FOR 2 SEC

LONGITUDINAL DYNAMICS
NASA

\[ \alpha \]

\[ \theta \]

\[ h \]

\[ n_z \]

\[ V_{c_{KT}} \]

\[ \delta_e \]

\[ q \]

\[ \Delta \delta_e = -17.05^\circ \text{ for 2 sec} \]

G.W. = 564,000 LB  h = 5000 FT

\( V_1 = 153 \text{ KT} \)  c.g. = 15%

\( \delta_F = 30 \text{ GEAR DOWN} \)

LONGITUDINAL DYNAMICS

BOEING NO. D6-30643
SECT PAGE 14.1-44
\[ GW = 564,000 \text{ lb} \quad h = 20,000 \text{ ft} \]
\[ M = 0.65 \quad c.g. = 14\% \]
$GW = 564,000 \text{ LB } \quad h = 20,000 \text{ FT}$

$M = 0.65 \quad c.g. = 32\%$

$\Delta \delta_e = -3.6^\circ \text{ for } 2 \text{ sec}$

LONGITUDINAL DYNAMICS
$GW = 564,000 \text{ lb}$  $h = 35,000 \text{ ft}$  
$M = .87$  $C.g. = 32\%$

**LONGITUDINAL DYNAMICS**
\[ \text{GW} = 564000 \text{lb} \quad h = 35000 \text{ ft} \]
\[ M = 0.87 \quad C.g. = 32\% \]

**LONGITUDINAL DYNAMICS**
14.1.7 FLIGHT CONTROLS

14.1.7.1 Force and Displacement

The force and displacement curves for the wheel and rudder are shown on Pages 14.1-52 and 14.1-53. These curves were obtained from the control loader in the FSAA cab. The wheel force in the cab should be compared to the force characteristics on Page 9.1-3. The rudder force in the cab should be compared to the force characteristics on Page 10.1-3. Wheel and rudder forces were not calculated in the computer.

The column force was calculated in the computer and a modified force increment was input to the control loader to augment the constant stick force gradient initially set in the column control loader. Tests were made to verify the computed and cab column force.

A condition was set up in the computer to check the $S_{col}$ to $S_e$ relationship and the resulting forces for two c.g. positions. The tabulated comparisons between Boeing and NASA simulations are on Pages 14.1-54, 55. The data is plotted on Page 14.1-56. Column force and displacement data in the FSAA cab was also obtained for this condition. The control column was deflected to a position and the computed values of stick force, $S_e$, and $S_{col}$ were obtained. Column stick force and $S_{col}$ were also obtained from the column control loader through a calibrated x-y plotter. The results are tabulated on Page 14.1-55 and plotted on Page 14.1-57.

Similar computed stick force data was obtained from the elevator stabilizer trade condition. The results are tabulated on Page 14.1-54 and plotted on Page 14.1-58.
Maneuvering column force data was obtained during the piloted checkout.
# Longitudinal Control Forces

<table>
<thead>
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<th>Condition No.</th>
<th>Flap Position</th>
<th>Gear</th>
<th>C.M.</th>
<th>1000 LB.</th>
<th>C.D.</th>
<th>% MAC</th>
<th>Altitude 1000 FT.</th>
<th>M</th>
<th>( \Delta P )</th>
<th>( F_s )</th>
<th>( \delta_e )</th>
<th>( \delta_{COL} )</th>
<th>Digital Output</th>
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**Computation Tolerance**

- \( \pm 10\% \)
- \( \pm 5\% \)
- \( \pm 1\% \)

1. **Computed Values** - Includes 2.5 lb breakout force in pull direction and 3.0 lb in push direction.
2. **Graphical Results** on page 14.1 - 58
3. **Graphical Results** on page 14.1 - 56
600,000 LB  \( M = 0.8 \)  \( h = 35,000 \text{ FT} \)

\[ \text{c.g.} = 11\% \quad \Delta_{FRL} = -2.1 \text{ DEG} \]

<table>
<thead>
<tr>
<th>( \delta_{\text{COLUMN}} )</th>
<th>( \delta_{\text{COMPUTED}} )</th>
<th>( \delta_{E_1} )</th>
<th>( \delta_{E_0} )</th>
<th>( \text{FSAA STICK FORCE} )</th>
<th>( \text{COMPUTED STICK FORCE} )</th>
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\[ \text{c.g.} = 32\% \quad \Delta_{FRL} = -2 \text{ DEG} \]

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<th>( \delta_{E_1} )</th>
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</table>

1. Includes 2.5 lb breakout force in pull direction and 3.0 lb in push direction.
2. \( \delta_{\text{COLUMN FSAA}} \) \( \neq \) FSAA stick force obtained from X-Y plotter output of control loader.
3. Assume 7 in. of FSAA column deflection = 12.67 deg. of 747 column deflection.

Graphical results on page 14.1-56
747
600,000 LB
M = 0.8
h = 35,000 FT

STICK FORCE in LB

COLUMN DEFLECTION in IN

7 IN = 12.67 DEG OF 747 COLUMN DEFLECTION

COLUMN FSAA CONTROL LOADER

FORCE "DROP OUT" CORRECTED AS OF AUGUST, 1970.
NOTE
1. FLAGGED SYMBOLS INDICATE OUTB'D. ELEV. BLOWN DOWN
2. ΔAP FROM TRIM COUNTER

Δ NASA COMPUTED VALUES

FLT. 73-5
M = .74 - .76
G.W. = 628000 LB.
C.G. = 11.7% MAC
ΔP_TRIM = 5.25 UNITS

FLT. 73-11
M = .74 - .76
G.W. = 604000 LB.
C.G. = 12.1% MAC
ΔP_TRIM = 5.12 UNITS

SIMULATOR (D6-20423, REV. D DATA)
M = .75
h = 33000 FT
G.W. = 628000 LB.
C.G. = 11.7% MAC
ΔP_TRIM = 5.4 UNITS

SIM. ΔE0 BLOWDOWN

15 10 5 0 -5 -10 -15
Δ E A.K INCL. - DEG.

ELEVATOR STABILIZER TRADES

COLUMN FORCES
SIMULATOR - FLIGHT TEST COMPARISON
14.1.7.2 RIGGING

The lateral control rigging was checked at high speed. The table on Page 14.1-60 shows a comparison of Boeing and NASA simulation results for various combinations of wheel and speed brake inputs.
### LATERAL CONTROL RIGGING

For $\dot{V} = 350$ KTS, ALT. = 5000 FT, G.W. = 400000 LB.

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<th>$\delta_{\text{WHEEL}}$</th>
<th>SPEED BRKE HANDLE</th>
<th>$\delta_{\text{A}}$</th>
<th>$\delta_{\text{SP}}$</th>
<th>$\delta_{\text{SP}}$</th>
<th>$\delta_{\text{SP}}$</th>
<th>$\delta_{\text{SP}}$</th>
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<td>R.H.</td>
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<td>5/3</td>
<td>6/7</td>
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<td></td>
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<td>DEG.</td>
<td>DEG.</td>
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**COMPUTATION TOLERANCE**

$\pm 1.0 \text{ DEG.}$
14.1.7.3 **BLOWDOWN**

The blowdown for elevator and spoiler and the ratio changer limit value for rudder was verified at different times throughout the simulation checkout. The elevator stabilizer trade test provided a data point for elevator blowdown and the lateral rigging check provided spoiler blowdown data.

A further check on blowdown was a comparison of initial accelerations with full control inputs. The tabulation on Page 14.1-62 is a comparison of initial accelerations and control surface positions between Boeing and NASA simulations.
**FLIGHT CONDITION:**
- \( h = 20000 \text{ FT} \)
- \( M = 0.800 \)
- C.G. = 32.7
- \( G.W. = 564000 \text{ LB} \)
- \( Sf = 0 \)
- GEAR UP
- Trim: \( \alpha = 1.61 \), \( \alpha_{EFL} = -6.6 \), \( V_e = 358.5 \text{ KT} \)

**DIGITAL OUTPUT**

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<th>BOEING</th>
<th>NASA</th>
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<td>41.71/0</td>
</tr>
<tr>
<td>2/11 DEG</td>
<td>0/28.29</td>
<td>0/28.27</td>
<td>28.30/0</td>
<td>28.22/0</td>
</tr>
<tr>
<td>3/10 DEG</td>
<td>0/28.29</td>
<td>0/28.27</td>
<td>28.30/0</td>
<td>28.22/0</td>
</tr>
<tr>
<td>4/9 DEG</td>
<td>0/28.29</td>
<td>0/28.27</td>
<td>28.30/0</td>
<td>28.24/0</td>
</tr>
<tr>
<td>5/8 DEG</td>
<td>0/20.0</td>
<td>0/20.0</td>
<td>20.0/0</td>
<td>20.0/0</td>
</tr>
</tbody>
</table>

**COMPARISON OF INITIAL VALUES**
(t=0) FOR VARIOUS CONTROL INPUTS

**CALC:** 6/19/70 **PRN:** REVISED DATE: **NO. Vol. II**

**CHECK** **APPD** **APPD**

**SECT** **PAGE:** 14.1 - C2
14.1.8 LANDING GEAR

The landing gear was static checked by allowing the airplane to settle on the ground and verifying the amount of strut compression. Dynamic responses were made by releasing the airplane from $\Delta h$ and $\Delta \theta$ initial conditions.

In consideration of the computer limitations and the overall simulation, the landing gear simulation was run at a computational frame time of 39 milliseconds. The time histories on Page 14.1-64 shows the response to a $\Delta h$ initial condition for two different computation times. The increase in computation speed, frame time to 13 ms, results in a smoother response than the "normal" frame time of 39 ms. Neglecting the limit cycle, the response with the 39 ms frame time appears to have the same frequency and slightly increased damping over the 13 ms response.

The time histories on Page 14.1-65 shows the response to a $\Delta \theta$ initial condition. The comments on the $\Delta h$ response are applicable to the $\Delta \theta$ responses.

The effects of slowing the computation speed, or increasing the frame time, are shown in the responses on Page 14.1-66. The amplitude of the limit cycle increased as the computation speed decreased.

The aircraft response due to landing gear deflections felt realistic to the Boeing test pilot in the moving base simulation. Even though a limit cycle existed in the computed response with a 39 ms frame time the amplitude of the computed limit cycle was low and the frequency was high enough (3 cps) that no objectionable characteristics were noted by the pilot.
LANDING GEAR RESPONSE

FRAME TIME = 39 MS (REAL TIME)
$\Delta h_{1C} = 2.5''$ COMPRESSION

GW: 436,000 LB  C.G. = 25%
\[ \Delta \theta_{ic} = -0.2^\circ \]

\[ G_W = 436,000 \text{ LB} \quad c.g. = 25\% \]

**LANDING GEAR RESPONSE**

**BOEING**

NO. D6-30643

Vol. II

SECT PAGE 14.1-66
14.2 PILOTED CHECKOUT

The piloted checkout provided an overall assessment of the simulation as well as quantitative data. Jack Waddell, Boeing 747 project pilot, flew the motion simulator for a total of 5 hours in 3 sessions. He qualitatively evaluated the following characteristics:

1. Airplane handling characteristics
   a. Dutch roll mode
   b. Spiral mode
   c. Short period mode
   d. Phugoid mode
   e. Roll rate
   f. Climb performance
   g. Flap extension and retraction
   h. Speed brakes

2. Engine response

3. Ground effect

4. Control forces

5. Takeoff (3 and 4 engine)

6. Landing

7. Stall

8. Air minimum control speed

9. Buffet

10. Stick shaker

The ground effect characteristics were modified during the approach and landing evaluation. The ground effect data on Pages 2.0-31, -32,
3.0-17, -18, and 4.0-35 were modified by the following factors:

- .7 times the pitching moment increment
- .3 times the drag increment
- .9 times the lift increment

The above changes to the ground effect data were substantiated by additional piloted testing on the Boeing 747 simulation. The resulting Boeing revisions are included in Section 19.

The buffet and stick shaker amplitude and frequency characteristics were tailored to the satisfaction of the pilot. His overall comments substantiated the validity of the simulation.

Rapid roll inputs resulted in lateral acceleration which the pilot felt were greater than in the airplane. The pilots vertical location above the c.g. (10 feet) was reduced to an effective distance of 6 feet in the simulator drive equations. The pilot felt the resulting simulator motion comparable to the airplane. The lateral acceleration due to the pilot location above the c.g. is attributed to aircraft flexibility.

Quantitative data were obtained with NASA pilots. The following pages contain the results of the piloted tests.
14.2.1 TAKEOFF

Two takeoffs at different gross weights were performed to verify the takeoff acceleration. The simulation of basic drag characteristics, thrust lapse rate, ground effect in taxi attitudes, rolling coefficient of friction and inertia effects are indirectly checked by timing takeoff acceleration.

The time to obtain rotation and lift-off speeds were determined by the airspeed indicator and a stop watch. Comparisons between Boeing and NASA simulations are shown in the table on Page 14.2-4. The NASA simulated takeoff started with \( V = 15 \) kt (at \( t = 0 \)) and takeoff thrust while the Boeing takeoff data started at \( V = 0 \) (\( t = 0 \)) and takeoff thrust. To compare the two, the time to accelerate from 0 to 15 knots must be added to the NASA data. This time is equal to \( 15 \) kt divided by the average acceleration. The average acceleration was determined from the curves on Pages 14.2-7 and 14.2-8.

Time histories from the NASA takeoff are shown on Pages 14.2-5 and -6. Velocity versus time data from the NASA test are plotted with the Boeing results on Pages 14.2-7 and -8.
# Takeoff Acceleration Table

<table>
<thead>
<tr>
<th></th>
<th>Flap Position</th>
<th>Gear</th>
<th>C. of W.</th>
<th>C. of G.</th>
<th>Altitude</th>
<th>ΔP ~ Units</th>
<th>T/O IEPR</th>
<th>Rotation</th>
<th>Lift-off</th>
<th>h = 35 FT</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Boeing</td>
<td>10</td>
<td>DN</td>
<td>707.2</td>
<td>14.0</td>
<td>S.L.</td>
<td>8.4</td>
<td>1420</td>
<td>151</td>
<td>47.5</td>
<td>168</td>
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<tr>
<td>NASA</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>45°</td>
<td>47.5°</td>
<td>45°</td>
</tr>
<tr>
<td>Boeing</td>
<td>20</td>
<td>DN</td>
<td>527.2</td>
<td>15.6</td>
<td></td>
<td>6.8</td>
<td>1380</td>
<td>131</td>
<td>30.0</td>
<td>140</td>
</tr>
<tr>
<td>NASA</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>27°</td>
<td>30°</td>
<td>27°</td>
</tr>
</tbody>
</table>

**Computation Tolerance**

<table>
<thead>
<tr>
<th></th>
<th>±2.5 sec</th>
<th>±2 sec</th>
<th>±100 ft</th>
</tr>
</thead>
</table>

1. Add 3.5 sec. to account for starting at 15 kt (V = 15 kt at t = 0)
2. Add 3.0 sec to account for starting at 15 kt

**NASA-Boeing Take-Off Acceleration Comparison**
$G = 707,200 \text{ LB}$

 SEA LEVEL

 $\Delta F = 10$  $\text{c.g.} = 14\%$

 $\Delta F_{R.L.} = -5.4^{\circ}$

 TAKE-OFF
TAKEOFF ACCELERATION
FLAPS 20
G.W. = 527200 LB.
C.G. = 15.6% MAC
TEST 59-10 CERT.
COND. 1.06.051.015.0

NOTE
1. IEPR(t=0) = 1.380
2. Ap = 6.8 UNITS
3. FLT. TEST SIM
   VR  128.6 KIAS 131
   VLOF 139.7  v  140
   t35°  37 SEC. 35

CALC CURNUTT 5-16-70 REVISED DATE TAKEOFF ACCELERATION
CHECK APR FLAPS 20 527200 LB.
APR ODEGARD 5-16-70

THE BOEING COMPANY
14.2.2 CLIMB PERFORMANCE

The purpose of this test was to verify the climb performance simulation. The pilot started the test near sea level and his task was to fly the air speed and EPR schedule prescribed for this condition, Page 14.2-10. At the check altitudes, the computer was put in "hold" and data was obtained from the cockpit instruments and the computer. The comparison between Boeing and NASA simulator results is tabulated on Page 14.2-10 and plotted on Page 14.2-11. The difficulty in performing this task is to have the airplane stabilized when putting the computer to "hold".
## Four-Engine Climb Performance

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
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<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Climb to Alt.</td>
<td>Up</td>
<td>Up</td>
<td>650</td>
<td>14</td>
<td>5</td>
<td>340</td>
<td>1.187</td>
<td>1.187</td>
<td>4.3</td>
<td>4.4</td>
<td>1770</td>
<td>1800</td>
<td>18200</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>10</td>
<td></td>
<td>1.225</td>
<td>1.235</td>
<td>3.8</td>
<td>4.0</td>
<td>1660</td>
<td>1750</td>
<td>16900</td>
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<td></td>
<td></td>
<td></td>
<td></td>
<td>15</td>
<td></td>
<td>1.265</td>
<td>1.27</td>
<td>3.2</td>
<td>3.1</td>
<td>1500</td>
<td>1400</td>
<td>15500</td>
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<td></td>
<td></td>
<td></td>
<td></td>
<td>20</td>
<td></td>
<td>1.309</td>
<td>1.32</td>
<td>2.5</td>
<td>2.8</td>
<td>1220</td>
<td>1100</td>
<td>14000</td>
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<td>25</td>
<td></td>
<td>1.345</td>
<td>1.345</td>
<td>1.7</td>
<td>1.5</td>
<td>760</td>
<td>500</td>
<td>12300</td>
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<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>30</td>
<td></td>
<td>1.417</td>
<td>1.425</td>
<td>2.3</td>
<td>2.5</td>
<td>460</td>
<td>500</td>
<td>10700</td>
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<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>34.3</td>
<td></td>
<td>1.47</td>
<td>1.47</td>
<td>2.7</td>
<td></td>
<td>0</td>
<td>0</td>
<td>9300</td>
</tr>
</tbody>
</table>

**Computation Tolerance**

\[ \pm 0.01 \] \( \pm 0.5 \text{ DEG.} \) \( \pm 5\% \)

**NASA Data was obtained from Cockpit Instruments**

1. **NASA Data at 21700 Ft.**
4 - ENGINE ENROUTE CLIMB
G.W. = 650000 LB.
C.G. = 14% MAC

NOTE
1. MAX. CLIMB THRUST (A/C AIRBLEED ON)
2. STANDARD DAY
3. NO FUEL BURNOFF
4. SEE TABLE (PAGE 52-3) FOR THRUST, IEPR,
   AND ATTITUDE INFORMATION

NASA

COCKPIT DATA

DIGITAL DATA

---

FOUR-ENGINE ENROUTE CLIMB
G.W. = 650000 LB.

RATE OF CLIMB ~ 100 FT/MIN

TIME TO CLIMB ~ MINUTES

MAX. ALTITUDE

NCLT

hp

~1000 FT.

hp

~1000 FT.

10

20

30

40

10

20

30

40

0

2

4

6

8

10

12

14

16

18

0

5

10

15

20

25

30

35

40
14.2.3 **INFLIGHT ACCELERATION AND DECELERATION**

The acceleration condition was performed by starting at $M = .65$ with maximum continuous thrust and accelerating to the maximum speed. Speed and time data were obtained from the cockpit. The comparison between Boeing and NASA simulations is shown on Page 14.2-13.

The deceleration check was performed with speed brakes and idle power, starting at $M = .85$. The NASA simulation results are plotted on Page 14.2-14. The differences in the NASA and Boeing data are the result of a difference in idle EPR. The Boeing test utilized a revised Mach and altitude correction to EPR which affected the idle thrust at high altitude and Mach number. The trend of the results, accounting for the higher idle thrust, appeared correct. The adjustment to the idle EPR was made in the NASA simulation after the checkout data were obtained.
G.W. = 450000 LB.
ALT. = 35000 FT.
CG = 25% MAC

TIME - MINUTES

MA CH NUMBER

FLAPS UP

NOTE: 1. D6-20423 REV. D SIM. DATA
2. IDLE THRUST (ZERO THROTTLE)

F_n = 1600 LB.
F_n = 3400 LB.

LEVEL FLIGHT DECELERATION
FLAPS UP
450000 LB. 35000 FT.
THE BOEING COMPANY

CALC CHECK APR APR GLENN
BRYANT 5-16-70
REVISED DATE 5-19-70

PAGE 14.2-14
14.2.4 STEADY TURNS

This test was conducted to check the stick force and elevator required to hold a bank angle, or load factor. The pilot trimmed the airplane for straight and level flight and then rolled into a steady coordinated turn at a specified bank angle. The airplane was allowed to descend in order to maintain a constant airspeed. When the condition was stabilized, the computer was put in "hold" and the computed values of $F_s$, $\phi_B$, and $\delta_e$ were obtained. The results are tabulated with the Boeing simulation results on Page 14.2-16. The data are plotted, along with flight test results, on Pages 14.2-17 and -18. The data scatter is a result of the difficulty in having a stabilized condition when going to "hold". A calculated value of $\eta = 1/\cos\phi_B$ was used in plotting the data.
**STEADY COORDINATEDTurns**

<table>
<thead>
<tr>
<th>FLAP POSITION</th>
<th>GEAR</th>
<th>C.M. 1000 LB</th>
<th>C.G. % MAC</th>
<th>ALTITUDE 1000 FT</th>
<th>$\Delta P$</th>
<th>VI/M</th>
<th>$\phi_B$</th>
<th>$F_g$</th>
<th>$\sigma_e$</th>
<th>$n_2$</th>
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</thead>
<tbody>
<tr>
<td></td>
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<td></td>
<td></td>
<td>BOEING</td>
<td>NASA</td>
<td>BOEING</td>
<td>NASA</td>
<td>BOEING</td>
<td>NASA</td>
<td>BOEING</td>
</tr>
<tr>
<td>UP</td>
<td>UP</td>
<td>540</td>
<td>31</td>
<td>34</td>
<td>.86</td>
<td>3.1</td>
<td>30</td>
<td>30.6</td>
<td>15.0</td>
<td>1.3</td>
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<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>45</td>
<td>47/47.5</td>
<td>33.0</td>
<td>14.0</td>
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<td>UP</td>
<td>560</td>
<td>14</td>
<td>34</td>
<td>.86</td>
<td>4.4</td>
<td>30</td>
<td>28.1</td>
<td>14.5</td>
<td>1.0</td>
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<td></td>
<td>50</td>
<td>50.7</td>
<td>44.0</td>
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<td>61.5</td>
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<td>-3.2</td>
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<tr>
<td>30</td>
<td>DN</td>
<td>459</td>
<td>30.8</td>
<td>11</td>
<td>127</td>
<td>5.7</td>
<td>30</td>
<td>31.1</td>
<td>11.5</td>
<td>-1.8</td>
</tr>
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<td></td>
<td></td>
<td>45</td>
<td>19.5</td>
<td>19.9/8.1</td>
<td>-7.6</td>
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</table>

**COMPUTATION TOLERANCE**

<table>
<thead>
<tr>
<th>BOEING</th>
<th>NASA</th>
</tr>
</thead>
<tbody>
<tr>
<td>± 10%</td>
<td>± 0.5 DEG.</td>
</tr>
</tbody>
</table>

* INITIAL BUFFET

\[ n_2 = \frac{1}{\cos \phi_B} \]
FLAPS UP  
GEAR UP

<table>
<thead>
<tr>
<th>NASA</th>
<th>SYM</th>
<th>A/P</th>
<th>FLT.</th>
<th>COND. NO.</th>
<th>C.G.</th>
<th>% MAC</th>
<th>G.W. ~ 1000 LB</th>
<th>ALT. ~ FT</th>
<th>M</th>
</tr>
</thead>
<tbody>
<tr>
<td>C</td>
<td>O</td>
<td>RA001</td>
<td>31-6</td>
<td>1.21.003.005.0, 6-7</td>
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<td>34000</td>
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<td>.86</td>
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<td>14</td>
<td>560</td>
<td></td>
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</tr>
</tbody>
</table>

SIMULATOR (D6-20423, REV. D DATA)

NOTE: 1. CONSTANT THROTTLE (P.L.F. @ $n_z = 1.0$)
2. SIMULATOR $F_s$ INCLUDES 2.5 LB. BREAKOUT FORCE

**BUFFET ONSET**

**LOAD FACTOR, $n_z$**

$F_s$ ~ LB.

$80$

$60$

$40$

$20$

$0$

$-2$

$-4$

$\theta_e$ ~ DEG.

$30^\circ$

$40^\circ$

$45^\circ$

$50^\circ$

$55^\circ$

$60^\circ$

$\cos^{-1}(1/n_z)$

**THE BOEING COMPANY**

STEADY COORDINATED TURN

35000 FT.  
M = .86

747

D6-30643  
Vol. II

THE BOEING COMPANY

CALC BYSTROM 4-4-70  
REvised CURNUtt 15-18-70  
DATE
NOTE: 1. CONSTANT THROTTLE (P.F. @ N_z = 1.0)
2. SIMULATOR F_S INCLUDES 2.5 LB BREAKOUT FORCE
14.2.5 **LONGITUDINAL STATIC STABILITY**

The airplane was trimmed for straight and level flight. Without changing the stabilizer or throttle, the pilot slowly changed the speed using the elevator. The thrust discrepancy was allowed to generate a rate of climb or descent. Computed values of $\Theta_e$, $F_s$, and $\Theta_B$ were recorded when the airplane was stabilized at the prescribed speed. The column was then released to see if the airplane would return to the trim speed. The results are plotted on Page 14.2-20.
<table>
<thead>
<tr>
<th>NASA</th>
<th>SYM</th>
<th>A/P</th>
<th>FLT.</th>
<th>COND. NO.</th>
<th>%MAC</th>
<th>GW</th>
<th>ALT.</th>
<th>TRIM</th>
<th>D/EPR</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>XLC</td>
<td>RAI01</td>
<td>4-3 CERT.</td>
<td>1.22.051.001-009</td>
<td>32</td>
<td>562000</td>
<td>10000</td>
<td>1.172</td>
<td></td>
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</tbody>
</table>

$\alpha_{FRL}^* = 1.26^\circ$

**Simulator** (DG-20423 REV. D DATA)

**Note:**
1. Shaded 'SYM30L' indicates trim point
2. Flagged symbols indicate return to trim
3. Thrust for level flight at the trim speed
4. Mass unbalance effects included in breakout

**Approach Stability**

<table>
<thead>
<tr>
<th>CALC.</th>
<th>HOLTZNER 5-30-7</th>
<th>REVISED</th>
<th>CORNUOT 5-19-70</th>
<th>DATE</th>
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<tr>
<td>APR</td>
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<td>APR</td>
<td>GLENN 5-20-70</td>
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</tr>
</tbody>
</table>

**The Boeing Company**

*AD 461 C.R*
14.2.6 STEADY SIDESLIPS

After a trimmed flight condition was set up, the pilot slowly applied rudder to hold a steady sideslip. A combination of lateral control and bank angle was used to maintain a straight and level flight path. When the airplane was stabilized, computed values of $\beta$, $\delta_R$, $\xi_w$, and $\phi_B$ were recorded. Comparisons between NASA and Boeing simulations and flight test are shown on Pages 14.2-22 and -23.
14.2.7 ROLL RATES

The pilot trimmed the airplane for straight and level flight and then rolled into a 30° bank. The yaw damper was on and the rudder pedals were not used. After the bank angle was established, a wheel input was rapidly applied and held until the airplane banked to 30° in the opposite direction. The time to bank through the 60° was obtained with a stop watch and the maximum roll rate and magnitude of wheel input were obtained from the computed time histories. The average roll rate was determined by dividing the change in bank angle, 60°, by the time required to bank 60°. The NASA simulator data is compared to Boeing results in the table on Page 14.2-25 and to the plotted data on Pages 14.2-26 thru -28.
<table>
<thead>
<tr>
<th>FLAP POSITION</th>
<th>GEAR</th>
<th>G*:1000 lb</th>
<th>ALTITUDE 1000 ft</th>
<th>V.IM</th>
<th>HYD. SYS. IMPERATIVE</th>
<th>DAMPER</th>
<th>θ_W TRIM ~DEG.</th>
<th>t-30/30 ~SEC.</th>
<th>P_AVG. ~DEG/SEC</th>
<th>P_MAX ~DEG/SEC</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>UP</td>
<td>710</td>
<td>10</td>
<td>200</td>
<td>ON</td>
<td>-3.4</td>
<td>23</td>
<td>32°</td>
<td>7.5</td>
<td>8.6</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>50</td>
<td>7.5</td>
<td>8.6</td>
<td>9.5</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>80</td>
<td>4.1</td>
<td>12.5</td>
<td>19.1</td>
</tr>
<tr>
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<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>4.8/4.5</td>
<td>13.4</td>
<td>19.1</td>
<td>19/17</td>
</tr>
<tr>
<td>0</td>
<td>UP</td>
<td>550</td>
<td>35</td>
<td>.86</td>
<td>ON</td>
<td>-.8</td>
<td>25</td>
<td>41°</td>
<td>8.6</td>
<td>7.1</td>
</tr>
<tr>
<td></td>
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<td></td>
<td></td>
<td></td>
<td></td>
<td>50</td>
<td>4.1</td>
<td>5.1</td>
<td>15.8</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>80</td>
<td>3.5</td>
<td>3.3/3.5</td>
<td>19.1</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>80</td>
<td>3.3/3.5</td>
<td>18.8/17</td>
<td>23.6</td>
</tr>
<tr>
<td>30</td>
<td>DN</td>
<td>564</td>
<td>180</td>
<td>142</td>
<td>ON</td>
<td>-7.6</td>
<td>20</td>
<td>3.8</td>
<td>16.0</td>
<td>21.5</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>25</td>
<td>12.0</td>
<td>5.0</td>
<td>14.0</td>
</tr>
<tr>
<td></td>
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<td></td>
<td></td>
<td></td>
<td></td>
<td>35</td>
<td>7.1</td>
<td>8.5</td>
<td>14.0</td>
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<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>80</td>
<td>5.5</td>
<td>6.7</td>
<td>15.5</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>80</td>
<td>5.8</td>
<td>10.3</td>
<td>16.5</td>
</tr>
</tbody>
</table>
FLAPS 10
10000 FT
** 564000 LB

NOTE:  
1. TRIM AT -30° BANK AND ROLL THROUGH +30° BANK.  
2. GEAR UP; YAW DAMPER ON.  
3. $I_{xx} = 14.3 \times 10^6$ SLUG-FT²  
4. REDUCE ROLL RATES 1 DEG/SEC. FOR 710000 LB.  
   ($I_{xx} = 20.3 \times 10^6$ SLUG-FT²)

FLIGHT TEST

HYBRID SIMULATOR (D6-20423, REV. D)

(710000 LB)

NASA

ROLL RATE CAPABILITY
FLAPS 10

THE BOEING COMPANY
NOTE: 1. TRIM AT -30° BANK AND ROLL THROUGH +30° BANK.
2. YAW DAMPER ON 1.
3. $I_{xx} = 14.0 \times 10^6$ SLUG-FT.
4. WHEEL EFFECT APPLIES TO TEST CONDITIONS
   ($I_{xx} = 17.8 \times 10^6$ SLUG-FT).

ROLL RATE CAPABILITY
FLAPS UP ~ 35,000 FT.

THE BOEING COMPANY
NOTE:
1. TRIM AT -30° BANK AND ROLL THROUGH ±10° BANK.
2. GEAR DOWN. YAW DAMPER ON.
3. \( L_{x} = 1.43 \times 10^4 \) SLUG FT²
4. WHEEL EFFECT APPLIES TO TEST CONDITIONS
   \( I_{x,y} = 18.7 \times 10^5 \) SLUG FT²

FLIGHT TEST
HYBRID SIMULATOR (DC-20423, REV D)

\( P_{\text{MAX}} \)
\( \frac{1}{2} \text{ DEG} \)
\( \text{SEC} \)

\( \frac{S_{w}}{60^\circ} \)
\( 60^\circ \)
\( 50^\circ \)
\( 25^\circ \)
\( 15^\circ \)

\( V_{x} \sim \text{KNOTS} \)

ROLL RATE CAPABILITY
FLAPS 30

THE BOEING COMPANY
14.2.8 **AIR MINIMUM CONTROL SPEED**

Minimum control speed in the air is defined as the lowest speed at which an airplane can maintain straight flight with a critical engine failed, rudder at full deflection, and bank angle at 5 degrees (dead engine high). The results from the NASA test are plotted on Page 14.2-30.
FLAPS 30
ΔR = +25°  Ψb = 0

NOTE:
1. DG = 20423 REV. D SIM. DATA
2. 555500 LB.
3. 1870 FT.
4. IEPR = 1.382 ON ENG. #1 AT Vmca, ENG. #2
   Δ #5 ADJUSTED FOR LEVEL FLIGHT
5. ENG. #4 WINDMILLING
6. GEAR, UP
7. C.G. = 26% MAC
8. CONSTANT HEADING

V_mca = 96 KTS.
15.0  **APPENDIX A - FLIGHT MANUAL**

Selected pages from the performance section of the FAA approved airplane flight manual are contained in this section. This information will allow the simulation to be operated in compliance with the appropriate performance criteria and certification requirements of FAR Part 25 and Part 36.
THIS AIRPLANE MUST BE OPERATED IN COMPLIANCE WITH THE
PRESCRIBED CERTIFICATE LIMITATIONS IN SECTION 1 HEREIN

Approved by: Robert E. Stanton
Chief, Aircraft Engineering Division,
FAA, WESTERN REGION
Date December 30, 1969

DS-13703
PUBLISHED BY THE BOEING COMPANY, COMMERCIAL AIRPLANE GROUP, RENTON, WASHINGTON U.S.A.
The information in this section is presented for compliance with the appropriate performance criteria and certification requirements of FAR Part 25 and Part 36.

STANDARD PERFORMANCE CONDITIONS

All performance in this section is based on the following:

1. Approved engine thrust ratings less installation losses, airbleed and accessory losses.
2. Full temperature accountability within the operational limits, except for landing distance, which is based on standard day temperatures.
3. Trailing edge flaps positions as follows:

<table>
<thead>
<tr>
<th>Trailing Edge Flaps</th>
</tr>
</thead>
<tbody>
<tr>
<td>Takeoff</td>
</tr>
<tr>
<td>Transition Flap Setting</td>
</tr>
<tr>
<td>Enroute</td>
</tr>
<tr>
<td>Approach</td>
</tr>
<tr>
<td>Landing</td>
</tr>
<tr>
<td>10, 20</td>
</tr>
<tr>
<td>1, 5</td>
</tr>
<tr>
<td>0</td>
</tr>
<tr>
<td>20</td>
</tr>
<tr>
<td>25, 30</td>
</tr>
</tbody>
</table>

4. Leading edge devices in the appropriate position for trailing edge flap position.

VARIABLE FACTORS AFFECTING PERFORMANCE

Details of the variable factors affecting performance are given under performance configuration, but certain assumptions relating to all performance data are as follows:

ICING PROTECTION

The effect of anti-icing systems operation is shown on applicable charts.

HUMIDITY

Humidity has no appreciable effect on the thrust of the engines; therefore, it has not been considered in the performance data.

WIND

The wind velocity used in calculations is factored to assure compliance with the relevant operating regulations. All charts should be entered with actual tower-reported wind components.
GENERAL

Definitions

Airspeeds

All airspeed and Mach values in this manual assume a zero instrument error.

All Indicated Airspeeds are based on normal static source position error.

Position error is the instrument indication error due to location of static ports.

Equivalent Airspeed, EAS - Airspeed indicator reading, as installed in the airplane, corrected for static source position error and compressibility.

Calibrated Airspeed, CAS - Airspeed indicator reading, as installed in the airplane, corrected for static source position error.

Indicated Airspeed, IAS - Airspeed indicator reading, as installed in the airplane, uncorrected for static source position error.

True Mach Number, M - Machmeter reading, as installed in the airplane, corrected for static source position error.

Critical Engine Failure Speed, V\textsubscript{f} - The speed at which, when an engine failure is recognized, the distance to continue the takeoff to a height of 35 feet will not exceed the usable takeoff distance; or, the distance to bring the airplane to a full stop will not exceed the accelerate-stop distance available. V\textsubscript{f} must not be less than the Ground Minimum Control Speed, VMCG, or greater than the rotation speed, VR, or greater than the maximum brake energy speed, VM\textsubscript{BE}.

Engine Failure Speed Ratio, V\textsubscript{f}/VR - The ratio of the engine failure speed, V\textsubscript{f}, for actual runway dimensions and conditions, to the rotation speed, VR.

Maximum Brake Energy Speed, VM\textsubscript{BE} - The maximum speed on the ground from which a stop can be accomplished within the energy capabilities of the brakes.
GENERAL

DEFINITIONS

AIRSPEEDS (Continued)

Rotation Speed, $V_R$ - The speed at which rotation is initiated during the takeoff.

Takeoff Safety Speed, $V_2$ - The scheduled target speed to be attained at the 35 foot height with one engine inoperative.

Air Minimum Control Speed, $V_{MCA}$ - The minimum flight speed at which the airplane is controllable with a maximum of 5° bank when the critical engine suddenly becomes inoperative with the remaining engines at takeoff thrust.

Ground Minimum Control Speed, $V_{MCG}$ - The minimum speed on the ground at which the takeoff can be continued, utilizing aerodynamic controls alone, when the critical engine suddenly becomes inoperative with the remaining engines at takeoff thrust.

Landing Reference Speed, $V_{REF}$ - The minimum speed at the 50 foot height in a normal landing. This speed is equal to 1.3 times the stall speed in the landing configuration.

Design Maneuvering Speed, $V_A$ - The maximum speed at which application of full available aileron, rudder or elevator will not overstress the airplane.
GENERAL

DEFINITIONS (Continued)

TEMPERATURE

ISA - International Standard Atmosphere, as accepted by the International Civil Aviation Organization.

OAT - Outside Air Temperature - the free air static (ambient) temperature.

SAT - Static Air Temperature - outside air (ambient) temperature as computed by the Air Data Computer and presented on the Static Air Temperature indicator.

TAT - Total Air Temperature - static air temperature plus adiabatic compression (ram) rise as indicated on the Total Air Temperature indicator.

WIND

Wind Velocity - The actual wind velocity at a 50 foot height reported from the tower and corrected by the wind component chart to a headwind or tailwind component parallel to the flight path.

HEIGHT

Gross Height - The geometric height attained at any point in the takeoff flight path using gross climb performance. Gross height is used for calculating actual pressure altitudes at which obstacle clearance procedures and wing flap retraction are initiated, and level-off-height scheduled.

Net Height - The geometric height attained at any point in the takeoff flight path using net climb performance. Net height is used to determine the net flight path which must clear any obstacles by at least 35 feet to comply with the regulations.

ICING

Icing Conditions - Icing may develop when visible moisture such as fog, rain, or wet snow is present with Static Air Temperature below 8°C (46°F).

TAKEOFF DATA

Balanced Field Length - The condition where V1 is selected to make the takeoff distance equal to the accelerate-stop distance.

Unbalanced Field Length - The condition where V1 is selected to make the takeoff distance and accelerate-stop distance unequal.
GENERAL

GRADIENT OF CLimb

Gross Gradient - The demonstrated ratio, expressed as a percentage, of:

\[
\frac{\text{Change in Height}}{\text{Horizontal Distance Traveled}}
\]

The gradients shown on the charts are true gradients, i.e., they are based on true, not pressure, rates of climb.

Net Gradient - The demonstrated gross gradient reduced by the increment as required by regulation.

BUFFET ONSET CHARACTERISTICS

The buffet boundary is a basic characteristic of the airplane that is defined by angle of attack and Mach numbers. Below approximately 0.85 M, the buffet is related to the wing maximum lift capability (therefore to true airplane stall). In the range above 0.85 M, buffet is related to the growth of shock waves on the wing.

At any flight condition, it is possible to determine the altitude, low-speed, high-speed, and maneuvering margins before buffet onset occurs. (See Cruise Maneuvering Capability chart.)

CROSSWIND VALUES (Takeoff and Landing)

The maximum demonstrated crosswind component is 30 knots reported wind at 50 foot height. This component is not considered to be limiting.

For performance scheduling, the full headwind component may be used provided that the corresponding crosswind component does not exceed 30 knots.

MINIMUM CONTROL SPEEDS:

The Air Minimum Control Speeds (VMCA), and the Ground Minimum Control Speeds (VMCG), of this airplane are shown on a chart in this Section, and on the takeoff speeds charts where applicable.
PERFORMANCE CONDITIONS AND PROCEDURES

Takeoff field length performance shown in this section accounts for 115 percent of the all-engines operating distance, or the total distance considering an engine failure recognition at V1, whichever is greater. These distances are based on a smooth, dry, hard-surfaced runway.

The appropriate airplane configuration, outlined under Performance Configurations, was used.

The conditions and procedures used in establishing the performance data in this manual are presented under each phase of operation. Procedures are guidance material only.

TAKEOFF

Conditions

Prior to takeoff, a review was made of stabilizer and flap settings, takeoff speeds, and that sufficient field length was available for the gross weight and ambient conditions. Corrections were applied, when necessary, for significantly altered ambient conditions or loading.

Procedures

Thrust was set to 1.1 EPR prior to brake release, or as the airplane was aligned with the runway. With no wind, or a direct headwind, EPR was advanced prior to brake release. Thrust levers were adjusted as necessary to obtain target EPR values by approximately 40 to 80 knots on the takeoff roll.

Rudder pedals were used for directional control through the nose wheel and rudder.

Rotation to takeoff attitude was initiated at VR. A speed not less than V2 was obtained at a height of 35 feet. The landing gear was retracted after a positive rate of climb was established.

A smooth positive rotation was used to the initial climb attitude (approximately 13 to 22 degrees depending upon gross weight and thrust available). Minor attitude variations were made after liftoff to achieve the initial climb speed. Engine failure results in approximately 2 to 2-1/2 degrees lower attitude than normal climb.

NOTE: With center of gravity at or near the aft limit, avoid sudden brake release and maintain forward pressure on the control column to approximately 80 knots to increase nose wheel steering effectiveness.
PERFORMANCE CONDITIONS AND PROCEDURES

REFUSED TAKEOFF (Anti-Skid On)

Conditions

Calculated accelerate-stop distances account for demonstrated recognition and reaction times, plus arbitrary time delays.

Reverse thrust was not used.

Procedures

When an engine failure occurred, the takeoff was refused when the failure was recognized prior to V1.

If the takeoff was refused for any reason, prior to V1, the following procedure was accomplished as rapidly as possible:

Wheel Brakes - MAXIMUM BRAKING APPLIED
All Thrust Levers - IDLE
Speed Brakes - UP

CLIMB-OUT (3 or 4 Engines)

Conditions

Climb gradient and obstacle clearance flight path performance is based on the most critical engine inoperative at V1.

Procedures

Takeoff flap setting and V2 speed were maintained to at least the height selected for initiation of flap retraction.

Flaps were retracted according to the Flap Retraction Speed Schedule, in this section.

Enroute procedures were followed after climbing to at least 1500 feet above runway elevation, or after all takeoff flight path obstacles had been cleared.
PERFORMANCE CONDITIONS AND PROCEDURES

OBSTACLE CLEARANCE

Conditions

With all engines operating, a speed not greater than \( V_2 + 10 \) knots was maintained until either the scheduled flap retraction height, or the minimum gross height for obstacle clearance (whichever was lower), was reached.

When engine failure occurred prior to \( V_2 \), and the takeoff weight was obstacle limited, \( V_2 \) was maintained up to the gross height required for obstacle clearance.

If an engine failure occurred after \( V_2 \), speed at engine failure (\( V_2 + 10 \) knots maximum) was maintained up to the gross height required for obstacle clearance.

Procedures

When the height selected for initiation of flap retraction was limited due to distant-obstacle considerations, the procedure was to initiate flap retraction and accelerate to final takeoff climb speed while maintaining constant altitude and initial takeoff thrust setting.

Final takeoff climb was continued to 1500 feet above runway elevation, or to the minimum gross height required for obstacle clearance, at final takeoff climb speed and maximum continuous thrust.

NOTES: The airplane should be levelled off, and flaps retracted at the selected level-off height, only if the limiting obstacle is beyond the Third Segment and an engine failure has occurred.

The height selected for initiation of flap retraction may be limited by available performance as described under Takeoff Flight Path, this section. Vertical clearance of either close-in or distant obstacles in the intended flight path must be established by reference to the appropriate obstacle clearance charts.
LANDING FIELD LENGTH

Conditions

All landing field lengths shown in this section are based on standard day temperatures on a smooth, level, hard-surfaced runway. Dry landing field lengths are demonstrated landing distances, from a 50 foot height at VREF, divided by a factor 0.6. Wet landing field lengths are determined by multiplying the dry landing field length by a factor of 1.15.

Procedures

Approach and landing were made with landing gear down, flaps in landing position, thrust reduced to flight-idle on all engines before touch-down, and automatically set to ground-idle on all engines after touch-down.

When the landing was made with anti-skid operating, full speed brakes and maximum wheel braking were applied 2 seconds or less after touch-down.

When the landing was made with anti-skid inoperative, speed brakes were raised immediately upon touch-down and steady, light braking was used if gross weight was approximately 500,000 lb (226,800 kg.) or less. Steady, light to moderate braking was used if landing weight was over 500,000 lbs (226,800 kg.). Brakes were modulated as necessary to prevent skidding.

NOTE: Landing field lengths are not based on use of reverse thrust.
FLAP RETRACTION SPEED SCHEDULE

Maximum level-off heights, Third Segment distances and Final Segment climb performance shown in this manual are based upon retracting the wing flaps during Third Segment acceleration using the schedule below. This schedule is recommended for all normal flap retraction operations.

During acceleration, select flap positions at the following initiation speeds:

**TAKEOFF FLAPS POSITION 10 OR 20**

<table>
<thead>
<tr>
<th>Initiation Speed, Knots</th>
<th>Select Flap Position</th>
</tr>
</thead>
<tbody>
<tr>
<td>V2 + 20</td>
<td>10</td>
</tr>
<tr>
<td>V2 + 40</td>
<td>5</td>
</tr>
<tr>
<td>V2 + 60</td>
<td>1</td>
</tr>
<tr>
<td>V2 + 80</td>
<td>0</td>
</tr>
</tbody>
</table>

Final Segment Climb Speed: V2 + 80 Knots.

When flaps are being retracted at a constant altitude due to engine failure, with a critical final segment obstacle, begin climbing when V2 + 80 knots is achieved, maintaining takeoff thrust setting until flaps are completely retracted.
The airplane configuration associated with the performance data in this manual is shown below. Performance conditions not shown below are on the appropriate charts.

<table>
<thead>
<tr>
<th>PERFORMANCE CONFIGURATION</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>THRUST</strong></td>
</tr>
<tr>
<td><strong>TAKEOFF</strong></td>
</tr>
<tr>
<td><strong>1st SEGMENT CLIMB</strong></td>
</tr>
<tr>
<td><strong>2nd SEGMENT CLIMB</strong></td>
</tr>
<tr>
<td><strong>3rd SEGMENT</strong></td>
</tr>
<tr>
<td><strong>FINAL TAKEOFF CLIMB</strong></td>
</tr>
<tr>
<td><strong>ENROUTE CLIMB</strong></td>
</tr>
<tr>
<td><strong>APPROACH CLIMB</strong></td>
</tr>
<tr>
<td><strong>LANDING CLIMB</strong></td>
</tr>
<tr>
<td><strong>LANDING</strong></td>
</tr>
</tbody>
</table>

Anti-skid is on for takeoff and landing with up to two brakes deactivated, unless anti-skid inoperative braking performance is used.

One or three Air Conditioning Packs are on for all takeoff thrust operations.

All three Air Conditioning packs are on for final takeoff climb. For enroute climb the number of the Air Conditioning packs on is noted on the charts.

Wing anti-icing is off during all flaps-extended operations.
NOISE CHARACTERISTICS

No determination has been made by the Federal Aviation Administration that the noise levels in this manual are, or should be, acceptable or unacceptable for operation at, into, or out of, an airport.

The noise levels tabulated below are the result of Federal Aviation Regulations, Part 36 certification tests:

<table>
<thead>
<tr>
<th>GROSS WEIGHT</th>
<th>FLAPS</th>
<th>NOISE LEVEL (EPNdB)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pounds</td>
<td>Kilograms</td>
<td>Position</td>
</tr>
<tr>
<td>710,000</td>
<td>322,056</td>
<td>10</td>
</tr>
<tr>
<td>673,000</td>
<td>305,273</td>
<td>20</td>
</tr>
<tr>
<td>564,000</td>
<td>255,830</td>
<td>30</td>
</tr>
</tbody>
</table>

The traded noise level is 112.0 EPNdB

Normal all engines takeoff procedures were used, with a climb-out at V2 + 10 knots at Takeoff Thrust with no thrust cut-back. The landing approach was made at VREF + 10 knots.
AIRPLANE FLIGHT MANUAL

PERFORMANCE

TAKEOFF EPR

40-80 KNOTS

PRESSURE ALTITUDE — 1000 FT.

NOTE: NO EPR REDUCTION REQUIRED FOR NACELLE ANTI-ICE "ON" AT TEMPERATURES BELOW 8°C (46°F) WITH 1 A/C PACK OPERATING ADD 0.1 EPR.

OFF

ON

INDICATED ENGINE PRESSURE RATIO

OAT — DEGREES C

FAA APPROVED 6-30-70

SECT

PAGE 15.0-17
PERFORMANCE
TAKEOFF EPR
GO-AROUND

PRESSES ALTITUDE — 1000 FT.

NOTE: NO EPR REDUCTION REQUIRED
FOR NACELLE ANTI-ICE "ON"
AT TEMPERATURES BELOW
-10°C (14°F)
WITH 1 A/C PACK OPERATING
ADD 0.1 EPR.
SET STABILIZER FOR TAKEOFF CENTER OF GRAVITY DETERMINED BY CALCULATED OR GRAPHICAL METHOD, WITH GEAR DOWN AND WITH TAKEOFF FLAPS.

\[ \delta_p = 3 - \delta_{FRL} \]
\[ \delta_{FRL} = 3 - \delta_p \]
NOTE APPLICABLE FOR TAKEOFF AND LANDING ALTITUDES ONLY.

GEAR UP EXCEPT AS NOTED.

STALL SPEEDS - KNOTS CAS

GROSS WEIGHT — 1000 LB.
MAXIMUM ALLOWABLE TAKEOFF WEIGHT

NORMAL TAKEOFF WEIGHT ANALYSIS

CHART READING PROCEDURE

The following steps will be adequate for most takeoff situations. The dotted guide-lines on typical charts and the Illustrative Examples are for guidance purposes.

1. Select most probable takeoff flap setting.

2. Runway Length Corrections (All Engines) chart - Enter with actual runway length available. Make slope and wind corrections. Read corrected runway length.

3. Runway Length and Vl Adjustments chart - Enter with actual runway length available on both the Runway Length, and Accelerate-Stop Distance scales. Make slope and wind corrections from the respective reference lines. Where the two corrected distances intersect on the "web" portion of the chart, read Vl/VR, and corrected runway length.

4. Maximum Takeoff Weight, Field Length Limits chart - Enter with the lesser of the corrected runway lengths from Step 2 or Step 3, airport pressure altitude, and temperature. Read gross weight.

5. Maximum Takeoff Weight, Climb Limits chart - Enter with airport pressure altitude and temperature. Read gross weight.

6. IF 200 MPH TIRES ARE INSTALLED, USE:
   Maximum Takeoff Weight, Tire Speed chart - Enter with airport pressure altitude and temperature. Make wind correction. Read gross weight.

NOTE: If obstacles are present, proceed with Steps 7 through 10.
MAXIMUM ALLOWABLE TAKEOFF WEIGHT

NORMAL TAKEOFF WEIGHT ANALYSIS

CHART READING PROCEDURE (Continued)

7. Takeoff Climb chart - Enter with least gross weight determined by Steps 4, 5, 6, structural limits, or operational requirements, then proceed to temperature and airport pressure altitude. Read Second Segment gradient. Correct the gradient for wind, if applicable, using Gradient Corrections chart.

8. Obstacle Clearance (Close-in Obstacles) chart - Enter with obstacle distance from end of takeoff distance required. Correct the distance for wind. Proceed vertically to zero-wind gradient available, from Step 7.

   (a) If this exceeds obstacle height, obstacle is cleared.
   (b) If obstacle height is not exceeded, follow dashed field-length-trade guide lines to obstacle height. Read gradient required for obstacle clearance. This gradient accounts for the shorter takeoff distances required when gross weight is reduced. (See SLOPED RUNWAY EFFECT ON OBSTACLE CLEARANCE).

9. Obstacle Clearance (Distant Obstacles) chart - Enter with obstacle distance from end of takeoff distance required. Proceed vertically to wind-corrected gradient available, from Step 7.

   (a) If this exceeds obstacle height, obstacle is cleared. However, now drop back down to obstacle height, then move left (horizontally) to wind-corrected gradient available. Read gross height.
   (b) If obstacle is not cleared at wind-corrected gradient available, follow dashed field-length-trade lines to the obstacle height. Read gross height and gradient required for obstacle clearance. Correct the gradient required to zero-wind gradient, on the Gradient Corrections chart. (See OBSTACLE IN THIRD SEGMENT, and OBSTACLE IN FINAL SEGMENT.)
MAXIMUM ALLOWABLE TAKEOFF WEIGHT

NORMAL TAKEOFF WEIGHT ANALYSIS

CHART READING PROCEDURE  (Continued)

10. Takeoff Climb chart - Enter gradient scale with the largest zero-wind gradient required for obstacle clearance as determined from Step 8 or 9. Proceed to airport pressure altitude, and temperature.  Read gross weight.

11. Takeoff Speeds chart - Enter with airport pressure altitude, temperature, and the least gross weight determined by Steps 4, 5, 6, 10, structural limits, or operating requirements.  Read VR. and V2. Read V1 at the V1/VR ratio from Step 3. The resulting V1 must exceed the minimum V1 (VMCG), (See V1 Less than VMCG), and the resulting VR must exceed the VMCA limit. (See VR Less than VMCA Limit).

12. Maximum Brake Energy Limit Speed chart - Enter with airport pressure altitude, temperature, and gross weight used in Step 11. Correct for runway slope, and wind.  Read VMEE. This speed must exceed V1 of Step 10. (See V1 Greater than VMEE).

13. Gross Height - Pressure Altitude Conversion chart - Enter with gross height obtained from Step 9. Correct for temperature and altitude.  Read pressure altitude increment.  Add this value to the airport pressure altitude.  Acceleration and flap retraction should not be scheduled below this altimeter reading.

14. Determine flap retraction and final climb speeds from the FLAP RETRACTION SPEED SCHEDULE.
MAXIMUM ALLOWABLE TAKEOFF WEIGHT

TAKEOFF WEIGHT ANALYSIS WITH IMPROVED CLimb PERFORMANCE

The following steps describe the procedure for finding the maximum allowable takeoff weight when using Improved Climb Performance. Improved Obstacle Clearance with speed increase is considered under a separate paragraph heading. The term, "normal", refers to the weights and speeds based on no increase in takeoff speeds.

1. Steps 1 through 6 of the Normal Takeoff Analysis are unchanged.

2. Maximum Takeoff Weight and Speeds (Improved Climb Performance) chart - Enter the weight portion of this chart with normal field length and climb limited gross weights. Follow the guidelines until they intersect. At the intersection, read gross weight and speed increase. Repeat, using tire speed and climb limited gross weights if 200 MPH tires are installed. Use lesser weight and speed increase.

3. Takeoff Speeds chart - Enter this chart, as in Normal Analysis, with pressure altitude and temperature, and with the gross weight determined from Step 2 above. Read VR, V2, and V1 for the V1/VR ratio in Step 3 of Normal Analysis.

4. Maximum Takeoff Weight and Speeds (Improved Climb Performance) chart - Enter the speed correction portion with the VR, V2 and V1 speeds determined in Step 3 above. Follow the guidelines to the speed increase determined in Step 2 above. Read Corrected VR, V2, and V1. V1 must be less than VMBE. (See V1 Greater than VMBE).

5. Determine flap-retraction speeds based on the V2 speed derived from Step 3 above.
MAXIMUM ALLOWABLE TAKEOFF WEIGHT

SELECTION OF V1

When the calculated or scheduled takeoff weight is not field-length limited, V1 may be raised or lowered to suit operating conditions within the restrictions imposed by the available runway length, VMCG, VR, and VMBE. Even when takeoff weight is field-length limited, a small reduction in V1 is available if the 3-engine field length requirement is shorter than the all-engine requirement. This is generally true for this airplane.

The following additional Steps are required in the Normal Takeoff Weight Analysis to determine the V1 limits:

1. Maximum Takeoff Weight (Field Length Limits) chart - Enter with airport pressure altitude, temperature, and actual scheduled takeoff weight. Read corrected runway length required.

2. Runway Length and V1 Adjustments chart - Enter, in upper right portion of the chart, with the corrected runway length obtained from Step 1. Where this runway length intersects the slope and wind corrected takeoff distance available line as determined in Step 3 of the Normal Analysis procedure, read minimum V1/VR. Where this runway length intersects the slope and wind corrected accelerate-stop distance line as determined in Step 3 of the Normal Analysis procedure, read maximum V1/VR.

If either intersect occurs off the limits of the chart, use minimum or maximum values shown on the chart.

3. Takeoff Speed chart - At the actual scheduled takeoff weight, and the V1/VR ratios determined in Step 2 above, read minimum V1 and maximum V1.

4. Choose a suitable, single value of V1 between the limits determined in Step 3 above. The speed selected must be greater than VMCG, and less than the VR and VMBE determined in Steps 11 and 12 of the Normal Analysis procedure.
MAXIMUM ALLOWABLE TAKEOFF WEIGHT

VL LESS THAN VMCG

Usually this will only occur when the takeoff weight is not field-length limited:

Takeoff Speeds chart - Read V1 at the field-length limited gross weight obtained from Step 4 of Normal Analysis procedure, and at the V1/VR ratio obtained from Step 3 of Normal Analysis. Use a V1 between VMCG and the maximum V1 so determined, but not exceeding VR.

VL GREATER THAN VMBE

If the minimum V1, as determined by Steps 1 through 3 under SELECTION OF V1, still exceeds the brake energy speed, the takeoff weight will have to be reduced:

Maximum Brake Energy Limit Speed chart - Determine ΔV1, as defined on the chart. Reduce gross weight as indicated and redetermine VR, V1 and V2 for the lower weight.

VR LESS THAN VMCA LIMIT

On this airplane, VMCG is greater than 1.05 VMCA; therefore, minimum VR is equal to VMCG. This implies that the available runway length must be sufficient to permit a V1 equal to VR. This condition will be satisfied if the slope - and wind-corrected accelerate-stop distance exceeds 4500 feet on the Runway Length and V1 Adjustments chart.

OPERATION WITH ONE AIR CONDITIONING PACK ON

Performance limited takeoff weights and takeoff climb gradients available can be increased by scheduling takeoff with only one air conditioning pack on. Performance increments are shown on the Takeoff With 1 A/C Pack On chart; these increments are applied to the weights and gradients obtained in steps 4, 5 and 7 of the Normal Takeoff Weight Analysis procedure. The gradient increment must be subtracted from the gradient required for use in step 10 to find an obstacle limited weight.
MAXIMUM ALLOWABLE TAKEOFF WEIGHT

SLIPPERY RUNWAY

Wet or icy runway conditions have the same effect on takeoff field length requirements as an inoperative anti-skid system; namely, reduced braking effectiveness. This makes the one engine inoperative condition critical.

The operator must review the existing runway conditions, decide how much additional stopping distance is desired, then proceed with the following steps to determine the changes in the field length limited takeoff weight, and V1 speed:

1. Omit Step 2 of the Normal Takeoff Weight Analysis procedure.

2. Runway Length and V1 Adjustments chart - Determine V1/VR as usual, except that the accelerate-stop distance must be reduced by an arbitrary amount equal to the desired extra stopping margin before making corrections for slope and wind. At the intersection of the corrected takeoff distance available and corrected accelerate-stop distance lines in the "web" portion of the chart, read corrected runway length and V1/VR.

3. Maximum Takeoff Weight chart - Enter the appropriate chart with the corrected runway length from Step 2 above. Read gross weight.

NOTE: If the V1, determined by using the V1/VR ratio from Step 2 above, is greater than the minimum V1 as determined under SELECTION OF V1, no reduction in field length limited takeoff weight is necessary. In fact, the use of minimum V1 in this case will give a greater stopping distance margin.
MAXIMUM ALLOWABLE TAKEOFF WEIGHT

EXAMPLES

Example lines shown on the charts do not necessarily reflect these examples.

ILLUSTRATIVE EXAMPLE 1

Given: Airport Conditions

Runway Length Available = 12,000 ft.
Runway Slope = 0.5% (Uphill)
Airport Elevation (Pressure Altitude) = 2000 ft.
Obstacles: 120 ft. high at 3400 ft.
500 ft. high at 25,000 ft.

Atmospheric Conditions

Reported Wind (at 50 ft.) = 10 kt. (Headwind)
OAT = 20°C (68°F)

Airplane Conditions

Takeoff flap position 10

This example illustrates the general use of the charts, and the obstacle clearance charts in particular, and is restricted to normal speed operation.

See example 5 for improved obstacle clearance with speed increase method.

See chart reading procedure for chart titles and methods of use.

1. From Runway Length Corrections (All Engines) chart, corrected runway length = 12,070 ft.
2. From Runway Length and V1 Adjustments chart, corrected runway length = 12,570 ft. and V1/VR = .970
3. From Maximum Takeoff Weight (Field Length Limits) chart, using the lesser of the corrected runway length from Steps 1 and 2, field length limited takeoff weight = 683,000 lb.
4. From Maximum Takeoff Weight (Climb Limits) chart, climb limited takeoff weight = 649,000 lb.
5. From Maximum Takeoff Weight (Tire Speed Limits) chart, 200 MPH tire speed limited takeoff weight = 743,000 lb. Maximum Takeoff weight = 649,000 lb.
6. From Takeoff Climb chart, using field length limited weight of 683,000 lb, second segment gross gradient at 400 ft. = 2.30%

Using Gradient Corrections chart, gradient corrected for wind = 2.37%

(Continued)
ILLUSTRATIVE EXAMPLE 1 (Continued)

7. From Close-in Obstacle Clearance chart, second segment gross gradient required for close-in obstacle clearance = 4.50%. Obstacle is not cleared with gradient available (uncorrected for wind). Change in reference zero = 5100 - 3400 = 1700 ft. due to the decrease in field length required for the reduced takeoff weight.

In this distance, reference zero shifts 1700 x .005 = 8.5 ft. (vertical) due to runway slope.

The obstacle height is now 128.5 ft. at 5100 ft. from reference zero and the required gradient is 3.08%.

8. From Distance Obstacle Clearance chart determine gross gradient required and minimum gross height.

Gradient capability (from Step 6) = 3.08% (corrected for wind = 3.17%) Corrected obstacle distance = 25,000 + 1700 = 26,700 ft.
Required gross gradient = 2.89% (less than gradient available). Distance from reference zero to reach obstacle height of 500 ft. is 23,300 ft.
Minimum gross height for flap retraction is 770 ft.

9. From Takeoff Climb chart, obstacle limited weight = 638,500 lb. at a gross gradient of 3.08%.

10. From Distance Obstacle Clearance chart, maximum level-off height is 940 ft. Therefore extended final segment climb is unnecessary in this case.

11. From Gross Height - Pressure Altitude Conversion chart, the pressure altitude increment for a gross height of 770 ft. is 750 ft. Therefore, the minimum pressure altitude for level-off and flap retraction is 2750 ft.

12. From Takeoff Speeds chart for $V_1/V_R = .970$ and 638,500 lb.

\[ V_R = 152.5 \text{ kt.} \quad V_1 = 147.5 \text{ kt.} \quad V_2 = 159 \text{ kt.} \]

13. Determine flap retraction and final climb speeds.
NOTE

HIGHER WEIGHTS ARE ALLOWED WITH IMPROVED CLIMB PERFORMANCE. SEE TEXT.
NOTE: ENTER CHART WITH TAKEOFF CLIMB GRADIENT CORRECTED FOR WIND AND LEVEL-OFF HEIGHT ABOVE 400 FT.
AIRPLANE FLIGHT MANUAL

PERFORMANCE
ENROUTE CLIMB WEIGHS
FOR POSITIVE MEI GRADIENT

ENGINE INOPERATIVE

FOR ENROUTE CLIMB SPEED SCHEDULE
SEE ENROUTE CLIMB, ENGINE
INOERATIVE CHART.

FAA APPROVED 12-30-69

BOEING 747

MC LAUGHLIN 12-8-69
CHART APPLICABLE TO
JT9D-3

1. ANTI-ICE OFF
2. NACELLE ANTI-ICE ON
3. WING & NACELLE ANTI-ICE ON

ANTI-ICE
A/C PACKS ON

GROSS WEIGHT - 1000 LB

FAA APPROVED 12-30-69

BOEING No. D6-30643
Vol. II
SECT PAGE 15.0-53
2 ENGINES INOPERATIVE

1 A/C PACK ON

TEMPERATURE RELATION TO 10A

400 440 480 520 560 600 640
GROSS WEIGHT — 1000 LB

1 ANTI-ICE OFF
2 NACELLE ANTI-ICE ON
3 WING & NACELLE ANTI-ICE ON

FAA APPROVED 12-30-69
APPROACH AND LANDING

Charts on the following pages present approach and landing gradients, maximum landing weights as limited by approach and landing performance, landing field length requirements, and landing weights for the maximum brake energy at which wheel thermal plugs will remain intact.

The speed schedules shown on the Approach and Landing Climb charts are those at which the gradients were calculated in accordance with FAR 25.121(d) and 25.119, and have no operational significance. The minimum landing approach speeds are shown on the Landing Field Length and Speed charts.

The ICE CORRECTION on the charts accounts for maximum probable performance effects of ice remaining on the airplane surfaces without anti-ice protection. The correction applies when operating in icing conditions during any part of the flight, unless the forecast temperature at the destination airport is high enough (above 8°C or 46°F) to ensure that the ice will melt off prior to approach and landing.

The Maximum Landing Weight, Climb Limits, can be increased for one air conditioning pack on operation. This information is included to permit a showing of compliance with FAR 25.1001 (c) at the increased allowable takeoff weight.

This information may also be used to schedule higher allowable landing weights provided approach and landing procedures are modified to have only one air conditioning pack on prior to the point where a go-around might be initiated. See Normal Procedures.
REDUCE GRADIENT BY 0.5% FOR ICE ACCUMULATION WHEN OPERATING IN ICING CONDITIONS DURING ANY PART OF THE FLIGHT WITH FORECAST LANDING TEMPERATURE BELOW 8°C.
AIRPLANE FLIGHT MANUAL

PERFORMANCE

MAXIMUM LANDING WEIGHT

CLIMB LIMITS

APPROACH FLAP POS. 20
LANDING FLAP POS. 28

NOTE: USE ICE CORRECTION WHEN OPERATING IN ICING CONDITIONS DURING ANY PART OF THE FLIGHT WITH FORECAST LANDING TEMPERATURE BELOW 0°C.

CORRECTION FOR A/C PACK OPERATION

WEIGHT INCREASE 1000 LBS.

PRESSURE ALTITUDE 1000 FT.

FAA APPROVED 6-30-70
The buffet frequency was estimated for the conditions listed in the table on Page 16.0-2. These conditions include stall buffet for each flap detent and compressibility buffet for $M_D$ and $V_D$ conditions. From this investigation it was concluded that:

Average buffet frequency $= 3.0 - 3.5$ CPS

Buffet amplitude range $= \pm 1g - \pm 6g$

For illustration, flight data is attached for the following five conditions.

<table>
<thead>
<tr>
<th>COND.</th>
<th>$V_c$ ~KT INITIAL-BUFFET</th>
<th>ALTITUDE ~1000FT</th>
<th>G.W ~1000LB</th>
<th>FLAPS/GEAR</th>
<th>C.G. ~%MAC</th>
<th>PAGE</th>
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<tbody>
<tr>
<td>$M_D$</td>
<td>--</td>
<td>32</td>
<td>464</td>
<td>UP/UP</td>
<td>27</td>
<td>16.0-3-4</td>
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<tr>
<td>$V_D$</td>
<td>447</td>
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<td>490</td>
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<td>25</td>
<td>16.0-5-6</td>
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<td>CLEAN STALL</td>
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<td>18.7</td>
<td>552</td>
<td>UP/UP</td>
<td>32</td>
<td>16.0-7-8</td>
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<td>STALL</td>
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<td>536</td>
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<td>32</td>
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<td>16.0-11-12</td>
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<td>Vc</td>
<td>ALTITUDE</td>
<td>G W</td>
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<td>C. G.</td>
<td>BUFFET FREQ</td>
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<td>~1000 ft</td>
<td>~1000 lb</td>
<td>/ /</td>
<td>~% MAC</td>
<td>~ CPS</td>
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<td>V₀</td>
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<td>UP / UP</td>
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<td>3.1 - 3.3</td>
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<tr>
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<tr>
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<td>137</td>
<td>16.5</td>
<td>538</td>
<td>5 / UP</td>
<td>32</td>
<td>3.0 - 4⁺⁺</td>
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<tr>
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<td>128</td>
<td>12.0</td>
<td>536</td>
<td>10 / UP</td>
<td>32</td>
<td>3.0 - 4⁺⁺</td>
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<td>2.8 - 3.5</td>
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<td>25 / DOWN</td>
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<td>525</td>
<td>30 / DOWN</td>
<td>33.3</td>
<td>2.8 - 4⁺⁺</td>
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</table>

**INITIAL BUFFET = ± 1g → ± 50 ± 75 g**

**REFERENCE:** DG-30642 "747-100 SIMULATOR FLIGHT DECK VIBRATION ENVIRONMENT"
APPENDIX C - AUTOThROTTLE

The block diagram on Page 17.0-2 is a simplified autothrottle block diagram for use in small perturbation 747 control simulation. This data is provided for information purposes and was not incorporated in the simulation.
18.0  **APPENDIX D - AUTOPILOT**

This section contains a description of the automatic pilot, flight director, yaw damper, and autothrottle systems for the 747 aircraft. This data is provided for information purposes. The autopilot and autothrottle were not incorporated in the simulation.
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SUMMARY

This document provides a description of the Automatic Pilot, Flight Director, Yaw Damper, and Autothrottle Systems for the 747 aircraft. It contains block diagrams, pictorials, and tables to describe operation of each of these systems as they are planned for incorporation in the aircraft.

Two autopilots give the flight crew a choice of fail-safe systems, each of which can provide all manual and path modes except LAND. Both autopilot channels are used to give "fail passive" control at low altitudes during the automatic landing sequence. Two separate dual-channel automatic pilot pitch trim systems and two yaw dampers give a high level of system integrity.
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7 Autopilot Pitch Axis ........................................ 45c
I. 747 INTEGRATED AUTOPILOT AND FLIGHT DIRECTOR

A. GENERAL

The Sperry SPZ-1 integrated autopilot flight director system was developed for the Boeing 747. The autopilot is used for path or manual control via the pitch wheel and turn knob.* Flight director commands are provided and may be used to monitor autopilot operation, or to be tracked with the autopilot manual mode, or to be tracked manually.*

The basic 747 is equipped with a dual autopilot system and a triple flight director system. Each pitch or roll computer contains all of the circuitry necessary to compute the respective pitch or roll autopilot and flight director commands. Each autopilot or flight director channel consists of two computers: one roll, and one pitch.

The general autopilot flight director system layout and interconnects used on the basic Boeing 747 are shown in Figure 1. A dual set of sensors provide navigational and performance signals to the autopilot flight director computers.

Only autopilot computers of channel A and B are connected to hydraulic servos. Computer C is used only for computing flight director commands and cannot be switched in as an autopilot. The number 1 sensor group serves both the A and C computers while sensor group 2 supplies information to the B computers.

Instrument switching allows either the pilot or co-pilot to select any one of the three flight director computer signals for display on his ADI.

The autopilot can be used as a single-channel system in the navigation, cruise, and manual modes of operation. Either channel A or B can be selected for these single-channel modes.

For automatic landing only, fail passive operation is obtained with the dual channel system. Both A and B autopilot channels must be engaged for this mode of operation.

The cockpit location of the controls and indicators for the autopilot/Flight Director, as well as the other systems of the IEFCS, are shown in Figure 1A.

*Maneuvering control for certain airlines will be via control wheel steering instead of turn and pitch knob.
B. SYSTEM INTRODUCTION

All components associated with each flight axis of each channel are packaged in separate computer units. The elements which comprise a basic autopilot/flight director system are:

1) One mode selector panel
2) One flight controller or optionally 3 force sensors.
3) Two flight mode annunciators
4) One monitor and logic unit
5) Three accessory boxes
6) Three roll computers
7) Three pitch computers
8) One automatic stabilizer trim unit

1. Mode Select Panel

The mode select panel contains the switches and logic for mode selection and control of all the autopilot/flight director computer channels. Figure 2 shows the mode select panel. The controls shown shaded in are for optional modes.

The engage switches are solenoid held in MAN and COMMAND with locking provisions at OFF. Each engage switch controls both the pitch and roll computer associated with that channel. Either channel A or B may be engaged in Manual or Command by choice of engage switches. Once one engage switch has been placed into the Manual or Command position, the other switch is locked off and cannot be moved from the OFF position except when LAND has been selected and a monitor check has been satisfactorily completed.

The course select switch may be placed in either the course 1 or course 2 position without regard to which channel of the autopilot has been selected. It is solenoid held in both positions. The switch drops to the dual position when the LAND mode has been selected. Course 1 position feeds the No. 1 VOR/LOC receiver output and No. 1 course error signal to all three A/P - F/D roll computers. and course 2 does likewise for the No. 2 VOR/LOC and No. 2 course error. In the dual position, LOC No. 1 and Course error No. 1 are fed to computers A and C while LOC No. 2 and course error No. 2 are fed to the B computer. Figure 3 is a pictorial diagram of the above switching functions and indicates the isolation and independence achieved for single and dual channel modes of autopilot operation.
FIG. 2a

MODE SELECT PANEL

(WITH CONTROL WHEEL STEERING AND MACH HOLD OPTION)
Heading select is accomplished for all three channels through a single set knob which positions two synchros fed from two separate magnetic heading reference units to maintain heading isolation.

The lateral navigation modes (INS, HDG, VOR/LOC), as well as the landing approach control modes ILS and LAND, are selected by the main mode selector switch. This is a simple rotary type switch without holding or centering devices. A second selector switch, also rotary type, allows selection of the Turbulence mode, V/S (vertical speed), IAS hold, or Mach hold. This second switch is solenoid held in each mode select position. The V/S and IAS modes may be selected alone or during the arm phases of ALT. Select, ILS, and LAND. When used during the armed phases of another mode, the switch will drop to OFF when the mode goes into the capture phase.

The switch will not hold in Turb. if the autopilot is not engaged. Also, if the switch is moved to Turb while the altitude select is engaged, the altitude select switch will drop to OFF. Further, if Turb. is engaged and ALT hold is selected, the Turb. switch will drop to OFF.

The Back Beam switch is designed to mechanically preclude accidental turn on and is solenoid held in the ON position. It will hold only if VOR/LOC is selected on the mode selector switch. Also available on the mode select panel are the ON/OFF switches and pitch trim controls for flight directors.

The mode selection and mode compatibility interlocks related to the position of the switches on the Mode Select Panel are shown in Table 1. (Pitch channel) and Table 2 (Roll channel) for both the autopilot and the Flight Director.

2. Flight Controller

The Flight Controller which contains the autopilot pitch wheels and turn knob is shown in Figure 4. The pitch wheel and turn knob provide attitude commands proportional to their respective rotations. The outputs for channels A and B of the autopilot are electrically independent. The potentiometers used with the pitch wheels are unclutched electrically from the wheels during all pitch path modes.

For customers who order the control wheel steering option the flight controller is not used. Instead three force transducers shown in Figure 4a are provided to maneuver the aircraft via the normal control column and control wheel with autopilot engaged.
# Autopilot-Flight Director Mode Chart - Pitch Modes

<table>
<thead>
<tr>
<th>PITCH MODES</th>
<th>ALL OFF</th>
<th>ALT HOLD</th>
<th>ALT SELECT</th>
<th>IAS HOLD</th>
<th>MACH HOLD</th>
<th>VERT SPEED</th>
<th>ILS (GLIDE SLOPE)</th>
<th>LAND (GS &amp; FLARE)</th>
<th>G/A ATTITUDE</th>
<th>TURBULENCE</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>A/P FD</td>
<td>A/P FD</td>
<td>A/P FD</td>
<td>A/P FD</td>
<td>A/P FD</td>
<td>A/P FD</td>
<td>A/P FD</td>
<td>A/P FD</td>
<td>A/P FD</td>
<td>A/P FD</td>
</tr>
<tr>
<td>COMMAND</td>
<td>PW</td>
<td>PTW</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>P</td>
<td>P</td>
<td></td>
<td>NOT APPL</td>
</tr>
<tr>
<td>MANUAL</td>
<td>PW</td>
<td>PTW</td>
<td>X</td>
<td>X</td>
<td>PW</td>
<td>PW</td>
<td>ALT HOLD PTW</td>
<td>ALT HOLD PTW</td>
<td></td>
<td>NOT APPL</td>
</tr>
<tr>
<td>OFF</td>
<td>O</td>
<td>PTW</td>
<td>O</td>
<td>X</td>
<td>O</td>
<td>O</td>
<td>P</td>
<td>P</td>
<td></td>
<td>PW PTW</td>
</tr>
</tbody>
</table>

- **O** NOT ENGAGED
- **X** ENGAGED OR OPERATIVE
- **PW** PITCH CONTROL WHEEL (OR CWS)
- **PTW** PITCH TRIM WHEEL

- [ ] SELECTED COMPATIBLE MODE, THAT IS: IAS HOLD, MACH HOLD, VERT SPEED, ILS ARMED OR LAND ARMED.
- [ ] SELECTED COMPATIBLE MODE, THAT IS: ALT HOLD, ALT SEL, IAS HOLD, OR VERT SPEED.
- [ ] NOT CERTIFIED BEYOND CATEGORY II.
- [ ] AUTOPilot ENGAGE switch will not hold in this position.
- [ ] A/P WARNING LIGHT "FLASHING AMBER" UNLESS BOTH A/P ENGAGE SWITCHES ARE IN "COMMAND" POSITION.

---

Table 1  Autopilot/Flight Director Mode Chart - Pitch Modes
## Autopilot-Flight Director Mode Chart - Roll Modes

<table>
<thead>
<tr>
<th>Roll Modes</th>
<th>INS</th>
<th>HDG (SELECT)</th>
<th>VOR/Loc</th>
<th>VOR/LOC Back Beam (Option)</th>
<th>ILS (LOC)</th>
<th>LAND (LOC)</th>
<th>G/A (Wing Level)</th>
<th>Turbulence (Not Available in ILS or Land Modes)</th>
</tr>
</thead>
<tbody>
<tr>
<td>A/P Engage Switch</td>
<td>A/P Engaged</td>
<td>A/P Capture</td>
<td>A/P Engaged</td>
<td>A/P Capture</td>
<td>A/P Engaged</td>
<td>A/P Capture</td>
<td>A/P Engaged</td>
<td>A/P Capture</td>
</tr>
<tr>
<td>Command</td>
<td>HDG SEL</td>
<td>HDG SEL</td>
<td>HDG SEL</td>
<td>X</td>
<td>X</td>
<td>NOT APPL</td>
<td>HDG SEL</td>
<td>X</td>
</tr>
<tr>
<td>Manual</td>
<td>TK HDG SEL</td>
<td>TK X</td>
<td>TK HDG SEL</td>
<td>TK</td>
<td>X</td>
<td>NOT APPL</td>
<td>TK HDG SEL</td>
<td>TK X</td>
</tr>
<tr>
<td>Off</td>
<td>0 HDG SEL</td>
<td>0 X</td>
<td>0 X</td>
<td>0 X</td>
<td>0 X</td>
<td>HDG SEL</td>
<td>0 X</td>
<td>0 X</td>
</tr>
</tbody>
</table>

- **O** NOT ENGAGED
- **X** ENGAGED OR OPERATIVE
- **TK** TURN CONTROL KNOB (OR CWS)
- **<** NOT CERTIFIED BEYOND CATEGORY II.
- **A/P** WARNING LIGHT "FLASHING AMBER" UNLESS BOTH A/P ENGAGE SWITCHES ARE IN "COMMAND" POSITION.
- **<** AUTOPILOT ENGAGE SWITCH WILL NOT HOLD IN THIS POSITION.
- **<** TURB SWITCH WILL NOT HOLD.

### Table 2 Autopilot/Flight Director Mode Chart - Roll Modes
Fig. 4. Flight Controller
CONTROL WHEEL STEERING
FORCE TRANSUCER

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3. Flight Mode Annunciation

Dual flight mode annunciators are provided on the 747. Flight director modes and autopilot modes are displayed side by side on each annunciator panel as shown in Figure 4a.

The basic modes annunciated are:

- ALT SEL (Altitude Select)
- NAV (Navigation)
- GS (Glide Slope)
- FLARE
- GO AROUND

In each mode the annunciator displays an amber light when the particular mode is armed and switches to a green light when the mode is initiated. Table 3 shows each autopilot and Flight Director mode as it will be displayed on the mode annunciator.

Also displayed on the annunciator panel are the autopilot and autothrottle warning and disengage lights.

A press-to-test feature is included in the flight mode annunciator. All amber lights are tested by depressing the left hand section of the panel. All green lights as well as the red warning lights of the autopilot and autothrottle are tested by depressing the right hand section of the panel.

4. Monitor and Logic Unit

The monitor and logic unit is a separate package which contains much of the autopilot/Flight Director engage interlock logic, dual-channel monitoring logic, mode annunciation logic, and the warning light circuits. The physical and electrical arrangement of the monitor and logic unit provides complete channel isolation.

5. Accessory Boxes

The accessory boxes are Boeing supplied units which provide the switching and interconnect functions necessary to interface the autopilot and flight director systems to other airplane systems.

6. Pitch and Roll Computers

The A/P - F/D pitch and roll computers include the computing and logic circuitry necessary to receive data from the aircraft sensors and produce autopilot servo and flight director commands for all modes of operation.
FLIGHT MODE ANNUNCIATOR

FLIGHT DIRECTOR MODES

AUTOPILOT MODES

BASIC

WITH AUTOthROTTLE OPTION

Figure 46
FLIGHT MODE ANNUNCIATOR
<table>
<thead>
<tr>
<th>MODE</th>
<th>AUTOPILOT AND FLIGHT DIRECTOR</th>
<th>FLIGHT MODE ANNUNCIATOR</th>
<th>CONDITION</th>
</tr>
</thead>
<tbody>
<tr>
<td>LOC</td>
<td>ARM</td>
<td>AMBER GREEN</td>
<td>ARM CAPTURE &amp; CONTROL</td>
</tr>
<tr>
<td>VOR</td>
<td>ARM</td>
<td>AMBER GREEN</td>
<td>ARM CAPTURE &amp; CONTROL</td>
</tr>
<tr>
<td>INS</td>
<td>ARM</td>
<td>AMBER GREEN</td>
<td>ARM CAPTURE &amp; CONTROL</td>
</tr>
<tr>
<td>ILS</td>
<td>ARM</td>
<td>AMBER GREEN</td>
<td>ARM CAPTURE &amp; CONTROL</td>
</tr>
<tr>
<td>LAND</td>
<td>ARM</td>
<td>AMBER GREEN</td>
<td>ARM CAPTURE &amp; CONTROL</td>
</tr>
<tr>
<td>ALT. SEL.</td>
<td>ARM</td>
<td>AMBER GREEN</td>
<td>ARM CAPTURE &amp; CONTROL</td>
</tr>
<tr>
<td>GO AROUND</td>
<td>ARM</td>
<td>AMBER GREEN</td>
<td>ARM CAPTURE &amp; CONTROL</td>
</tr>
<tr>
<td>F/D ONLY</td>
<td>X</td>
<td>AMBER GREEN</td>
<td>X</td>
</tr>
<tr>
<td></td>
<td>X</td>
<td>FLARE</td>
<td>X</td>
</tr>
<tr>
<td></td>
<td>X</td>
<td>USE</td>
<td>X</td>
</tr>
<tr>
<td></td>
<td>X</td>
<td>LC</td>
<td>X</td>
</tr>
</tbody>
</table>

Table 3 Flight Mode Annunciator Chart
7. **Automatic Stabilizer Trim Unit**

The automatic stabilizer trim unit (ASTU) contains the computational circuitry necessary to provide automatic stabilizer trim whenever the autopilot is engaged. The ASTU contains two independent self monitored trim channels. One channel provides trim operation while the other is in standby. Automatic transfer is provided to the standby channel in the event that the trim monitor detects a malfunction.

8. **Self-Test and Maintenance Monitoring**

Self-test of the pitch and roll computers, the automatic stabilizer trim unit and the monitor and logic unit is performed with a go/no-go readout by means of the Built-In-Test-Equipment (BITE).

The BITE switches and lights, located on the front panels of each of the above units, permit the rapid isolation of a faulty unit, while the complete system is installed on the airplane.

A typical BITE test is shown in Figure 4b. The BITE sets up input signals and sensors to adjust two or more signal path gains in the unit under test (UUT) as required to achieve a null summation of the signal path outputs, if all circuits are normal. The null condition is sensed by BITE null detection logic. Simultaneously, specific portions of the unit logic are addressed by BITE. The status of the unit logic circuits and the output of the BITE signal null detector are combined in BITE logic to produce a go/no-go output to the BITE readout lights.

Maintenance monitoring has been included in the system as a basic feature to help in isolating the cause of autopilot warning or disengagement during dual-channel autopilot operation.

The maintenance monitoring circuits and readouts are included in the Monitor and Logic Unit. The readout is located in the top portion of the Monitor and Logic Unit front panel, above the BITE readout lights. The maintenance monitor readout consists of four latching indicators which trip to indicate that autopilot warning or disengagement occurred for one of the following reasons:

1. Power loss to channel A (single or dual channel operation)
2. Power loss to channel B (single or dual channel operation)
3. Pitch channel camout monitoring trip (dual channel operation)
4. Roll channel camout monitoring trip (dual channel operation)

The above faults are monitored and displayed permanently by the latched indicators, until these are manually reset.
NOTE: UUT = UNIT UNDER TEST

FIG. 4b - TYPICAL BITE TEST

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9. **ILS Deviation Warning System** (Optional)

The Deviation Warning System is a dual system, monitoring the Captain's and First Officer's navigation receiver outputs. (See Figure 4d)

The system warns the pilots when the outputs of the navigation receivers exceed 20 \( \mu \)A from the localizer beam centerline or 75 \( \mu \)A from the Glide Slope beam centerline with a delay time of 2.2 seconds. The warning system is in operation when the autopilot is engaged in the ILS or LAND mode and the radio altitude is below 500 feet altitude. From 500 feet to 200 feet the system provides a warning if the receiver output exceeds either GS or LOC thresholds. If the nav. receiver signal causing the warning is reduced below the detection threshold, the warning light will be turned off. However, below 200 feet altitude the system becomes latching and once the warning system has been tripped the warning light will remain on even though the signal error has been reduced below the threshold. Below Flare altitude (53 feet) the warning system monitors only LOC deviation signal errors.

The two monitor systems are independent. However, to provide greater redundancy the deviation warning signals are cross fed such that if one monitor is tripped it will also switch on the warning light driver stage of the second monitor, thus illuminating all warning lamps associated with the ILS deviation warning system.

Confidence tests of the system can be performed either on the ground or in flight. The confidence tests are pilot activated and monitored. The tests are activated by proper mode selection and engagement of the A/P and F/D and the deflection of the VHF NAV test switch.
ILS DEVIATION WARNING
SYSTEM BLOCK DIAGRAM

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Page 21d
C. DESCRIPTION OF THE AUTOPILOT ACTUATOR SYSTEM

When the autopilot is engaged, autopilot commands are coupled into the primary flight control system via parallel servo actuators. Thus, the control wheel and column, as well as the control surfaces, move in response to autopilot commands.

The lateral control system of the 747 utilizes a pair of hydraulic central control actuators which control the hydraulic-powered surface actuators. These central control actuators accept commands from either the autopilot or the manual control system. During all manual or autopilot operation, the two central control actuators are slaved together through a cross link.

The 747 lateral autopilot servo system is mechanized as shown in Figure 5. The pitch axis autopilot servo mechanization is similar in concept to the lateral system.

The mechanization of the servo makes the system fail safe for all single-channel operation and fail passive for all dual-channel (LAND) operation.

1. Single-Channel Operation

Single-channel autopilot operation is used in all modes except LAND. In single-channel operation, either the A channel or B channel autopilot may be selected. The autopilot drives an autopilot servo actuator integral with the central control package. The servo actuator output displacements are proportional to the autopilot command signals. The autopilot actuator drives the manual controls via a force detent. When either autopilot is engaged, both central control packages are driven from the engaged autopilot actuator.

Autopilot authority in the lateral axis is stroke limited to the equivalent of twenty-five degrees of control wheel displacement. In the pitch axis, authority is limited by reacting the force detent against the manual feel pressure system.

The pilot can overpower the autopilot at any time by applying approximately fourteen pounds at the control wheel in the lateral axis and about twenty-seven pounds in the longitudinal axis.
2. **Engage Synchronization**

Synchronization loops are provided to eliminate engage transients.

When the autopilot is disengaged, the autopilot actuator is hydraulically de-energized and caged to the null position. The synchronizer loop around the servo amplifier holds the servo amplifier output near null. As soon as the autopilot is engaged, hydraulic pressure builds up first in the force detent mechanism moving the autopilot actuator, which is still not pressurized, to a position matching that of the control wheel. Since the autopilot LVDT is active, the voltage resulting from a change in autopilot actuator position will be fed back to the servo amplifier and synchronized within 0.25 sec.

Additional loops are included in the roll autopilot and the pitch autopilot, to synchronize the attitude commands.

3. **Dual-Channel Operation**

During dual-channel operation, both autopilot actuators are coupled to the manual controls via their respective force detents as shown in Figure 5. The force detents are mechanized to give the characteristics shown in Figure 6a.

To reach equilibrium, the forces applied to the cross link must sum to zero. The forces applied to the cross link are from the two force detents, feel system and control system friction. Since the initial force gradients of the detents are steep, disagreement between the two autopilot actuator positions of more than approximately 0.5° aileron will cause one or both of the detents to reach its maximum force level.

If one channel fails hardover, the second channel and the feel system will keep airplane controls in the trim position. If the second channel should command in the same direction as the hardover, the surfaces will correctly respond to the second channel as illustrated in Figure 6b.

Except in the case of rather precise agreement between autopilot channels, the resultant dual-channel autopilot command is the lesser of the two autopilot commands. Several pertinent points can be concluded:

(a) A hardover command results in a passive failure with negligible surfaces deflection.

(b) If the two autopilot commands are opposing, the output is zero and the airplane remains in trim.
APPROXIMATELY FLAT FORCE GRADIENT

\[ \Delta \delta_{\text{cmd}} = (\delta_{\text{cmd auto}} - \delta_{\text{cmd cross shift}}) \]

FIG. 6a GENERAL FORCE DETENT CHARACTERISTICS

FIG. 6b GENERAL OPERATING CHARACTERISTICS OF DUAL CHANNEL SERVO SYSTEM

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(c) During normal operation, the airplane will track the autopilot command having the lesser value. If one autopilot fails passive, the resultant output is nearly zero. Thus, the dual-channel servo actuator system provides true "Fail Passive" operation for use in the LAND mode.

D. FLIGHT DIRECTOR OPERATION AND MODES

The flight director and autopilot computation paths become separate just prior to the autopilot path integrator as shown in the roll axis and pitch axis computer block diagrams. Command and rate limits are incorporated into the separate autopilot and flight director computer circuits. These limits are switched as a function of the mode selected and the associated submodes. The Flight Director system characteristics are summarized in Table 5 on page 38 for the roll axis and Table 6 on page 45a for the pitch axis.

As shown in the autopilot-Flight Director mode charts (Tables 1 and 2 on pages 17 and 18), the flight director computes and displays the navigational and vertical path data even when the autopilot engage switches are in the OFF position. When the autopilot is off, the pilots can fly the displayed flight director commands using the primary flight controls. When the autopilot is engaged in MANUAL, the flight director commands can be followed using the pitch wheel and turn knob. All modes of the autopilot except turbulence are also provided for the flight director.

There are two control modes which are exclusively flight director modes. These are go-around and back beam (optional). Go-around provides a wings leveling command in the lateral axis and a fixed pitch attitude climb command in the pitch axis. The go-around mode is initiated by the pilot operating either of two go-around switches located on the inboard throttle levers. Operating a go-around switch will cause the autopilot engage switch to drop to the OFF position from either MANUAL or COMMAND.

The back beam mode is initiated by selecting the VOR/LOC mode, then placing the solenoid held back beam switch in ON. In this mode, the flight director provides localizer back beam intercept, capture and track commands which can be flown by the pilot using either the turn knob with the autopilot in MANUAL, or the primary flight controls with the autopilot off. During back beam, any one of several pitch control modes can be chosen to provide flight director pitch commands.
E. ROLL AUTOPILOT

The lateral autopilot computer block diagram is shown in Figure 11 on Page 42. Gains, transfer functions, special gain programs, mode engage and switching logic are summarized in Table 4 and Figures 8, 9 and 10 on Page 37 and following. Finally, a block diagram of the lateral flight control system is shown in Figure 12 on Page 43.

The roll attitude and rate loops (see Figure 11) are basic for all modes of operation. These loops provide both roll mode damping and spiral mode stabilization.

The rate signals are sensed by the gyros installed in each roll autopilot computer. The roll attitude signals are from the Inertial Navigation System. Fixed attitude and rate gains of

\[
\frac{\Delta x}{\phi} = 3.2 \text{ Deg} \quad \text{and} \quad \frac{\Delta x}{\dot{\phi}} = 3.6 \text{ Deg/Sec}
\]

When engaged in MANUAL, the roll autopilot responds to bank commands inserted via the turn knob. When zero bank is commanded (turn knob in detent), wings leveling occurs after which the autopilot holds airplane heading. When engaged in COMMAND, the pilot has the option of control by any of the following modes: Heading Select, VOR/Localizer, INS, ILS, or LAND.

Prior to the autopilot engagement, either in the single channel or the dual channel mode, synchronization loops operate to eliminate autopilot engage transients.

1. Engage Synchronization

In addition to the servo amplifier output synchronization mentioned earlier (See page 24), a second loop (command synchronizer) holds the servo amplifier input near zero by synchronizing the attitude command, prior to autopilot engagement in any single channel mode.

A third loop, used only for the dual channel mode, synchronizes the attitude command by nulling the servo amplifier input through feedback to the lateral path integrator of the channel yet to be engaged. This loop operates only after both autopilot switches are in the command position and until the second autopilot channel becomes engaged. During the operation of this loop, the previously mentioned command synchronizer loop is inhibited.


The manual mode is engaged by placing the autopilot engage switch in the MANUAL position. The autopilot cannot be engaged with the
turn knob out of detent. When the turn knob is in its center detent position, wings leveling occurs and the autopilot flies to hold heading. The heading reference is established by the Magnetic Heading Reference Unit (MHRU).

The turn knob produces a bank angle command proportional to turn knob displacement. The turn knob output is generated by a shaped pot with a dead-zone in the center equivalent to ±16° of knob rotation. The maximum range of rotation of the turn knob is ±28° which commands ±30° bank angle. Maximum roll rate for the Manual Mode is ±4°/sec. Provisions are available for a 7 degrees/second rate limit.

When the magnetic heading clutch in the MHRU is engaged, the heading error synchro is clutched to a magnetic heading repeater and provides a heading error signal. When the turn knob is in detent, the clutch automatically engages.

Proportional and integral heading error signals produce the bank angle command necessary to maintain the airplane heading. The integral of heading error reduces heading errors to zero in the presence of thrust asymmetry or other lateral control system mis-trim. Both gains are scheduled as a function of true airspeed to maintain consistent system performance throughout the flight regime.

Certain airline customers will have control wheel steering incorporated instead of a turn knob. The control wheel steering option will be usable at all times when the autopilot is engaged in Manual Mode. In addition, when the autopilot is engaged in "Command" roll control wheel steering is available in the arm phase of lateral path modes such as LOC, VOR, or INS. Thus the control wheel steering will be available to establish the intercept angle desired prior to the capture maneuver on these modes. A special switch is included on the Mode Select Panel as shown in Figure 2a. This switch allows either CWS or Heading Select to be used for the above mentioned arm phases.

The roll control wheel steering block diagram is shown in figure lla. Force on the control wheel commands roll rate via an integral path. The integrator is bypassed with a displacement or "boost" path to minimize the velocity error or overshoot which results when force is abruptly removed from the wheel. Constant CWS gains are used over the speed range of the 747 so that light feel forces are present at all times in roll.

Two electronic detents are used. The lower detent activates the CWS mode and is equivalent to the turn knob detent. The higher value detent is used in the autopilot to disconnect path modes and drop the engage switch from command to manual. This action overrides these modes and provides the CWS function.
3. **Turbulence Mode**

The turbulence mode may be engaged at any time except when the nav. mode selector is on ILS or LAND.

If the autopilot engage switch is in COMMAND when the turbulence mode is engaged, the switch will revert from COMMAND to MANUAL.

The system configuration in turbulence is identical with the MANUAL configuration except that the gains are reduced by approximately one half and the heading hold signal is removed.

4. **Command**

   a. **Heading Select Mode**

   The heading select mode allows the pilot to use the autopilot to fly on a desired heading.

   The desired heading is selected by means of the heading knob on the Mode Select Panel. The heading select mode is engaged by placing the Nav. mode select switch in HDG and positioning the autopilot engage switch in COMMAND.

   The command signal is the Heading Selector error (instantaneous heading of the airplane minus the selected heading). The gain is scheduled as a function of true airspeed to maintain consistent system performance throughout the flight regime.

   The attitude command limit for the Heading Select mode is ±30° with the option for change to ±10° for TAS above 500 feet/sec. The roll rate limit is variable from 1.5 deg/sec to ±3 deg/sec as a function of the amount of heading select error.

   b. **Localizer Mode**

   Use of this mode requires the following pilot procedures:

   (1) Tune in the localizer receivers.

   (2) Dial in the runway heading with the course selectors on the mode select panel.

   (3) Position the mode select switch in VOR/LOC.

   (4) Dial in the desired localizer beam intercept heading displayed on the Heading Select window.

   (5) Position the Automatic pilot engage switch in COMMAND.

   The autopilot is in the Heading Select mode until the localizer capture sensor operates and the capture mode is initiated.
After capture, the system switches to the localizer on-course mode when the on-course logic is satisfied.

The basic damping signal in the localizer mode is ground heading, obtained by summing drift angle with course error.

The use of ground heading is contingent upon receipt of a drift angle valid signal (DAV) from the INS. If this signal is lost, the roll computer automatically reverts to the use of derived beam rate, washed out heading, and lagged roll for localizer mode damping. The beam displacement and integral parameters are the same for the drift angle valid and non-valid conditions.

The localizer system has three submodes of operation; namely: capture, on-course, and on-course approach. The circuit implementation of the system is such that it will automatically switch to the proper submode configuration when predetermined requirements are satisfied.

(1) LOC Capture

The localizer capture is initiated when a summed combination of intercept angle and derived beam rate becomes equal to or less than the instantaneous beam error or when beam error is less than one degree. The exact capture and on-course logic and gains are shown in Table 4.

When the drift angle valid signal is present, the displacement command is the localizer beam error and damping is provided by the derived ground heading signal. If the drift angle valid signal is lost, the ground heading signal is removed and derived beam rate and course error are substituted.

(2) LOC On-Course

The LOC on-course submode is initiated when predetermined conditions of bank angle, beam displacement and beam rate are satisfied, as summarized in Table 4.

The localizer on-course configuration is similar to the capture configuration, with the following modifications:

a signal proportional to the integral of beam error is introduced to reduce the steady state error in presence of thrust asymmetry and lateral mistrim conditions,

a thirty-three second time constant high pass filter in the ground heading and course error path washes out steady state errors due to INS and course error signals offsets, as well as heading errors due to crosswinds in the drift angle non-valid condition,
a signal proportional to lagged roll is added for increased damping in the drift angle non-valid condition.

(3) LOC On-Course Approach

The autopilot switches from the on-course submode to on-course approach submode at 1500 feet of altitude.

Mechanization of this submode is similar to that of the on-course submode when the drift angle is valid. When the drift angle is not valid, the washed out course error signal is removed to improve wind shear performance. The beam displacement and integral gains are scheduled linearly with radio altitude to compensate for beam convergence while the damping parameter gains are increased to improve close-in performance.

The submode gains, command limits and engage logic are summarized in Table 4.

C. VOR Mode

The procedure for the pilot to engage this mode is identical to that of the localizer except that the VOR frequency has to be selected rather than the LOC frequency. The VOR mode has three submodes; namely: capture, on-course, and over the station. The basic damping parameter is ground heading. If the drift angle valid signal is lost, the INS drift angle signal is removed, leaving the course heading error as the system damping signal. The displacement command is the beam error for both drift angle valid and non-valid conditions. System gains are scheduled as a function of TAS to maintain good performance throughout the flight regime. The system gains, limits and mode initiation logic are summarized in Table 4.

(1) Capture

When the VOR mode is first selected and the aircraft is outside the capture threshold, the autopilot is in the heading select mode which steers the airplane to the desired intercept angle established with the heading select control on the Mode Select Panel. The capture sensor is armed.

The variable engage point logic used for VOR capture is shown in Table 4. For intercept angles between 90 degrees and 10 degrees, the capture starts at beam error between 1.8 and 0.2 dots. The greater the intercept angle, the earlier the capture maneuver is initiated.

A 34 degrees course cut limit is provided.
(2) On-Course

The on-course submode is initiated when the bank angle is less than 3 degrees and the course error is less than 15 degrees.

The beam displacement gain is reduced to one-half the value used during capture. A beam integral signal is introduced to reduce the beam error in presence of thrust asymmetry and lateral control surface mistrims. A 200-second washout of the ground heading signal is also introduced to eliminate INS drift angle and course error static offsets. In the drift angle non-valid condition, this washout improves the system performance under cross wind conditions.

Maximum position and rate commands are limited to +10 degrees and +1.5 degrees/second respectively during the VOR on-course submode.

(3) Over the Station

The VOR over-the-station sensor initiates this submode upon detection of beam rates higher than 0.5 degrees/second. The system automatically reverts to the on-course mode, after passing the station, when the beam rate signal has decreased below 0.5 degrees/second for 20 seconds.

During the time that the airplane is over the station, the VOR beam signal is removed and the autopilot command signal is either ground heading for the drift angle valid condition, or course error for drift angle non-valid condition. If the pilot desires to make a course change while over the station, he may dial in the change in course setting and the system will track outbound on the new radial.

d. INS Mode

The autopilot may be used to capture and track any of the great circle routes that have been programmed into the INS computer.

The INS mode is armed by placing the Nav. mode selector switch of the autopilot mode select panel in the INS position. Figure 7 shows a typical control sequence. If the aircraft is further than 7.5 nautical miles off the desired great circle course, the autopilot is in the heading select mode, and steers the airplane to the desired intercept angle established with the heading select knob. In order for the Heading Select mode to operate on those airplanes with remote set preselect heading where the heading error signal is developed in the HSI, the INS-RADIO switch must be left in RADIO position until the INS capture point is reached. When the 7.5 mile point is reached, the INS mode is initiated.

Cross-track deviation and track angle error outputs of the INS are used to compute the desired steering command.
Control is similar to VOR control with the following exceptions:

a. Since the INS provides the equivalent of a non-convergent beam, the distance from destination does not effect system performance.

b. The damping signal, track angle error, is not susceptible to cross-wind effects which would result in lateral displacement from the desired course.

(1) INS Capture

The capture maneuver is automatically initiated at 7.5 nautical miles from the desired INS course. The bank angle command is limited to 30 degrees and bank rate is limited to 1.5 degrees per second during capture. The system gains are summarized in Table 4. A 25-degree course cut limiter is employed.

(2) INS On-Course

The capture is complete and on-course control is initiated when the cross-track deviation is reduced below 1,070 feet and track angle is below 3 degrees. The same gains are employed in the autopilot as during capture. The on-course sensor downshifts the bank and bank rate limiters to 10 degrees and 1.5 degrees per second respectively.

(3) Waypoint switching

Automatic switching from one great circle course to another is provided. On the INS system Control and Display Unit, the "Auto-Manual" switch must be set to the "Auto" position to use this feature. If this is not done, the autopilot will overfly the waypoint and continue on the extension of the great circle. If the "Auto" sequence is used, switching of the autopilot to the next route occurs a short distance before the waypoint is reached. (See Table 4)

Automatic switching from one great circle course to another is provided. On the INS system Control and Display Unit, the "Auto-Manual" switch must be set to the "Auto" position to use this feature. If this is not done, the autopilot will overfly the waypoint and continue on the extension of the great circle. If the "Auto" sequence is used, switching of the autopilot to the next route occurs a short distance before the waypoint is reached. (See Table 4)

It will occur at 3.5 nautical miles for cases where the angular change between successive courses is small. Restoration of the capture bank limits is provided automatically. When the on-course conditions are again satisfied, the reduced bank and bank rate limits are automatically reinstated.

e. ILS Mode

The ILS mode is identical to the Localizer Mode for the roll autopilot. Glideslope control is armed by this mode and the autopilot continues to fly toward the glideslope beam on either pitch attitude, vertical speed, altitude or IAS hold (optional mode) until a predetermined glideslope signal level is reached. (See pitch axis system description)
f. Land Mode

The autopilot LAND mode provides the Boeing 747 with a fail-passive dual-channel automatic approach and landing system. The LAND mode features dual ILS and flare coupling.

The LAND mode is properly selected when the following prerequisites are satisfied:

1) Nav. Mode Selector switch in LAND
2) Both course controls set to the runway heading
3) Heading select control set for the desired intercept angle with the localizer
4) VHF/NAV receivers set to the proper localizer frequency
5) Both autopilot engage switches in the command position.

Upon selection of this mode, single-channel operation, identical to that for ILS mode, is initiated. Dual-channel operation does not begin until after the autopilot is on Loc. approach, glide slope capture is completed, and the airplane is less than 1,500 feet altitude above the surface. The second channel synchronizes to the controlling channel until it is engaged; at this point, the flare computer is armed and equalization and monitoring of both channels begins.

(1) Equalization

Equalization is accomplished by taking the difference of the central control actuator and the autopilot actuator LVDT signals and feeding this signal back to the autopilot path integrator.

The output position of the central control actuator will be equal to the output position of the autopilot actuator having the least value (see Section C). Thus, the channel having the greater command will be different from the central control actuator output by the difference between the A and B autopilot commands.

The channel with the lower command signal receives no equalization signal since the difference between commanded and output position is zero, while the channel having the higher signal receives an equalization that tends to reduce its output signal to match that of the controlling channel. It should be noted that each channel has an independent equalization system and that there are no cross ties between channels.

The equalization signal in the lateral axis is limited, so that ramp faults of relatively low values may be detected. The equalization signal is also gain scheduled as a function of radio altitude from 1 to 0.5. Thus, equalization is decreased as the airplane approaches touchdown.
(2) Monitoring

The monitoring system on the 747 dual-channel autopilot acts on the measured difference between the output position of each autopilot actuator and the output position of the central control actuator associated with that autopilot actuator. A block diagram of the monitoring system is shown in Figure 13. The two actuator position sensors used in the equalization circuit also feed the monitor. Each channel has a pair of monitors: one for pitch and one for roll. A failure is indicated by the monitor if an autopilot channel disagrees with the central control actuator output by a given amount for a set length of time. These values are presently set at six degrees and two seconds in the roll channel. A steady red cockpit warning light is initiated any time a monitor is tripped. Thus, early warning of any potential problems is brought quickly to the pilot's attention without disconnecting the autopilots. Should the warning light be activated by noise or radio beam masking, it is not latched to the ON condition; and thus, goes out at the conclusion of these temporary disturbances.
### Table 5

**Summary of Flight Director Roll Axis**

<table>
<thead>
<tr>
<th>MODE</th>
<th>DESCRIPTION</th>
<th>SYMBOL</th>
<th>VALUE</th>
<th>UNIT</th>
<th>ROLL ANGLE</th>
<th>GAIN</th>
<th>RATE</th>
<th>LIMIT</th>
<th>HEADING</th>
<th>SELECT</th>
<th>COMMAND</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>CMDBAR/φ</td>
<td>0.032</td>
<td>IN/DEG</td>
<td>0.6 DEG/DEG/SEC</td>
<td>SAME AS AUTOPILOT</td>
<td>RATE LIMIT</td>
<td>9°/SEC</td>
<td>INS CAP</td>
<td>4°/SEC</td>
<td>INS O/C</td>
</tr>
<tr>
<td></td>
<td></td>
<td>CMDBAR/φ</td>
<td>0.032</td>
<td>IN/DEG</td>
<td>0.6 DEG/DEG/SEC</td>
<td>SAME AS AUTOPILOT</td>
<td>RATE LIMIT</td>
<td>9°/SEC</td>
<td>LOC CAP</td>
<td>4°/SEC</td>
<td>LOC O/C</td>
</tr>
<tr>
<td></td>
<td></td>
<td>CMDBAR/φ</td>
<td>0.032</td>
<td>IN/DEG</td>
<td>0.6 DEG/DEG/SEC</td>
<td>SAME AS AUTOPILOT</td>
<td>RATE LIMIT</td>
<td>9°/SEC</td>
<td>VOR CAP</td>
<td>4°/SEC</td>
<td>VOR O/C</td>
</tr>
</tbody>
</table>

The submodes engaged logics are identical to that of autopilot.

Backup beam is available in flight director only (option).

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FIG. 8  TRUE AIRSPEED GAIN PROGRAM
Fig. 10

Time vs. Gain Factor

0 50 100 150 200
Time (Seconds)

0.5 1.0
Gain Factor

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FIG. 11a

747 ROLL CONTROL WHEEL STEERING BLOCK DIAGRAM

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F. PITCH AUTOPilot

A summary of Autopilot/Flight Director pitch axis characteristics is contained in Tables 6 and 7.

The 747 autopilot operates through a parallel servo system. Thus all actuator response due to a command from the autopilot is also fed back to the control column. The feedback path is the primary elevator control system. At the frequencies which the autopilot system normally operates, the control column can be assumed to be directly proportional to the elevator surface motion. Figure 23a shows the linearized transfer functions for the Autopilot servo and elevator power control unit. The normal gain between $\delta$ column and $\delta$ elev. is $0.78$ degrees of column per degree of elevator displacement about surface neutral. The full column to elevator curve is shown in Figure 23b.

Figure 1 is the block diagram of the basic pitch autopilot control system. Pitch attitude and rate loops are basic for pitch modes of operation. These loops provide both short and long period stabilization and damping. The rate signals are sensed by gyro's installed in each autopilot computer. Attitude signals are from Inertial Navigation System (INS). Fixed attitude and rate gains of $\frac{\delta e}{\delta} = 3.5$ Deg and $\frac{\delta e}{\delta} = 2.2$ Deg/Sec are employed for flaps up flight. Rate damping is increased from 2.2 to 3.5 for flaps down flight. The pitch rate is passed through a $1/(0.1s+1)$ lag for higher frequency suppression, a $1/(0.05s+1)$ lag for structural mode decoupling and a washout $1/(ts+1)$ where $t = 2$ for flaps up and to provide roll compensation. The compensation signal ($1-\cos \phi$) is gain scheduled as a function of airspeed.

When the autopilot is on, the automatic trim system maintains pitch trim of the airplane. This is true for all autopilot modes except turbulence. When this mode is engaged, pitch trim is not active.

When engaged in MANUAL, the pitch autopilot responds to commands inserted via the pitch knob. When engaged in COMMAND, the pilot has the option of control by any of the following modes: ALT. HOLD, ALT. SELECT, IAS HOLD, V/S (Vertical Speed Control), MACH HOLD, ILS, or LAND.

1. Engage Synchronization

Before the autopilot is engaged, the output of the servo amplifier is fed back to the path integrator at a high gain. This loop maintains the output of the servo amplifier at zero in order to obtain transient-free elevator when the A/P is engaged. At engagement, the synchronizing path is opened. The integrator hold circuit tracks airplane attitude which serves as the reference for attitude hold, the initial mode of the A/P.


The MANUAL mode is engaged by placing the autopilot engage switch in the MANUAL position. Synchronization is provided so there is no attitude transient when the mode is engaged.

a. PITCH WHEEL: If the pilot desires to change the airplane pitch attitude, the pitch wheel on the flight controller is used. The pitch wheel produces an attitude command proportional to wheel displacement.
b. CONTROL WHEEL STEERING (CWS): (CWS is alternative to Pitch Wheel.) CWS enables the pilot, by applying a force to the column to insert a command signal into the A/P to change the airplane attitude. The force signal is processed as in Figure 11a. Force on the control column commands pitch rate via an integral path, the gain of which is programmed with True Airspeed to give uniform performance over the flight regime. When the force is removed, the path integrator is synchronized to the airplane attitude at the time of force release.

When the A/P is in MANUAL or in COMMAND and not in any path mode, the A/P will be in the CWS mode. If in COMMAND and in a path mode and the high detent force (19 pounds) is exceeded, the path mode drops off and the A/P drops to MANUAL except when in Altitude Hold. When in CWS and the deadzone (low detent) is exceeded, the automatic trim is inhibited. Whenever an attitude of \( \pm 25 \) degrees is reached and the pilot applies a force in the direction to increase this attitude, the rate path is inhibited (MANEUVER LIMIT CONTROL).

3. Turbulence

In a turbulent environment TURB. may be engaged on the Turb/Speed mode select switch when the NAV mode switch is not in ILS or LAND. With TURB engaged, the autopilot is automatically switched from COMMAND to MANUAL and the flight director pitch trim control becomes effective. The pitch attitude and pitch rate gains are reduced by one-half. The automatic stabilizer trim is off in the turbulence mode.
<table>
<thead>
<tr>
<th>PARAMETER</th>
<th>SYMBOL</th>
<th>SUBMODE</th>
<th>VALUE</th>
<th>UNITS</th>
<th>SHAPING</th>
<th>PROGRAM</th>
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<tbody>
<tr>
<td>BAR SENSITIVITY</td>
<td></td>
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<td>$\omega_{11-CD}$</td>
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<td>18.8</td>
<td>DEG</td>
<td>$\frac{1}{2}$</td>
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<td>DEG/FT</td>
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<td>CAS</td>
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<td>DEG/DEG</td>
<td>$258 + 1$</td>
<td>CAS</td>
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<td>DEG/KT</td>
<td>$206 + 1$</td>
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<td>DEG/MACH</td>
<td>206 + 1</td>
<td>CAS</td>
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<td>DEG/DEG</td>
<td>206 + 1</td>
<td>CAS</td>
</tr>
<tr>
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<td>DEG/FT</td>
<td>206 + 1</td>
<td>CAS</td>
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<td>DEG/DEG</td>
<td>206 + 1</td>
<td>CAS</td>
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<td>GLIDE SLOPE</td>
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<td>CAPTURE</td>
<td>2.5</td>
<td>DEG/FT/SEC</td>
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<td>CAS</td>
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<td>CAS</td>
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<td>DEG/DEG</td>
<td>206 + 1</td>
<td>CAS</td>
</tr>
</tbody>
</table>

1) AUTOMATIC STABILIZER TRIM IS INHIBITED DURING THESE MODES.
2) SEE DIAGRAM 1 FOR CAS GAIN PROGRAM.
3) THE GAIN PATH INCLUDES THE PATH FILTER AS WELL AS AN 8 DEG & AMPLITUDE LIMIT.
4) WITH THE RADIO ALTIMETER, VALID SIGNAL PRESENT GAINS ARE PROGRAMMED AS PER DIAGRAM 2 WITH RADIO ALTITUDE, OTHERWISE, THEY ARE PROGRAMMED WITH THE TIME BASE PROGRAM SHOWN IN DIAGRAM 4.
5) SEE DIAGRAM 3 FOR TAS GAIN PROGRAM.
6) IN FOR GLIDE SLOPE CAPTURE, TRACK; FLARE; GO AROUND ONLY.
### AUTOPILOT PITCH AXIS

<table>
<thead>
<tr>
<th>Pitch Mode</th>
<th>Parameter</th>
<th>Submode</th>
<th>Value</th>
<th>Units</th>
<th>Shaping</th>
<th>Program</th>
<th>Remarks</th>
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<tbody>
<tr>
<td><strong>ATTITUDE HOLD</strong></td>
<td>Attitude</td>
<td>Other</td>
<td>3.2</td>
<td>deg/deg</td>
<td></td>
<td></td>
<td><strong>ROLL LOSS AT MID-BAND NOT INCLUDED.</strong></td>
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<td></td>
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<td>Flaps Up</td>
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<td>deg</td>
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<td>Beta-Cosm</td>
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<td>Pthn Knob</td>
<td>Stroke</td>
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<td>deg/stroke</td>
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<td>TURBULENCE</td>
<td>All Above Gains</td>
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<td>CONTROL WHEEL STEERING</td>
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<td>ALTITUDE SELECT</td>
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<td>deg</td>
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<td></td>
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<td>Beta</td>
<td>-</td>
<td>deg</td>
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<td>-</td>
<td>integral</td>
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<td>GLIDE SLOPE</td>
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<td>Flare</td>
<td>Beta</td>
<td>-1.9</td>
<td>deg/sec</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>Flare Ratio</td>
<td>Beta</td>
<td>.106</td>
<td>deg</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>FLARE SYNCHRONIZATION</td>
<td>Flare</td>
<td>Beta</td>
<td>20</td>
<td>deg/sec</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>FLARE SYNCHRONIZATION</td>
<td>Flare</td>
<td>Beta</td>
<td>7.24</td>
<td>deg/sec</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>ATTITUDE MEMORY SYNCHRONIZATION</td>
<td>Flare</td>
<td>Beta</td>
<td>30</td>
<td>deg/sec</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

1. **AUTOMATIC STABILIZER TRIM IS INHIBITED DURING THESE MODES.**
2. **SEE DIAGRAM 7 FOR CAS GAIN PROGRAM.**
3. **THE GAIN PATH INCLUDES THE PATH FILTER AS WELL AS AN 8 DEG/SEC AMPLITUDE LIMIT.**
4. **WITH THE RADIO ALTIMETER, VALID SIGNAL PRESENT GAINS ARE PROGRAMMED AS PER FIGURE 18 WITH RADIO ALTITUDE, OTHERWISE, THEY ARE PROGRAMMED WITH THE TIME BASE PROGRAM.**
5. **SEE FIGURE 16 FOR TAS GAIN PROGRAM.**
6. **LIMTED AT 3.5 DEGREES ELEVATOR DIFFERENCE, WARNING LIGHT DELAY DELAY 1 SECOND.**

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**TABLE 7**

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**REV d**

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**D6 - 30643 - 10/3/45**
4. Command

When engaged in COMMAND and no other pitch mode is selected, the A/P is in pitch attitude hold and the pitch knob is operative.

a. Altitude Hold

The altitude Hold mode holds the airplane at the altitude existing when the mode is engaged. This mode can be engaged in either MANUAL or COMMAND. If the mode is engaged with the airplane climbing or descending at a reasonable rate, the airplane returns to and holds the engage altitude.

The Air Data Computer provides the reference signal for this mode. This altitude error signal is provided after clutching a mechanically nulled synchro to the altitude shaft at mode engage. Aircraft altitude rate (or vertical speed) is also sensed and provided by the Air Data Computer.

The block diagram and gains for the Altitude Hold mode are included as part of Figure 15.

The altitude error signal commands attitude changes required to maneuver the airplane toward zero altitude error. Altitude rate is added to improve damping. Integral control on the altitude error removes standoffs. The gains used are programmed down at high speeds to about 1/3 of the low speeds. (See Figure 16.)

b. Altitude Select

The Altitude Select mode allows the pilot to select a desired flight altitude. If the selected altitude is more than 1200 feet from actual altitude, the pilot also selects the desired mode of climb or descent. With these selections made and the mode engaged, the autopilot maneuvers the airplane to smoothly capture and hold the selected altitude. This mode is particularly useful when a number of successive altitude changes are required.

Altitude Select has three submodes: arm, capture, and track. (See Figures 15 and 17.)

(1) Arm

In the arm submode, the pilot selects some other pitch mode of the autopilot, such as vertical speed, IAS hold, or pitch knob, to command the aircraft to climb or descend toward the desired altitude. In this submode, the altitude select switch is engaged and the capture logic is armed.
**ALTITUDE SELECT MODE**

**SEQUENCE:**

1. **DESIRED ALTITUDE SELECTED**
   - ALT SELECT ARMED
   - PITCH MODE CONTROL IN DIRECTION OF ALTITUDE
     - PITCH KNOB
     - IAS HOLD
     - VERTICAL SPEED
     - MACH HOLD
   - ALT SEL ANNUNCIATOR — AMBER

2. **SELECTED ALTITUDE CAPTURED**
   - ALT SELECT MODE IN CONTROL
   - PITCH MODE SWITCH MOVES TO OFF
   - ALT SEL ANNUNCIATOR — GREEN

3. **CHANGE IN SELECTED ALTITUDE**
   - $\Delta h < K_h$
   - ALT SEL ANNUNCIATOR — GREEN

4. **CHANGE-PITCH MODE SWITCH IN CONTROL**
   - $K_h < \Delta h < K_h$
   - ALT SEL ANNUNCIATOR — AMBER

**Fig. 17 Altitude Select Mode**
(2) Capture

Capture is initiated when the aircraft's altitude approaches the selected altitude to within 1200 feet, and the altitude error (in feet) is less than 15 times the altitude rate (in feet/sec) as sensed by the coincidence detector. The detector uses course altitude and rate data from the CADC to determine the switching point. When the detector is satisfied, it switches off the previously used climb or descent mode and initiates capture. During capture, the fine altitude data from the CADC and altitude rate information are summed as indicated by the following control equation to command the aircraft to maneuver and capture the selected altitude:

\[ \Delta \theta_c = G_m \left( 0.006 \Delta h + 0.9 k \right) \]

where:
- \( G_m \) is the gain program with computed airspeed (unity at low speed)
- \( \Delta \theta_c \) is pitch attitude command in degrees
- \( \Delta h \) is altitude error in feet
- \( k \) is rate of climb in feet per second

This produces a flare toward the selected attitude which is rate limited to reduce "g" forces if initiated from steep climbs or descents.

(3) Track

The track phase is initiated when the airplane approaches within 100 feet of the selected attitude, with a rate not exceeding 5 feet/seconds. The control equation is as follows:

\[ \Delta \theta_c = G_m \left( 0.029 \Delta h + 0.05 k \right) \]

with the units defined as above.

c. Vertical Speed

The vertical speed mode is selected by the Turb/Speed switch on the Mode Select Panel. Prior to selection, the vertical speed wheel on the Mode Select Panel is synchronized to aircraft.
vertical speed. It is driven by an electro-mechanical servo follow-up to accomplish this repeating function. Upon engagement of this mode, the motor of this follow-up de-activates to provide the reference required to generate a vertical speed error signal. Should the pilot desire to operate at a different vertical speed, the command wheel is moved to establish a new reference. The vertical speed wheel has 1000 ft/minute graduations between the limits of operation which are +1000 ft/minute to -8000 ft/minute.

The block diagram and gains used for the vertical speed mode are shown in Figure 15. An acceleration limiter provides smooth system response to command changes. Gain scheduling is included to allow good performance over the speed range of the 747.

d. IAS Hold

This mode gives the pilot automatic Indicated Airspeed hold capability.

A clutched synchro on the computed airspeed shaft of the CADC provides an IAS error signal which is the difference between actual IAS and the IAS at the time of mode engage. The error signal commands pitch attitude required to hold the reference airspeed. The block diagram and gains used are shown in Figure 15.

3. Mach Hold

The Mach Hold mode provides the capability to automatically hold the airplane Mach existing when the mode is engaged. The mode is selected by the Turb/Speed select switch on the Mode Select Panel. The Mach error signal is generated in the CADC by a clutched synchro. The error signal commands the airplane pitch attitude required to zero the Mach error signal. The block diagram and gains are shown in Figure 15.

f. ILS Mode

The pitch axis control system for dual channel automatic landing is shown in Figure 19. The ILS mode uses part of one channel, that is, glide slope capture and control. Upon selection of ILS, glide slope capture is armed and the glide slope indicator light on the mode annunciator panel is amber. During the glide slope arm phase, a glide slope intercept path from above or below the glide path can be flown on ALT, HOLD, ALT SELECT, MANUAL, MACH, IAS, or V/S. When the glide slope receiver is within ±30 mv (±14") of beam center, the vertical beam sensor is tripped and glide slope capture begins. A sink rate error signal, barometric altitude rate plus a 12.5 fps bias, is switched into the path integrator for 10 seconds to produce a pitch angle approximately equal to that needed to fly the glide slope.
Normal acceleration, through a washout to eliminate steady state accelerometer outputs and filtered through a 10 second lag to pseudo-integrate, is switched in for beam damping. During the 10 second pitch down, beam error is synchronized by a limited integrator at the integrator's high rate. At the end of the 10 seconds, the sink rate error signal is switched off and glide slope control begins.

Beam error minus the synchronized signal stored on the rate-limited integrator is switched in, with this stored value bleeding down at the low rate of the integrator to give an easing on of the signal. On glide slope, beam error is programmed with radio altitude to provide a nearly constant elevator deflection per foot of beam error. The gain programmer is shown in Figure 18. In the event of a radio altimeter failure, a time-based gain program is also provided. (Figure 18-2).

g. Land Mode

The LAND Mode is shown in Figure 19. It is selected on the NAV mode selector when dual-channel, fail passive localizer, glide slope, and flare is planned. Sequencing during the LAND mode is shown in Figure 20.

Upon selecting of the LAND mode and with one channel in COMMAND, the autopilot is in single channel operation. The mode is the same as ILS except that the autopilot warning light flashes amber and the autopilot will automatically disengage at 150 ft. if the second channel is not engaged.

The second channel is locked in OFF until the automatic confidence test of the dual-channel camout monitor is completed. This takes about 1 second after LAND is selected and the first channel is engaged in COMMAND. Upon moving the second channel engage switch to COMMAND, the flashing amber warning light is extinguished and the second channel is armed. The system will operate in this configuration until the dual channel engage interlock logic is satisfied.

The autopilot remains in single channel operation until all of the following conditions are met:

LO on-course
Rad. Altitude less than 1,500 ft.
Glide Slope control
Rad. Altitude valid

When these conditions are met, dual-channel operation begins and is annunciated after a three second delay by an amber A/P FLARE ARM light on the Mode Annunciator Panel. The non-latching autopilot camout monitor gives the pilot a steady red warning light on the Mode Annunciator Panel if a
Fig. 21 Land Mode Interlocks
camout exists (Page 61) in either channel for two seconds. Interlocks for the LAND mode are shown in flow diagram, Figure 21.

(2) Synchronization

When in LAND with both channel engage switches in COMMAND, but still flying single channel, the second channel actuator is in the caged position and its servo amplifier output is being nulled by the path integrator by means of the synchronization loop. At A/P FLARE ARM, the hydraulic pressure builds up first in the force detent mechanism moving the autopilot actuator to a position matching that of the elevator. The LVDT signal generated when the autopilot actuator is moved into position is fed back to the servo amplifier, and is nulled out by the synchronization loop. The actuator is then pressurized and the synchronization loop is opened. Since the servo amp was held to zero, the actuator does not move until it is commanded from the autopilot.

(3) Glide Slope

The glide slope control law for each channel is the same as that of ILS. Figure 19 shows all switches in position for glide slope control.

During dual-channel operation, equalization is in effect, without signal intertie between channels. An equalization command proportional to the difference, if any, between the autopilot actuator and the elevator is fed back to the path integrator. The equalizer signal to the path integrator is limited. The equalization gain is programmed down as a function of altitude since the requirement for equalization to reduce null offsets is likewise reduced.

(4) Flare

At the flare point, the FLARE light on the mode annunciator changes from amber to green. The camout monitor remains active.

Since the gain programmer is zero, beam error and the gain programmed portion of equalization are not applied during flare. However, a small fixed gain equalization signal remains on throughout flare.

The flare law:

\[ h_e = h_{DER} + \frac{h}{6} + 2; \quad h_{DER} = h \left( \frac{S}{S+1} \right) + a \left( \frac{1}{S+1} \right) \]

commands an airplane descent rate, linearly decreasing with altitude, from the descent rate at flare to 2 feet/second.
The descent rate is derived from the radio altitude and vertical accelerometer signals. The descent rate error is summed with filtered acceleration for damping and then passed through direct and integral paths to produce an attitude command. A synchronizer loop maintains the command to zero until flare.

\[ \Theta_{CF} = k_{h} e + k_{a} a + \frac{1}{3} (k_{a} e + k_{a} a) + \Theta_{bias} \]

The flare command is limited to 7.5° nose up and 7.5° nose down pitch attitude.

(5) Fail-Passive Actuators

The autopilot actuators are located on the two inboard elevator control packages. Figures 22 and 23 show the autopilot elevator control configuration.

The feel computer provides a centering force for all autopilot and manual control commands. Feel force is programmed as a function of dynamic pressure and stabilizer position and provides authority limitation to the autopilot. The autopilot actuators are tied to the main valve inputs through force limited detents. In single-channel operation, only one detent is engaged and both main valves follow the engaged autopilot actuator.

Figure 5 shows a schematic of the detents and servo system mechanization for the roll axis. Pitch mechanization is similar. In dual-channel operation, both autopilot actuators are powered and the detents are engaged. If the autopilot actuator positions disagree, the elevators will follow the command nearer zero. The detent monitors associated with each autopilot channel measure the disagreement of the autopilot actuators with the output. A disagreement of 6.0° elevator will give a camout and a consequent pilot warning light after a one-second delay. Dual-channel elevator authority is double the single-channel authority or approximately +10, -30 degrees at final approach speed. The detent authority remains constant at 27 pounds stick force, maximum. At maximum q conditions, elevator single-channel authority diminishes to +1.5 degrees elevator.
G. AUTOPILOT AUTOMATIC STABILIZER TRIM

The autopilot is provided with two separate stabilizer trim systems. During single-channel cruise operation of the autopilot, the A trim system is utilized with the "A autopilot" and the B trim system is utilized with the "B autopilot". During dual-channel operation of the autopilot, both trim systems are capable of operation. One of the two is utilized and the other is armed and in a standby state. Suitable monitoring and logic is included to effect an automatic transfer to the standby trim, should a malfunction occur in the active channel of trim.

The auto-trim unit drives the stabilizer to reduce steady state elevator displacement from neutral when the autopilot is engaged. This reduces to a low level the transient which occurs when autopilot is disengaged. Auto-trim is obtained during all autopilot modes except Turbulence.

The stabilizer has no direct connection with the primary control system. Thus, motions of the stabilizer show up on the column only as the amount of elevator surface which must be held on the column to maintain airplane trim. When the autopilot is engaged the autopilot commands the amount of elevator needed to maintain the desired flight path and airplane trim. When the autopilot is commanding an elevator position greater than the trim threshold (Figure 24) the trim system will drive the stabilizer until the elevator required to maintain the desired flight path is reduced below 0.185 degrees of elevator. The effect of automatic trim as seen on the column is a smooth return of the column to near neutral as the trim system operates. The rate of stabilizer trim is inversely proportional to the impact pressure ($\varphi$). Thus, the column rate of return is similarly reduced.

Autopilot Stabilizer Trim Unit (ASTU) channel is shown in Figure 24. The arm and control circuits are identical. Trim is effected by the presence of discretes from both arm and control when there is hydraulic pressure to operate the brake pressure switch. When the elevator exceeds the trim threshold for five seconds, the stabilizer is driven until the elevator decreases to 0.185 degrees. The trim rate and thresholds are variable with feel pressure as indicated in Figures 24b and 24c.

The trim warning monitor is activated 0.5 seconds after an active or passive failure of the arm or control circuits or brake pressure switch. A warning is also obtained for an out-of-trim condition sustained for 12 seconds.

Dual-Channel Autopilot Mode (Autoland)

One auto-trim channel is engaged in this condition; and in the event of failure, this channel is automatically disengaged and the other standby channel is engaged by the changeover switches in the Boeing accessory box. Thus, "fail operational" trim is provided during automatic landing. Should a camout occur, the ASTU is inhibited from moving the stabilizer.
"A" TRIM SYSTEM

FEEL PRESSURE LVDT NO.1

ELEV. LVDT NO.1

BIAS

FEEL PRESSURE LVDT NO.2

ELEV. LVDT NO.2

BIAS

FEEL PRESSOR

ELEV. PROCESSOR

BIAS

DETECTOR

BIAS

PHASE SENSITIVE CONTROL (5 SEC DELAY)

LIMITED AUTHORITY GAIN CHANGE PATH

STAB WARNING LIGHT

TO "B" TRIM MONITOR FROM

STAB TRIM MONITOR

UP

DOWN

CONTROL CIRCUIT

"B" TRIM SYSTEM

IDENTICAL TO "A" TRIM SYSTEM

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AUTOMATIC STABILIZER TRIM

THE BOEING COMPANY
II. MACH TRIM SYSTEM

(System not installed on 747 airplane)

Pages 63 - 68 deleted.
III. YAW DAMPER SYSTEM

A. GENERAL

The 747 directional control system comprises dual rudders, each independently powered by a dual tandem hydraulic actuator. Dual redundant yaw damping is provided on the 747 by incorporating an independent electro-hydraulic augmentation system with each rudder segment.

Yaw damping signals are connected to the rudder actuators in a series fashion such that the pilot's rudder pedals are not displaced by yaw damper commands. This feature permits the yaw dampers to be operative at all times through take-off, cruise, and landing without interfering with normal pilot rudder control.

A preflight cockpit operated confidence test is provided to check each yaw damper system prior to departure from the ramp area. Disengage switches mounted on the pilot's overhead panel are provided to enable the flight crew to shut off either yaw damper system should a malfunction occur. Figure 27 is a pictorial diagram of the lower rudder yaw damper system.

In addition to providing additional damping of basic airframe lateral directional oscillations (dutch roll), the 747 yaw damper system has an added feature designed to improve airplane response to turning maneuvers in flap down flight conditions. This system is called the "turn coordinator" and deflects rudders proportional to roll rate in a "turn coordinating" sense, thereby improving roll control response.

Due to a favorable phase relationship, the turn coordinator feature has the added benefit of further improving basic dutch roll damping beyond that available from the yaw damping mode alone.

The "turn coordinator" system is used at flaps down condition only. An "easy on/off" circuit is implemented in the "turn coordinator" command path to eliminate transient rudder kicks resulting from the flap switching should the aircraft not be at zero roll attitude.

The design objective is to provide additional dutch roll damping and turn coordination with a system which provides no potentially hazardous failure conditions. (See document D6-13647, IEFCS Failure Analysis).
B. SYSTEM DESCRIPTION

Figure 28 is the block diagram of the 747 dutch roll damper and turn coordination system. Its mathematical model is shown in Figure 29.

1. Dutch Roll Damping Signals

Dutch roll damping is provided by a yaw rate (\(\dot{\psi}\)) signal. At flaps down condition, additional bank angle signal is provided to increase system damping. Yaw rate is sensed by a rate gyro, mounted in the yaw damper chassis. Bank angle is obtained from the Inertial Navigation System (INS). These signals are independently demodulated.

2. Band Pass Filter

At flaps up condition, the yaw rate signal passes through a band pass filter into the servo amplifier. The band pass filter is composed of R-C components and operational amplifiers. The transfer function of the filter can be expressed in Laplace form as the following:

\[
\frac{2.72S}{(2.72S+1) (.272S+1)}
\]

At flaps down condition, roll attitude signal passes through a similar band pass filter and is summed with the filtered yaw rate signal into the servo amplifier. This additional filtered roll attitude signal provides additional dutch roll damping.

The functions of the band pass filter are:

1. To washout the steady yaw rate and roll attitude signals and to eliminate null offset of sensors.

2. To provide dutch roll damping signals with minimum phase shift at dutch roll frequencies, in order to achieve optimum damping.

3. To reduce high frequency signal amplitudes so as to minimize possible coupling with structural modes.

The Bode and phase angle plots of the band pass filter are shown in Figures 31 and 32.

3. Dutch Roll Damper Gain

At flaps down condition, the yaw rate and the roll attitude gains are

\[
2.5 \frac{\dot{\alpha}}{\dot{\psi}} \text{ (degree) and } \frac{\dot{\alpha}}{\dot{\theta}} \text{ (degree)}
\]

respectively.
At flaps up condition, the yaw rate gain is 1.25 R (degree)
\[ \frac{\delta R}{\psi} \ \text{(degree/\text{sec})} \] For detail information, refer to Figure 33 root loci plot.

4. **Turn Coordination**

A shaping circuit is used to derive a roll rate signal from the roll angle input. The rate circuit used has a transfer function of 
\[ \frac{1}{(s + 1)^2} \] which yields a derived roll rate signal from roll angle at turn entry frequencies, but cuts off at frequencies above 1.4 cps in order to minimize the effects of system noise.

The turn coordinator system operates at flaps down flight conditions only. Switching is accomplished by a flaps switch.

The system gain is 0.693 R (degree)
\[ \frac{\delta R}{\psi} \ \text{(degree/\text{sec})} \]

5. **Easy On-Off Circuit**

An easy on-off circuit is implemented in the filtered roll attitude signals path to eliminate transient rudder kicks when the roll attitude signal is turned off or on by the flap actuated switches.

6. **Yaw Damper Electro-Hydraulic Servo**

The servo valve amplifier accepts dutch roll damper, turn coordinator, and servo feedback inputs and provides an output to the electro-hydraulic transfer valve. The transfer valve controls motion of the yaw damper actuator which is linked to the main power control unit valve via a summing link. The electro-hydraulic transfer valve is supplied system pressure via a solenoid operated shut-off valve when the yaw damper is engaged. The yaw damper servo actuator is self-centered by caging springs when the system is de-energized. This preserves the integrity of manual commands when the yaw damper is not energized.

The maximum rudder rate which the yaw damper can command is controlled by the area of the servo actuator orifice and by the area of the piston and is ± 15 deg/sec at no load. The maximum rudder displacement is controlled by the summing lever stops and is ± 3.6 degrees.
There are two feedback paths in the yaw damper. The first path directly feeds back yaw damper actuator position. This feedback is required so that yaw damper servo output will follow command inputs. The second path feeds back yaw damper actuator position via \( \frac{855 + 1}{8} \) in parallel with the normal position feedback. When the yaw damper system is disengaged, the servo amplifier output is fed back through this lag to provide synchronization.

### 7. Self-Test and Confidence Test

The yaw damper computer provides self-test circuitry for the two categories of fault isolation testing: Line Replaceable Unit test (electronic components) and system test (electronic components plus the actuator loop). The self-tests are performed by positioning the test switch and momentarily depressing the press-to-test switch on the yaw damper front panel. The monitor light on the front panel will indicate the test results (go or no go).

Confidence Test is a pilot actuated and monitored system test. Channel confidence tests can be initiated singularly or simultaneously by use of the cockpit mounted test switches. By observing the upper and lower rudder surfaces position indicator located on the pilot's instrument panel, the operating status of the systems can be assessed.

The confidence test and self-test are interlocked to prevent test initiation in flight.

Separate confidence tests are provided for the dutch roll damping and the turn coordinator functions.

### 8. Performance

Analog computer simulation and digital computer root locus analysis were conducted to investigate yaw damper system performance. Results indicate that the system performs well at all flight conditions. The lowest augmented airplane dutch roll damping ratio is 0.30 at the landing approach 33 degrees flaps down condition. System performance for various flight conditions are summarized in root loci plots in Figure 33.

### 9. Tolerance

Phase shifts and gain variations affect system performance. Root locus analysis were conducted to investigate the effect of tolerances on performance. The Bode plot and phase angle plot for the nominal and the maximum time constants of the rudder to sensors input transfer function are shown in Figures 31 and 32 respectively. The two flight conditions, which describe the lower and upper bound of the unaugmented dutch roll frequency
were investigated. They are the landing approach 33 degrees flaps down and maximum dynamic pressure conditions where the dutch roll frequencies are 0.12 cps and 0.20 cps respectively.

The yaw damper system performance for the above two conditions is summarized in Figures 34 and 35. If the system components are within the design tolerances, the system performances as described by the envelope in the figures is satisfactory. The upper and lower bounds of the envelope are the gain tolerance limits while the left and right bounds are the phase angle tolerance limits.

10. Failure Mode Design Features

At flaps down condition, three sensed signals are used in the yaw damper. The yaw rate, roll angles, and derived roll rate signals each provide an increment of dutch roll damping. If failures occur in any of these signal chains such that one or more of the signals is not driving the rudder, the remaining signals will continue to provide some dutch roll damping. All three of these signals are washed out and thus cannot command a continuous steady-state rudder position.

At flaps up condition, roll angle and derived roll rate signals are removed by the easy on-off circuit and the delayed flaps energized switch. Monitoring circuitry is implemented to detect failure of this flap actuated switching. A red light is illuminated on the yaw damper control panel if flap switching failure occurs.

Any type of malfunction causing a no-signal condition or a sustained large signal will not cause more than a modest transient airplane response. Control excursions are positively limited by hydraulic system velocity and force limits, plus displacement stops effective at all speeds.
IV. AUTOTHROTTLE SYSTEM

A. GENERAL

The 747 is equipped with a single-channel autothrottle system. This system is designed to capture and hold a selected indicated airspeed during terminal area maneuvering, and approach and landing flight regimes by automatically positioning the throttles. It may be used for indicated airspeeds up to 400 kts. Figures 36 and 37 pictorially describe the system. Salient features of the autothrottle are:

1. Clutches in the mechanical drive to the throttles enable the pilot to override the action of the system at any time.
2. The system is operable during manual flight control or while the Autopilot is engaged.
3. The system limits the rate of change of commanded airspeed so that throttle motions occur smoothly.
4. Thrust changes following pitch maneuvering are minimized, and the magnitude of transient thrust changes arising from wind gusts are limited by a gust filter.
5. A throttle retard function, interlocked with the Autopilot Land Mode, is provided to automatically retard the throttles during an autopilot flare.

B. SYSTEM DESCRIPTION

A block diagram of the autothrottle system is shown in Figure 38. A more detailed description is provided below.

1. Command and Airspeed Error Signals

The system engage switch and the speed control knob are situated on the AFCS Mode Select Panel.

The Autothrottle computer receives an airspeed error control signal from the Captain's airspeed indicator. Whenever a difference exists between the pilot-selected airspeed command, indicated in digital form on the AFCS Mode Select Panel, and the actual captain's airspeed, throttle action to reduce the error is commanded. The servo-motor is geared to the throttle levers via clutches which allow the pilot to easily override the system.
AUTOTHROTTLE SPEED CONTROL SEQUENCE

- SPEED COMMAND SIGNAL
- MODE SELECT PANEL
- CAPTAIN'S AIRSPEED INDICATOR
- POST OFFICER'S AIRSPEED INDICATOR
- AUTOthROTTLE ENGAGE LIGHT
- AUTOthROTTLE DISCONNECT BUTTONS
- AUTOthROTTLE ENGAGE SWITCH
- AUTOthROTTLE CLUTCH PACK AND ENGINE CONTROL STAND
- PALM-OPERATED GO-AROUND SWITCHES
- AUTOthROTTLE CHAIN DRIVE
- CONTROL CABLES TO THROTTLES
- SERVO COMMAND AND FEEDBACK
- SERVO MOTOR AND TACH
- LATERAL ACCELEROMETER
- AUTOthROTTLE COMPUTER

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2. **Accelerometer, Attitude, and Elevator Signals**

The longitudinal accelerometer, which is an integral part of the computer, provides rate of change of speed information.

The pitch attitude input, obtained from the Inertial Navigation System, cancels the attitude component of the accelerometer output.

3. **Computation**

The computer utilizes transformer coupling for the A.C. input signals. A differential amplifier is used for the D.C. radio altimeter signal to provide isolation and noise rejection.

All A.C. signals are demodulated prior to shaping, filtering, and error level detection. The processed D.C. signals are summed and modulated for use in the A.C. power amplifier which drives the servo-motor. Prior to engagement, the system is synchronized for inputs other than airspeed error. See block diagram Figure 38.

The basic airspeed error input signal is processed through an acceleration limiting circuit. This circuit asymmetrically limits the airspeed command rate. The rate limited command signal is passed through an asymmetric gain program which reduces the gain for overspeed errors larger than 2 kt to .25 of its value. This gain reduction for large overspeed errors compensates for the fast deceleration of the engine and the higher authority of the throttle in the aft direction. The IAE signal is summed with the compensated longitudinal accelerometer signal and passed through the gust filter to suppress the effects of air turbulence on system activity.

4. **Throttle Control and Limits**

Control of the throttle levers is accomplished by a proportional plus integral servo configuration. The servo proportional response is obtained by integrating the tachometer output.

Switches are provided to limit forward and aft throttle motion. The forward limit position is set to avoid exceeding the maximum allowable engine pressure ratio or temperature; the aft position closely corresponds to the flight idle thrust value. Whenever
the throttles are driven to either limit position, the
integrator is put into a "hold" condition to prevent it
from computing an erroneous throttle position. The system
remains engaged and will drive the throttles out of the
limit position when the appropriate signal is developed.

5. ADI Signal

The demodulated output from the airspeed indicator, filtered
by a 3-second lag, is supplied to fast-slow indicators on the
two Attitude Director Indicators in the cockpit.

6. Airspeed Error Warning

When the Autothrottle is engaged, an airspeed error greater
than ten knots causes the amber Flight Mode Annunciator light
to illuminate.

7. Disengage

A pilot can disengage the system by means of any of the
following:

(a) The engage switch on the AFCS mode selector panel.

(b) Disconnect switches on throttle levers 1 and 4.

(c) Go-around switches on throttle levers 2 and 3.

8. Flare

In conjunction with the Land Mode of the Autopilot, the system
provides automatic throttle retard during the flare maneuver.
The conditions necessary to activate this function are that
the Autopilot must have been armed for flare, the Autopilot
flare must have commenced, and the radio altimeter signal
must be below the trigger altitude. The logic requirements
prevent inadvertent operation of the retard function.

9. Test

Incorporated in the computer is an automatic test arrangement
which can isolate a failure to the computer or servo-motor
without requiring the use of supplementary ground-test equipment.
This test is accomplished by means of a rotary switch and a push-
button switch located on the front panel of the computer. The
rotary switch is used to select the unit to be tested (i.e., the
computer or the servo-motor) and the push-button initiates
the test. The rotary switch will remain in the selected
position until it is manually returned to the "OFF" position. The Autothrottle system cannot be engaged, and a steady red warning light shows on the front panel (and in the cockpit) when the switch is not in the "OFF" position.

The test results are indicated by lights located on the computer's front panel. A "test-in-progress" light is illuminated on initiation of a test; and a "go" light is illuminated upon successful completion. A failure is indicated by the "test-in-progress" light being extinguished without a "go" indication.

The equipment functionally tests the entire Autothrottle system in approximately one minute. The testing is performed by injecting test signals into the various computer inputs while monitoring both the input and output of the feedback integrator in the servo loop. If each of these monitoring devices yields the proper indication, then the test step is completed and the next step is performed. If each test step is successful, the program will proceed to the last step and yield a "go" indication is withheld.
APPENDIX E - REVISED SIMULATION DATA

The data in this section contains revisions that Boeing recommends incorporating in the NASA simulation.
SUMMARY OF AREAS AND DIMENSIONS

<table>
<thead>
<tr>
<th>ITEM</th>
<th>VALUE</th>
<th>DIMENSION</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing Area (S)</td>
<td>5500</td>
<td>Ft.²</td>
</tr>
<tr>
<td>Wing Mean Aerodynamic Chord (MAC)</td>
<td>27.31</td>
<td>Ft.</td>
</tr>
<tr>
<td>Wing Span (b)</td>
<td>195.68</td>
<td>Ft.</td>
</tr>
<tr>
<td>Wheel Base</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wing Gear</td>
<td>78.96</td>
<td>Ft.</td>
</tr>
<tr>
<td>Body Gear</td>
<td>88.96</td>
<td>Ft.</td>
</tr>
<tr>
<td>Wheel Tread</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wing Gear</td>
<td>36.16</td>
<td>Ft.</td>
</tr>
<tr>
<td>Body Gear</td>
<td>12.5</td>
<td>Ft.</td>
</tr>
<tr>
<td>Effective Engine Moment Arms</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Inboard</td>
<td></td>
<td></td>
</tr>
<tr>
<td>YEI</td>
<td>39.6</td>
<td>Ft.</td>
</tr>
<tr>
<td>ZEI</td>
<td>3.3 (air)</td>
<td>8.5 (ground)</td>
</tr>
<tr>
<td>Outboard</td>
<td></td>
<td></td>
</tr>
<tr>
<td>YE0</td>
<td>69.4</td>
<td>Ft.</td>
</tr>
<tr>
<td>ZE0</td>
<td>3.1 (air)</td>
<td>4.8 (ground)</td>
</tr>
</tbody>
</table>

Note: The transition between the ground and air values for the effective engine pitching arms, ZEI and ZE0, is a function of the averaged main landing gear compression ratio, η.

For 0 ≤ η ≤ 1: \[ ZE0 = ZEAIR + ηΔZE0 \]
\[ ZEI = ZEAIR + ηΔZEI \]

where \[ ΔZE0 = ZEOGROUND - ZEAIR = 1.7 \text{ Ft.} \]
\[ ΔZEI = ZEIGROUND - ZEAIR = 0.2 \text{ Ft.} \]

and \[ η = \frac{1}{18n} \sum_{n=1}^{6} \text{Main Landing Gear OIeO Compression (inches)} \]

where n = number of main landing gears.
NOTE 1. LOW SPEED

ANGLE OF ATTACK FOR STICK SHAKER ACTUATION

THE BOEING COMPANY

CALC | LOW | REVISED DATE |
-----|------|--------------|
CHECK | FOSTER | 1-24-68 | LOW | 4-4-69 |
APR | | | LOW | 1-28-70 |
APR | | | SYSTROM | 6-30-70 |
INK | ODEGARD | 6-30-70 |

REF: P 2.0-25
NOTE: 1. LOW SPEED

2. USE DATA ON P. 20-36 FOR FLAPS UP.

---

**INITIAL BUFFET**

**ANGLE OF ATTACK FOR INITIAL BUFFET**

**THE BOEING COMPANY**

**PAGE** 19-0-6

---

<table>
<thead>
<tr>
<th>CALC</th>
<th>LOW</th>
<th>REVI</th>
<th>DATE</th>
</tr>
</thead>
<tbody>
<tr>
<td>CHECK</td>
<td>1-24-68</td>
<td>LOW</td>
<td>6-4-69</td>
</tr>
<tr>
<td>APR</td>
<td>LOW</td>
<td>1-29-70</td>
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<tr>
<td>APR</td>
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<tr>
<td>INK</td>
<td>ODEGARD</td>
<td>6-30-70</td>
<td></td>
</tr>
</tbody>
</table>

**REF.: P. 20-36**
NOTE: FOR SPEED BRAKES AT THE GROUND STOP, USE THE IN-FLIGHT DETENT CURVES FOR PANELS 1, 2, 3, 4, 8, 9, 10. FOR PANELS 11, 12, USE THE CURVE FOR PANELS 3, 4. FOR PANELS 9, 10, PANELS 6, 7, REMAIN AT 20° FOR ALL WHEEL ANGLES.

INTERMEDIATE IN-FLIGHT SPEED BRAKE (SPEED BRAKE HANDLE = 21.5°)

REFERENCE WHEEL ANGLE, \( \delta_{wref} \) - DEG

LATERAL CONTROL
SPOILER PROGRAM AT COMBINED LATERAL CONTROL - SPEED BRAKES

THE BOEING COMPANY
NOTE
1. Speed brake handle friction force = 20 lb. pull, 10 lb. push.


3. Speed brakes beyond the in-flight detent are available only on the ground.

---

LATERAL CONTROL
SPOILERS - SPEED BRAKE PROGRAM

THE BOEING COMPANY
APPENDIX F - AIRPLANE RESPONSE TO CONTROL INPUTS

Time histories of airplane response to control inputs are presented in the following pages.

<table>
<thead>
<tr>
<th>CONTROL</th>
<th>CONDITION</th>
<th>PAGE</th>
</tr>
</thead>
<tbody>
<tr>
<td>Elevator</td>
<td>Climbout</td>
<td>20.0-2</td>
</tr>
<tr>
<td></td>
<td>Approach</td>
<td>20.0-3, 20.0-4, 20.0-5</td>
</tr>
<tr>
<td>Aileron</td>
<td>Climbout</td>
<td>20.0-6</td>
</tr>
<tr>
<td></td>
<td>Approach</td>
<td>20.0-7, 20.0-8</td>
</tr>
<tr>
<td>Rudder</td>
<td>Approach</td>
<td>20.0-9, 20.0-10</td>
</tr>
<tr>
<td></td>
<td>Brake Release and Acceleration</td>
<td>20.0-11</td>
</tr>
<tr>
<td></td>
<td>Rotate and Initial Climbout</td>
<td>20.0-12, 20.0-13, 20.0-14</td>
</tr>
</tbody>
</table>
SC4020 PLOT 01/21/70 0940
747 SIMULATOR
FLIGHT DECK VIBRATION ENVIRONMENT
747-100
RA101
1P3-300A3
VOL. 11
COND NO 1 14 051 002 0 TEST 004-03
THE BOEING COMPANY
TAFE 16.0 7
STALL CHARACTERISTICS
GROSS WEIGHT 552,000 LBS
CG 32.0 PERCENT MAC
FLAP SET 0 DEG—GEAR UP
ENTRY RATE 1.0 KNOTS/SEC
APPENDIX - DERIVATION OF LANDING GEAR EQUATIONS

**Gear Height**

The complete equations for determining landing gear height are derived by methods of descriptive geometry as illustrated in Figure 21.

Let the coordinates of the axle of the strut be $X_L$, $Y_L$, and $Z_L$ and $r$ be the tire radius. With zero bank angle, pitch the aircraft about the c.g. to an angle $\theta_B$ as shown in the side view of Figure 21. The height of the tire relative to the c.g. due to the pitch angle $\theta_B$ is

$$h_\theta = X_L \sin \theta_B - Z_L \cos \theta_B - r.$$  

After the pitch rotation $\theta_B$, view the aircraft as shown in the rear view of Figure 21 and bank the aircraft to $\phi_B$. In the rear view the change in vertical distance from the c.g. to tire is

$$Y_L \sin \phi_B + (Z_L + r) \cos \phi_B - (Z_L + r).$$

Resolving this displacement of the tire relative to the c.g. from the rear view to the side view and into a change in height gives

$$h_\phi = \left[ Y_L \sin \phi_B + (Z_L + r) \cos \phi_B - (Z_L + r) \right] \cos \theta_B$$

or

$$h_\phi = \left[ Y_L \sin \phi_B + (Z_L + r) (\cos \phi_B - 1) \right] \cos \theta_B.$$  

In the NASA simulation $(Z_L + r)$ is replaced by an equivalent distance $S_z$. This simplification assumes the tire will contact the runway at a point aligned with the oleo strut as illustrated in Figure 22.

The total height of the tire above the runway is then given by

$$h_G = h + h_\theta - h_\phi.$$  

If $h_G$ is negative, the gear will be in contact with the runway and the oleo strut compression $\Delta S_T$ is obtained from

$$\Delta S_T = \frac{h_G}{\cos \theta_B \cos \phi_B}.$$
1. \( h_\theta = X_L \sin \theta_B - Z_L \cos \theta_B - r \)
2. \( h_\phi = \left[ Y_L \sin \phi_B + (Z_L + r) \cos \phi_B - 1 \right] \cos \theta_B \)
3. \( h_G = h + h_\theta - h_\phi \)
4. \( \Delta T = \frac{h_G}{\cos \theta_B \cos \phi_B} \)

**Derivation of Equations for Determining Gear Height**

Figure 21
Tire Forces

The orthogonal forces generated by a tire in contact with the runway are shown in Figure 23.

The tire drag force vector $F_\mu$ is aligned by the intersection of a plane through the tire tread and the ground plane. The normal force vector, $F_N$, is perpendicular to the ground plane. The side force, $F_s$, generated by the tire is perpendicular to the plane containing $F_\mu$ and $F_N$. The tire forces are resolved through the aircraft pitch attitude $\theta_B$ and bank angle $\phi_B$ to obtain the body axes forces exerted on the aircraft. The resultant forces in body axes are derived by descriptive geometry methods as illustrated in Figure 24 for normal force, $F_N$, and in Figure 25 for drag force, $F_\mu$, and side force, $F_s$.

In the side view of Figure 24, prior to any axes rotation, the normal force vector $F_N$ acts downward, perpendicular to the ground plane. If the aircraft is pitched through an angle $\theta_B$, the normal force in body axes due to the pitch rotation is

\[
F_{XB\theta} = -F_N \sin \theta_B
\]

\[
F_{YB\theta} = 0
\]

\[
F_{ZB\theta} = F_N \cos \theta_B
\]
RESOLUTION OF WHEEL NORMAL FORCE INTO AIRPLANE BODY AXES

FIGURE 24
If the aircraft is viewed from the rear, as shown by the rear view of Figure 24, and is rotated through a bank angle $\phi_B$, the normal force in body axes is

1. $F_{XB} = -F_N \sin \theta_B$
2. $F_{YB} = F_N \cos \theta_B \sin \phi_B$
3. $F_{ZB} = F_N \cos \theta_B \cos \phi_B$

Equations (1) through (3) represent the resolution of the normal force into body axes.

In the plan view of Figure 25, prior to any axes rotation, the gear drag force acts in the direction of tire rotation and the side force acts perpendicular to the side of the tire. The wheel steering angle is $\delta_S$. The forces resolved into body axes are

$$
F_{X_{B_0}} = F \mu \cos \delta_s - F_s \sin \delta_s
$$
$$
F_{Y_{B_0}} = F \mu \sin \delta_s + F_s \cos \delta_s
$$
$$
F_{Z_{B_0}} = 0
$$

If the aircraft is pitched through an angle $\theta_B$, as shown in the side view of Figure 25, the forces resolved in body axes are

$$
F_{XB_\theta} = (F \mu \cos \delta_s - F_s \sin \delta_s) \cos \theta_B
$$
$$
F_{YB_\theta} = F \mu \sin \delta_s + F_s \cos \delta_s
$$
$$
F_{ZB_\theta} = (F \mu \cos \delta_s - F_s \sin \delta_s) \sin \theta_B
$$

The bank angle axes rotation as shown in the rear view of Figure 25 gives the final value of wheel drag and side force in body axes as

$$
F_{XB} = (F \mu \cos \delta_s - F_s \sin \delta_s) \cos \theta_B
$$
$$
F_{YB} = (F \mu \sin \delta_s + F_s \cos \delta_s) \cos \phi_B
+ (F \mu \cos \delta_s - F_s \sin \delta_s) \sin \theta_B \sin \phi_B
$$
$$
F_{ZB} = -(F \mu \sin \delta_s + F_s \cos \delta_s) \sin \phi_B
+ (F \mu \cos \delta_s - F_s \sin \delta_s) \sin \theta_B \cos \phi_B
$$

Equations (4) through (6) when combined with Equations (1) through (3) represent the forces in aircraft body axes generated by a tire contacting the runway.

For the NASA simulation, small angle approximations were used to resolve the forces into body
RESOLUTION OF WHEEL DRAG AND SIDE FORCE INTO AIRPLANE BODY AXES

FIGURE 25
axes. The normal force Equations (1) through (3) were simplified to

\begin{align*}
(7) \quad F_{X_B} &= -F_N \theta_B \\
(8) \quad F_{Y_B} &= F_N \phi_B \\
(9) \quad F_{Z_B} &= F_N = F_{GZ}
\end{align*}

where $F_{GZ}$ = forces generated by compression of the oleo strut.

The drag force equations (4) through (6) were simplified to

\begin{align*}
(10) \quad F_{X_B} &= F \mu - F_s \delta_s \\
(11) \quad F_{Y_B} &= F \mu \delta_s + F_s \\
(12) \quad F_{Z_B} &= -F_s \phi_B + F \mu \theta_B
\end{align*}

Combining Equations (7) through (12) gives the final equations used for the NASA simulation

\begin{align*}
F_{X_B} &= F \mu - F_s \delta_s - F_{GZ} \theta_B \\
F_{Y_B} &= F \mu \delta_s + F_s + F_{GZ} \phi_B \\
F_{Z_B} &= F \mu \theta_B - F_s \phi_B + F_{GZ}
\end{align*}