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REAL-TIME SIMULATION PROGRAM FOR DE HAVILLAND (CANADA) "BUFFALO" AND "TWIN OTTER" STOL TRANSPORTS

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16. Abstract

Simulation models of two representative STOL aircraft - the DeHavilland (Canada) "Buffalo" and "Twin Otter" transports - have been generated.

The aircraft are described by means of non-linear equations that will accommodate gross changes in angle of attack, pitch angle, flight path angle, velocity, and power setting. Aircraft motions in response to control inputs and external disturbances are related to Earth-fixed coordinates. The equations are programmed to run in "real time" so that they can be used in conjunction with a manned cockpit simulator. Provisions are made for pilot control inputs to the simulation, and conventional panel display parameters are generated.

The report includes representative simulation results which demonstrate that the simulation is an adequate representation of the two STOL aircraft being modeled.

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LIST OF SYMBOLS

a	A	Aircraft lift curve slope	rad^{-1}
AR	AR	aspect ratio of wing = b^2/S	-
B_{nm}	BNM	elements of A-frame to L-frame transformation matrix	-
b	B	wing span	ft
c	C	mean chord of wing	ft
C_D	CD	aircraft drag coefficient	-
C_{D_f}	CDF	aircraft parasite drag coefficient	-
ΔC_D	DELCD	aircraft drag coefficient less wing drag coefficient	
C_L	CL	aircraft lift coefficient	-
C_{L_o}	CLO	trimmed aircraft lift coefficient	-
C_{m_t}	CMT	pitching moment coefficient which may be made variable to shape trim δ_e vs V_R curve ($C_{m_t} = 0$ in this report)	
C_{m_q}	CMQ	pitching moment coefficient due to pitch rate	-
C_{m_α}	CMALF	pitching moment coefficient due to angle of attack	-
$C_{m_{\dot{\alpha}}}$	CMDALF	pitching moment coefficient due to angle of attack rate	-
$C_{m_{\delta e}}$	CMDLE	pitching moment coefficient due to elevator deflection	-
C_{ℓ_p}	CLP	rolling moment coefficient due to roll rate	-
C_{ℓ_β}	CLB	rolling moment coefficient due to sideslip angle	-
$C_{\ell_{\delta_a}}$	CLDLA	rolling moment coefficient due to aileron deflection	-

C_{ℓ_r}	CLR	rolling moment coefficient due to yaw rate	-
$C_{\ell_{r_{fin}}}$	CLRFIN	fin contribution to rolling moment coefficient due to yaw rate	-
C_{n_p}	CNP	yawing moment coefficient due to roll rate	-
$C_{n_{p_{fin}}}$	CNPFIN	fin contribution to yawing moment coefficient due to yaw rate	-
C_{n_r}	CNR	yawing moment coefficient due to yaw rate	-
$C_{n_{r_{fin}}}$	CNRFIN	fin contribution to yawing moment coefficient due to yaw rate	-
C_{n_β}	CNB	yawing moment coefficient due to sideslip angle	-
$C_{n_{\delta_r}}$	CNDLR	yawing moment coefficient due to rudder deflection	-
C_{y_p}	CYP	side force coefficient due to roll rate	-
C_{y_r}	CYR	side force coefficient due to yaw rate	-
C_{y_β}	CYB	side force coefficient due to sideslip angle	-
C_{T_1}	CT1	empirical coefficient in thrust equation	fps^{-1}
C_{T_2}	CT2	empirical coefficient in thrust equation	fps^{-2}
D	DRAG	aircraft drag	lbs
e	E	aircraft efficiency factor	-
g	G	gravitional constant = 32.2	ft/sec^2
h	H	altitude = $- z_L$	ft
h_{ATM}	HATM	characteristic density altitude of atmosphere	ft

$\bar{i}_{()}, \bar{j}_{()}, \bar{k}_{()}$	-	unit vectors along the X,Y, and Z, axes of the () coordinate frame, respectively	-
IAS	AIRSPD	indicated airspeed	mph
I_x, I_y, I_z	IX,IY,IZ	aircraft rolling, pitching, and yawing moment of inertia, respectively	slug-ft^2
J_{xz}	-	product of inertia = $\int$ xz dm	slug-ft^2
L	LIFT	aircraft lift	lbs
L,M,N	-	scalar component of the applied external moment along the X_A, Y_A, and Z_A axis, respectively	ft-lbs
L_p	LP	rolling moment due to roll rate	ft-lbs/$\frac{rad}{sec}$
L_r	LR	rolling moment due to yaw rate	ft-lbs/$\frac{rad}{sec}$
L_v	LV	rolling moment due to sideslip velocity	ft-lbs/fps
L_{δ_a}	LDLA	rolling moment due to aileron deflection	ft-lbs/rad
1/m	OOM	1/aircraft mass	slugs^{-1}
N_p	NP	yawing moment due to roll rate	ft-lbs/$\frac{rad}{sec}$
N_r	NR	yawing moment due to yaw rate	ft-lbs/$\frac{rad}{sec}$
N_v	NV	yawing moment due to sideslip velocity	ft-lbs/fps
N_{δ_r}	NDLR	yawing moment due to rudder deflection	ft-lbs/rad
P,Q,R	P,Q,R	scalar components of the angular rotation vector of the aircraft along the X_A, Y_A, and Z_A axis, respectively	rad/sec
q	DYN	dynamic pressure	lbs/ft^2
S	S	wing area	ft^2

T	THRUST	aircraft thrust	lbs
-	TMDLE	elevator input delay (See Section IV-B)	sec
-	TMTHR	throttle input delay (See Section IV-B)	sec
T_{static}	TSTAT	aircraft thrust at zero airspeed	lbs
U,V,W	U,V,W	scalar component of aircraft velocity along the X_A, Y_A, and Z_A axis, respectively	fps
U_W,V_W,W_W	UW,VW,WW	scalar component of aircraft with respect to airmass along the X_A, Y_A, and Z_A axis, respectively	fps
V_R	VR	resultant velocity of aircraft with respect to airmass	fps
W	WEIGHT	aircraft weight	lbs
X,Y,Z	-	scalar component of the applied external non-gravitational force along the X_A, Y_A, and Z_A axis, respectively	lbs
$X_{()},Y_{()},Z_{()}$	-	axes defining the () coordinate frame	-
x_L,y_L,z_L	X,Y,-H	displacements along the respective axes of the L coordinate frame	ft
$\dot{x}_L,\dot{y}_L,\dot{z}_L$	X DOT, Y DOT, -H DOT	velocities along the respective axes of the L coordinate frame	fps
$\dot{x}_{W_L},\dot{y}_{W_L},\dot{z}_{W_L}$	XSS,YSS, ZSS	steady state airmass velocity along the X_L, Y_L, and Z_L axes, respectively	fps
Y_p	YP	side force due to roll rate	$\text{lbs}/\frac{\text{rad}}{\text{sec}}$
Y_r	YR	side force due to yaw rate	$\text{lbs}/\frac{\text{rad}}{\text{sec}}$
Y_v	YV	side force due to sideslip velocity	lbs/fps

α	ALF	angle from the remote wind vector V_R to the X_A axis	rad
$\dot{\alpha}$	DALF	$d\alpha/dt$	rad/sec
α_B	-	angle from the remote wind vector V_R to the X_B axis	rad
α_{B_O}	ALFBO	angle between the body-fixed X_A and X_B axes	rad
$\alpha_{B_{OL}}$	ALFBOL	value of α_B for which no lift is developed by the aircraft	rad
β	BETA	aircraft sideslip angle	rad
γ	-	angle from the horizontal reference line to the remote wind vector V_R: $\gamma = \Theta - \alpha$	rad
δ_a	DLA	aileron deflection	rad
δ_e	DLE	elevator deflection	rad
δ_r	DLR	rudder deflection	rad
Θ	THETA	(See definition of Euler angles Ψ,Θ,Φ below)	
Θ_B	-	angle from the horizontal reference line to the X_B axis	rad
ξ	THROT	pilot throttle input as fraction of maximum input	-
ρ	RHO	atmospheric air density	slugs/ft^3
ρ_o	RHOSEA	atmospheric air density at sea level, std day	slugs/ft^3
σ	SIG	ρ/ρ_o	-
Ψ,Θ,Φ	PSI, THETA PHI	Euler angles relating L, C, and A coordinate frames (further defined in Figure 3)	rad

SUBSCRIPTS

A	aircraft body coordinate frame
B	body reference coordinate frame
C	Earth-aircraft control coordinate frame
L	Earth local-vertical coordinate frame
cr	design economy cruise condition
o	equilibrium or reference condition
OL	zero lift value

I Introduction

Simulation models of two representative STOL aircraft have been generated. The models are documented in this report.

The computer simulation is to be used as a tool in the development of STOL terminal area guidance and navigation systems.

This intended use has determined the form of the simulation: The aircraft are described by means of non-linear equations that will accomodate gross changes in angle of attack, pitch angle, flight path angle, velocity, and power setting. Aircraft motions in response to control inputs and external disturbances are related to Earth-fixed coordinates. The equations are programmed to run in "real time" so that they can be used in conjunction with a manned cockpit simulator. Provisions are made for pilot control inputs to the simulation, and conventional panel display parameters are generated.

The aircraft which are modeled - the DHC "Twin Otter" and the DHC "Buffalo" - are described in Figures 1 and 2, respectively. They were selected as representative light and medium propeller-driven STOL transports. Their selection does not imply that there are not other STOL aircraft representative of these classes. Similarly, the material contained in this report should not be used as the basis for an evaluation of the flying qualities of the "Buffalo" or "Twin Otter" or of the suitability of these aircraft for any specific mission.

The aircraft are modeled only to the extent necessary to yield a representative vehicle model controllable by a guidance or navigation system. Certain simplifying assumptions - specified in the following sections - are made. These assumptions are justified for the present model application but may render the model unsuitable for other possible applications.

The simulation is described in detail in the following sections of this report. In Section II, all required equations are developed. Section III tabulates numerical values to be used in these equations for the "Buffalo" and "Twin Otter".

The simulation program is presented in Section IV. A listing of all computer statements is included. Finally, in Section V, representative simulation results are shown. These results demonstrate that the simulation is an adequate representation of the two STOL aircraft.

II Description of Mathematical Model

The mathematical model consists of all equations required to describe the motions of the aircraft in space resulting from external disturbances, control inputs, and the aircraft's aerodynamic characteristics. These equations are presented in this Section. First, however, it is necessary to define the reference coordinate frames to be used.

IIA Definition of Reference Coordinate Frames

Reference coordinate frames to be used in this analysis are defined in this section. Insofar as possible, axis systems have been defined so that senses of rotation and translation are similar for small rotations. Positive force, moment, and motion vector components are defined to be in the positive sense of the axis. To the largest extent possible, the symbols and conventions used are consistent with those in common usage in the guidance and control fields and with those used by NASA for aircraft stability and control work.

The Earth Local-Vertical Frame (L) is a local geographic frame. Its origin is fixed at a point on the Earth's surface with Z_L along the vertical defined by the local gravity vector (positive downward), X_L parallel to geographic North (positive to the North), and Y_L parallel to geographic East (positive to the East).

The Aircraft Body Coordinate Frame (A) is fixed to the aircraft and rotates and translates with the aircraft. Its origin is the center of mass of the aircraft. The X_A axis is chosen in a forward direction in the plane of symmetry that is parallel to the initial or equilibrium direction of the remote wind. Thus the A-frame axes, by the commonly accepted definition, are "stability axes". Because the X_A axis is initially aligned with the remote wind, the initial angle of attack $\alpha(0) = \alpha_o$ is zero. (In this report, Θ and α when not subscripted to indicate reference frame, are assumed to be referenced to the A-frame. Further, since in the simulation documented in this report the aircraft is placed in equilibrium at t = 0, "equilibrium" and "initial" conditions are equivalent.) The Y_A axis is normal to the aircraft's plane of symmetry (positive to the right), and the Z_A axis is in the plane of symmetry (positive downward) and orthogonal to the X_A and Y_A axes. The A-frame is related to the L-frame (and to the next-defined C-frame) in Figure 3.

<u>The Earth-Aircraft Control Coordinate Frame (C)</u> is also centered at the center of mass of the aircraft. The Z_C axis is aligned with the local gravity vector (positive downward) and is therefore parallel to the Z_L axis. The X_C axis is the intersection of the horizontal plane with the vertical plane containing the X_A axis. The Y_C axis completes the orthogonal right-hand system. The C-frame is an intermediate frame needed to define the Euler angles describing the relationship between the Earth local-vertical (L) frame and the Aircraft body (A) frame. In their order of rotation (which must be preserved) the Euler angles are defined as:

1. Heading (Ψ): angle of rotation about Z_L from X_L to X_C;
2. Pitch (Θ): angle of rotation about Y_C from X_C to X_A;
3. Roll (Φ): angle of rotation about X_A from Y_C to Y_A

These Euler angle rotations are shown in Figure 3.

<u>The Body Reference Coordinate Frame (B)</u> is introduced and defined in this report primarily to clarify the definition of trim angle of attack. Like the A-frame, this frame is fixed to, and translates and rotates with, the aircraft and has as its origin the center of mass of the aircraft. The X_B axis, however, is fixed in a forward direction in the plane of symmetry parallel to a fuselage waterline or datum line. The X_B axis is displaced from the X_A axis by the angle α_{B_O}. The Y_B axis coincides with the Y_A axis, and the Z_B axis (positive downward) forms an orthogonal set.

The angle α_{B_O} is sometimes called α_{trim}, the trimmed angle of attack. It is the angle between the initial (equilibrium) remote wind vector and the X_B axis. Unlike α_O, it has a non-zero value. It is evident from Figure 4 that

$$\alpha_B = \alpha + \alpha_{B_O}$$

$$\Theta_B = \Theta + \alpha_{B_O} \qquad (1)*$$

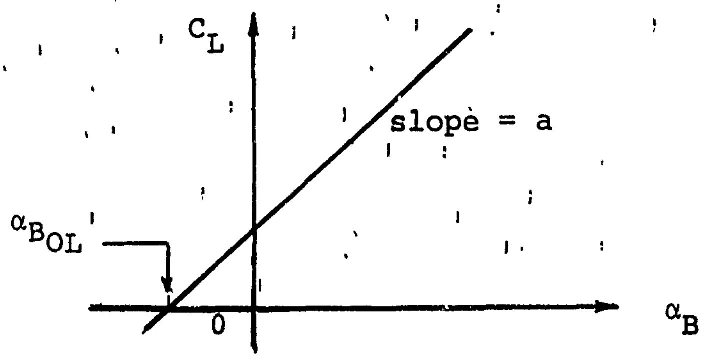
*Numbered equations are mechanized in the simulation. Other equations are introduced as necessary for purposes of clarification, but are not numbered.

and, for equilibrium level flight, that

$$\Theta_{B_o} = \alpha_{B_o} \ .$$

The trim angle α_{B_o} can be approximated in the following fashion in the absence of wind tunnel or flight test data:

Assuming a constant aircraft lift curve slope, a, sketch the aircraft's lift curve:



From the sketch it is apparent that

$$C_L = a\ (\alpha_B - \alpha_{B_{OL}})$$

or, at equilibrium,

$$C_{L_o} = a\ (\alpha_{B_o} - \alpha_{B_{OL}})$$

Next, assume that wing incidence has been chosen by the aircraft manufacturer to produce a level fuselage attitude ($\alpha_{B_o} = 0$) when the aircraft is in flight at "Economy Cruise Speed" at 10000 ft and at an arbitrarily - chosen average gross weight. Using the relation $W_{cr} = C_{L_{cr}}\, q_{cr} S$, calculate the lift coefficient at the flight condition. The angle of attack for zero lift can then be calculated from the above equation as

$$\alpha_{B_{OL}} = -\frac{C_{L_{cr}}}{a}$$

The same equation can be manipulated to give an expression for the trim angle α_{B_O} at any other trim lift coefficient:

$$\alpha_{B_O} = \frac{C_{L_O}}{a} + \alpha_{B_{OL}} \tag{2}$$

(In Appendix B of Reference 1, $C_{L_{cr}}$ was estimated to be .44 for the "Buffalo" and .48 for the "Twin Otter". For both aircraft, a = 5.2/rad, so

$$\begin{aligned} \alpha_{B_{OL}} &= -.085 = -4.8° \text{ (Buffalo)} \\ &= -.092 = -5.3° \text{ (Twin Otter)} \end{aligned}$$

These values are used in this report.)

IIB Velocity Resolutions

Use must be made of the above-defined Euler angles to relate a vector quantity in the A-frame to its components in the L-frame and vice versa. In general, a vector $\bar{R}$ can be resolved into its A-frame or L-frame components:

$$\begin{aligned} \bar{R} &= R_{X_A} i_A + R_{Y_A} j_A + R_{Z_A} k_A \\ &= R_{X_L} i_L + R_{Y_L} j_L + R_{Z_L} k_L \end{aligned}$$

where i, j, and k are unit vectors in the indicated frames.

L-frame components of $\bar{R}$ can be expressed in terms of A-frame components of $\bar{R}$ and the Euler angles:

$$\begin{bmatrix} R_{X_L} \\ R_{Y_L} \\ R_{Z_L} \end{bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{13} \\ B_{21} & B_{22} & B_{23} \\ B_{31} & B_{32} & B_{33} \end{bmatrix} \begin{bmatrix} R_{X_A} \\ R_{Y_A} \\ R_{Z_A} \end{bmatrix}$$

where $B_{11} = \cos \Psi \cos \Theta$ (3)

$$B_{12} = \cos \Psi \sin \Theta \sin \Phi - \sin \Psi \cos \Phi \tag{4}$$

$$B_{13} = \cos \Psi \sin \Theta \cos \Phi + \sin \Psi \sin \Phi \tag{5}$$

$$B_{21} = \sin \Psi \cos \Theta \tag{6}$$

$$B_{22} = \sin \Psi \sin \Theta \sin \Phi + \cos \Psi \cos \Phi \tag{7}$$

$$B_{23} = \sin \Psi \sin \Theta \cos \Phi - \cos \Psi \sin \Phi \tag{8}$$

$$B_{31} = -\sin \Theta \tag{9}$$

$$B_{32} = \cos \Theta \sin \Phi \tag{10}$$

$$B_{33} = \cos \Theta \cos \Phi \tag{11}$$

Conversely, A-frame components of any vector $\bar{R}$ can be expressed in terms of L-frame components:

$$\begin{bmatrix} R_{X_A} \\ R_{Y_A} \\ R_{Z_A} \end{bmatrix} = \begin{bmatrix} B_{11} & B_{21} & B_{31} \\ B_{12} & B_{22} & B_{32} \\ B_{13} & B_{23} & B_{33} \end{bmatrix} \begin{bmatrix} R_{X_L} \\ R_{Y_L} \\ R_{Z_L} \end{bmatrix}$$

Thus, in the simulation, the A-frame components of aircraft velocity ($\dot{x}_A \equiv U$, $\dot{y}_A \equiv V$, and $\dot{z}_A \equiv W$) are computed and used to obtain velocity components with respect to the ground:

$$\dot{x}_L = B_{11} U + B_{12} V + B_{13} W \qquad \text{(fps)} \tag{12}$$

$$\dot{y}_L = B_{21} U + B_{22} V + B_{23} W \qquad \text{(fps)} \tag{13}$$

$$\dot{h} = -\dot{z}_L = -B_{31} U - B_{32} V - B_{33} W \qquad \text{(fps)} \tag{14}$$

IIC Provisions for Atmospheric Disturbances (Winds)

Winds are input into the simulation in the L-frame. Components are $\dot{x}_{W_L}$ (positive North), $\dot{y}_{W_L}$ (positive East), and $\dot{z}_{W_L}$ (positive downward). The winds are resolved into A-frame components in equations 15-17 in order to compute airspeed components:

$$U_W = U - [B_{11} \dot{x}_{W_L} + B_{21} \dot{y}_{W_L} + B_{31} \dot{z}_{W_L}] \qquad \text{(fps)} \tag{15}$$

$$V_W = V - [B_{12} \dot{x}_{W_L} + B_{22} \dot{y}_{W_L} + B_{32} \dot{z}_{W_L}] \qquad \text{(fps)} \tag{16}$$

$$W_W = W - [B_{13} \dot{x}_{W_L} + B_{23} \dot{y}_{W_L} + B_{33} \dot{z}_{W_L}] \qquad \text{(fps)} \tag{17}$$

Material contained in this report is sufficient to allow introduction of steady state wind components. The desired winds are simply input as $\dot{x}_{W_L}$, $\dot{y}_{W_L}$, and $\dot{z}_{W_L}$. The report does not document wind gust or wind shear models. However, these models, when developed, can be readily incorporated into the simulation with only minor modifications to the program being required.

IID Airframe Equations of Motion

In Reference 1, general 6 degree of freedom airframe equations of motion were developed as

$m\,[\dot{U} + QW - RV + g \sin \Theta] = X$	(longitudinal force)
$m\,[\dot{V} + RU - PW - g \cos \Theta \sin \Phi] = Y$	(side force)
$m\,[\dot{W} + PV - QU - g \cos \Theta \cos \Phi] = Z$	(normal force)
$I_x \dot{P} + (I_z - I_y)\, QR - J_{xz}\, (\dot{R} + PQ) = L$	(rolling moment)
$I_y \dot{Q} + (I_x - I_z)\, RP - J_{xz}\, (R^2 - P^2) = M$	(pitching moment)
$I_z \dot{R} + (I_y - I_x)\, PQ - J_{xz}\, (\dot{P} - QR) = N$	(yawing moment)

where the body-axis angular rates P, Q, and R, can be used to obtain Euler angle rates according to the equations

$$\dot{\Psi} = Q \frac{\sin \Phi}{\cos \Theta} + R \frac{\cos \Phi}{\cos \Theta} \qquad \text{(rad/sec)} \tag{18}$$

$$\dot{\Theta} = Q \cos \Phi - R \sin \Phi \qquad \text{(rad/sec)} \tag{19}$$

$$\dot{\Phi} = P + \dot{\Psi} \sin \Theta \qquad \text{(rad/sec)} \tag{20}$$

These nine equations, together with equations 12-14, provide an almost exact description of the motions of an aircraft operating near the Earth's surface. They involve, as shown in Reference 1, only four assumptions:

1. Aircraft mass is constant
2. The Earth can be considered an inertial frame
3. The aircraft is a rigid body
4. The aircraft is symmetrical about its x - z plane.

For purposes of this simulation, the above 6 rigid body airframe equations have been approximated as

$$\dot{U} = RV - QW - g \sin \Theta + X/m \qquad (\text{ft/sec}^2) \qquad (21)$$

$$\dot{V} = PW - RU + g \cos \Theta \sin \Phi + Y/m \qquad (\text{ft/sec}^2) \qquad (22)$$

$$\dot{W} = QU - PV + g \cos \Theta \cos \Phi + Z/m \qquad (\text{ft/sec}^2) \qquad (23)$$

$$\dot{P} = L/I_x \qquad (\text{rad/sec}^2) \qquad (24)$$

$$\dot{Q} = M/I_Y \qquad (\text{rad/sec}^2) \qquad (25)$$

$$\dot{R} = N/I_z \qquad (\text{rad/sec}^2) \qquad (26)$$

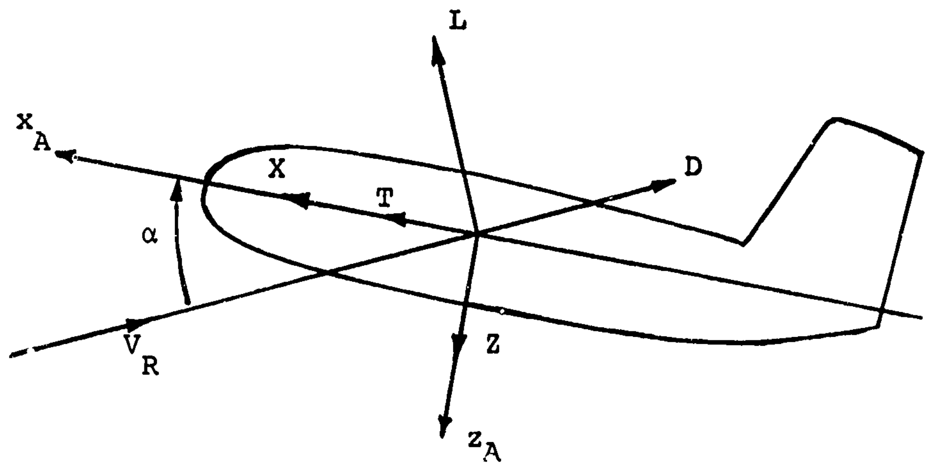
The omitted terms in the moment equations involve either products of angular velocities (e.g. QR) felt to be small compared with other equation terms, or terms containing J_{xz} which will be neglected. Experience has shown that, for purposes of this simulation, these terms can be omitted with negligible effect on results.

The terms X, Y, Z, L, M, and N of equations 21 - 26 represent the aerodynamic forces and moments acting on the aircraft. The lateral terms (Y, L, N) will be expressed in a quasi-linear form (as in Reference 1), but the longitudinal forces and moment (X,Z,M) must be non-linear in order to permit large excursions in forward velocity.

The longitudinal aerodynamic force terms are, from the sketch,

$$X = T - D \cos \alpha + L \sin \alpha \qquad (\text{lbs}) \qquad (27)$$

$$Z = -(L\cos \alpha + D \sin \alpha) \qquad (\text{lbs}) \qquad (28)$$



The terms X_q, Z_q, $Z_{\dot{w}}$, and Z_{δ_e} have been neglected in this analysis because of their small contribution to the overall forces.

It is also assumed that all thrust forces act along the X_A axis. Thus moment effects of thrust changes are neglected, as are forces and moments produced by special lift devices operating within or outside of the propeller slipstream. These effects are neglected because the airframe data required to model them are not available.

Equations 27 and 28 are solved (as are the other simulation equations) once every computer iteration cycle. Thrust, drag, and lift force components are summed to produce resultant X and Z forces acting on the aircraft.

Expressions for the total thrust, lift, and drag forces are next developed.

Thrust is computed from an empirically-derived expression (developed in the appendix) which accounts for the effects of altitude h, airspeed V_R, and throttle setting ξ:

$$T = \frac{\sigma\, T_{static}}{1 + C_{T_1} V_R + C_{T_2} V_R^2} \cdot \xi \qquad \text{(lbs)} \tag{29}$$

where $0 \leq \xi \leq 1.0$,

$$\sigma = e^{-h/h_{atm}}, \qquad \text{(-)} \tag{30}$$

and

$$V_R = [U_w^2 + V_w^2 + W_w^2]^{1/2} \qquad \text{(fps)} \tag{31}$$

Lift and drag are calculated from the standard relationships:

$$L = C_L\, qS \qquad \text{(lbs)} \tag{32}$$

$$D = C_D\, qS \qquad \text{(lbs)} \tag{33}$$

where

$$C_L = C_{L_o} + \alpha a \qquad \text{(-)} \tag{34}$$

$$C_D = C_{D_f} + C_L^2/\pi e AR \qquad \text{(-)} \tag{35}$$

$$q = \frac{1}{2} \rho\, V_R^2 \qquad \text{(lbs/ft}^2\text{)} \tag{36}$$

$$\rho = \sigma\, \rho_o \qquad \text{(sl/ft}^3\text{)} \tag{37}$$

and

$$\alpha = \tan^{-1} W_w/U_w \qquad \text{(rad)} \tag{38}$$

The expression for pitching moment used in the simulation is

$$M = qSc[C_{m_t} + C_{m_\alpha} \alpha + \frac{c}{2V_R} (C_{m_{\dot\alpha}} \dot\alpha + C_{m_q} Q) + C_{m_{\delta_e}} \delta_e] \qquad \text{(ft/lbs)} \tag{39}$$

where the coefficients of the variables are constants. The term C_{m_t} is zero in this report, but is included to facilitate later shaping of the trimmed δ_e vs V_R curve. To do this, C_{m_t} would be made a function of V_R.

Rate of change with time of angle of attack is obtained by differentiating equation 38:

$$\dot{\alpha} = \frac{d}{dt} \left(\tan^{-1} \frac{W_w}{U_w}\right)$$

$$= \frac{U_w \dot{W}_w - W_w \dot{U}_w}{U_w^2 + W_w^2}$$

If the approximation is made that $\dot{U} \approx \dot{U}_w$ and $\dot{W} \approx \dot{W}_w$, the above expression can be manipulated to produce

$$\dot{\alpha} = (\dot{W} - \frac{W_w}{U_w} \dot{U}) \frac{\cos^2\alpha}{U_w} \quad \text{(rad/sec)} \qquad (40)$$

which is the expression used in the simulation.

The lateral force (Y) and moments (L and N) are developed in conventional linearized form (as in Reference 1) except that total variables are used rather than perturbation values, and that coefficients of the lateral variables are made functions of lift and drag coefficient, airspeed, and dynamic pressure, all of which are determined by solution of the longitudinal equations.

The lateral force and moment expressions used in the simulation are:

$$Y = Y_v V_w + Y_r R + Y_p P \quad \text{(lbs)} \qquad (41)$$

$$L = L_v V_w + L_r R + L_p P + L_{\delta_a} \delta_a \quad \text{(ft-lbs)} \qquad (42)$$

$$N = N_v V_w + N_r R + N_p P + N_{\delta_r} \delta_r \quad \text{(ft-lbs)} \qquad (43)$$

The terms Y_{δ_r}, L_{δ_r}, and N_{δ_a}, sometimes included in the lateral equations, have been omitted in the present analysis because of their negligible effects.

The coefficients of these equations are

$$Y_v = \frac{1}{2} \rho V_R S C_{y_\beta} \qquad (\text{lbs/fps}) \qquad (44)$$

$$Y_r = \frac{1}{4} \rho V_R Sb C_{y_r} \qquad (\text{lbs}/\frac{\text{rad}}{\text{sec}}) \qquad (45)$$

$$Y_p = \frac{1}{4} \rho V_R Sb C_{y_p} \qquad (\text{lbs}/\frac{\text{rad}}{\text{sec}}) \qquad (46)$$

$$L_v = \frac{1}{2} \rho V_R Sb C_{\ell_\beta} \qquad (\text{ft-lbs/fps}) \qquad (47)$$

$$L_r = \frac{1}{4} \rho V_R Sb^2 C_{\ell_r} \qquad (\text{ft-lbs}/\frac{\text{rad}}{\text{sec}}) \qquad (48)$$

$$C_{\ell_r} = C_{\ell_{r_{FIN}}} + C_L/4 \qquad (-) \qquad (49)$$

$$L_p = \frac{1}{4} \rho V_R Sb^2 C_{\ell_p} \qquad (\text{ft-lbs}/\frac{\text{rad}}{\text{sec}}) \qquad (50)$$

$$L_{\delta_a} = q Sb C_{\ell_{\delta_a}} \qquad (\text{ft-lbs/rad}) \qquad (51)$$

$$N_v = \frac{1}{2} \rho V_R Sb C_{n_\beta} \qquad (\text{ft-lbs/fps}) \qquad (52)$$

$$N_r = \frac{1}{4} \rho V_R Sb^2 C_{n_r} \qquad (\text{ft-lbs}/\frac{\text{rad}}{\text{sec}}) \qquad (53)$$

$$C_{n_r} = C_{n_{r_{FIN}}} - C_{d_{wing}}/4 \qquad (-) \qquad (54)$$

$$N_p = \frac{1}{4} \rho V_R Sb^2 C_{n_p} \qquad (\text{ft-lb}/\frac{\text{rad}}{\text{sec}}) \qquad (55)$$

$$C_{n_p} = C_{n_{p_{FIN}}} - \frac{C_L}{4} \left(1 - \frac{a}{\pi AR}\right) \qquad (-) \qquad (56)$$

$$N_{\delta_r} = q Sb C_{n_{\delta_r}} \qquad (\text{ft-lbs/rad}) \qquad (57)$$

The equation for sideslip angle is

$$\beta = \tan^{-1} \frac{V_W}{U_W} \qquad \text{(rad)} \qquad (58)$$

Linear and angular rates are integrated to produce the required linear and angular displacements. Initial values of displacements are provided for where necessary:

$$U = U(0) + \int_0^t \dot{U}\, dt \qquad \text{(fps)} \qquad (59)$$

$$V = V(0) + \int_0^t \dot{V}\, dt \qquad \text{(fps)} \qquad (60)$$

$$W = W(0) + \int_0^t \dot{W}\, dt \qquad \text{(fps)} \qquad (61)$$

$$P = \int_0^t \dot{P}\, dt \qquad \text{(rad/sec)} \qquad (62)$$

$$Q = \int_0^t \dot{Q}\, dt \qquad \text{(rad/sec)} \qquad (63)$$

$$R = \int_0^t \dot{R}\, dt \qquad \text{(rad/sec)} \qquad (64)$$

$$\Psi = \int_0^t \dot{\Psi}\, dt \qquad \text{(rad)} \qquad (65)$$

$$\Theta = \int_0^t \dot{\Theta}\, dt \qquad \text{(rad)} \qquad (66)$$

$$\Phi = \int_0^t \dot{\Phi}\, dt \qquad \text{(rad)} \qquad (67)$$

$$x_L = \int_0^t \dot{x}_L\, dt \qquad \text{(ft)} \qquad (68)$$

$$y_L = \int_0^t \dot{y}_L\, dt \qquad (69)$$

$$h = -z_L = h(0) + \int_0^t \dot{h}\, dt \qquad (70)$$

IIE Definition of Required Display Quantities

Provisions are made in the simulation for displaying parameters that are commonly available on a cockpit instrument panel. These parameters are tabulated here (and are defined if they have not been previously defined):

Indicated Airspeed	$IAS = \frac{\sigma^{1/2}}{1.46} V_R$	(mph)
Altimeter Output	h	(ft)
Directional Gyro Output	$57.3\ \Psi$	(deg)
Pitch Attitude Gyro Output	$57.3\ \Theta_B$	(deg)
Roll Attitude Gyro Output	$57.3\ \Phi$	(deg)
Rate of Climb Indicator Output	$\dot{h}/60$	(fpm)
Turn Rate Indicator Output	$57.3\ R$	(deg/sec)
Slip Indicator Output	$\left[\frac{g \cos \Theta \sin \Phi - \dot{V} - RU + PW}{g \cos \Theta \cos \Phi - \dot{W} - PV + QU}\right]$	(rad)

III Tabulation of Numerical Data for "Buffalo" and "Twin Otter"

Numerical data for the two aircraft to be modeled are tabulated in this section. Unless otherwise indicated, the values have been taken from Reference 1. It should be recognized that stability derivative values tabulated here are not based on wind tunnel or flight test results, but have been generated using analytical expressions presented in Reference 1.

Parameter	Value	
	Buffalo	Twin Otter
a, rad^{-1}	5.2	5.2
AR	9.75	10
b,ft	96	65
c,ft	10.1	6.5
C_{D_f}	.032	.039
ΔC_D	.030	.035
C_{m_t}	0	0
C_{m_q}	-35.6	-24.6
C_{m_α}	-.78	-.78
$C_{m_{\dot{\alpha}}}$	-6.05	-6.15
$C_{m_{\delta_e}}$	2.12	1.73
C_{ℓ_p}	-.53	-.53

Parameter	Value	
	Buffalo	Twin Otter
C_{ℓ_β}	-.125	-.103
$C_{\ell_{\delta_a}}$	.20	.38
$C_{\ell_{r_{fin}}}$	.038	.033
$C_{n_{p_{fin}}}$	.025	.033
$C_{n_{r_{fin}}}$	-.169	-.168
C_{n_β}	.101	.121
$C_{n_{\delta_r}}$	.107	.107
C_{y_p}	-.055	-.085
C_{y_r}	.368	.429
C_{y_β}	-.362	-.492
C_{T_1}, fps^{-1} (1)	.00370	.00378
C_{T_2}, fps^{-2} (1)	6.51×10^{-6}	9.07×10^{-6}
e	.75	.75
h_{ATM}, ft (2)	32500	32500
I_x, slug-ft^2	273000	24300
I_y, slug-ft^2	215000	22000
I_z, slug-ft^2	447000	41000
J_{xz}, slug-ft^2	0	0

Parameter	Value	
	Buffalo	Twin Otter
S, ft^2	945	420
T_{static}, lbs (1)	22400	5750
W, lbs	40000	12000
$\alpha_{B_{OL}}$, rad (3)	-.085	-.092
ρ_o, slugs/ft^3	.002378	.002378

Notes 1. From Appendix, this report.

2. Atmospheric density ratio calculated as $\sigma = e^{-h/32500}$ compares with standard atmosphere data as follows:

h	σ	
	standard	calculated
0	1	1
5000	.862	.858
10000	.738	.735
15000	.629	.630
20000	.533	.540

3. from Section IIA, this report

IV Simulation Program

The equations of Section II have been programmed for real-time solution on an XDS9300 digital computer at the TSC Simulation Facility.

Because the simulation is a simple one, a flow chart is not presented. The program listing, together with the discussion presented here, should be sufficient to completely describe the simulation. The listing is included in this report as Table I.

IV-A Interface with GAT-1 Cockpit

Provisions are made to drive the simulation manually using a GAT-1 fixed-base cockpit modified for the purpose. Commands from the cockpit are:

Elevator trim (ELTRM)
Longitudinal stick displacement (DLE)
Lateral stick displacement (DLA)
Rudder pedal diaplacement (DLR)
Throttle setting (THROT)

The scaling voltages used are given in Table I.

Similarly, the display quantities presented at the GAT-1 panel (listed in Section II-E) are scaled as shown in Table I.

IV-B Definition of Initial Values of Variables

It is convenient to be able to begin a simulation run with the aircraft trimmed at a level flight condition. Accordingly, provisions are made in the simulation for inputting desired initial conditions, and then for calculating required initial values of other parameters to produce a trimmed flight condition.

Non-zero initial values are normally input for altitude $h(0)$ and airspeed $V_R(0)$. In addition, non-zero steady state wind values can also be specified. Zero initial values are set in the first computer iteration for these parameters:

$$\dot{U},\ \dot{V},\ \dot{W},\ \dot{P},\ \dot{Q},\ \dot{R},\ \dot{\psi},\ \dot{\Theta},\ \dot{\Phi},\ P,\ Q,\ R,$$

$$\Psi,\ \theta,\ \Phi,\ V_W,\ W_W,\ x_L,\ y_L,\ \alpha,\ \dot{\alpha},\ \beta$$

An initial computation is made to calculate initial values of other parameters, using the following equations:

$$\sigma = e^{-h/h_{ATM}}$$

$$\rho = \sigma\rho_o$$

$$q = \frac{1}{2}\rho V_R^{2}$$

$$U_W = V_R$$

$$\dot{x}_L = U = V_R + \dot{x}_{w_L}$$

$$\dot{y}_L = V = \dot{y}_{w_L}$$

$$-\dot{h} = W = \dot{z}_{w_L}$$

$$C_L = C_{L_o} = W/qS$$

$$C_D = C_{D_f} + C_L^{2}/\pi eAR$$

$$D = C_D\, qS$$

$$\Theta_B = \alpha_{B_o} = C_{L_o}/a + \alpha_{B_{OL}}$$

$$\delta_e = 0$$

$$\xi = D(1 + C_{T_1} V_R + C_{T_2} V_R^{2})/\sigma \cdot T_{static}$$

The last two equations define required pilot inputs for initial trim. In the simulation, provision is made for inputting these trim values for a specified length of time, after which the actual control signal from the cockpit is used. The magnitude of the delays are TMTHR seconds for throttle setting ξ, and TMDLE seconds for elevator input δ_e. This scheme permits setting up an inital trimmed condition without the need for cockpit control manipulation. It is useful when, for example, step response runs are to be made.

V Simulation Results

Simulation results are presented in this section. These results are in the form of time responses to various step control inputs.

The time responses are presented in a manner that permits direct comparison with the linearized results generated in Appendix D of Reference 1. In general, agreement between the two sets of responses is very close.

It should be noted, however, that Reference 1 and this report utilize the same analytically-derived data. Therefore agreement between these two reports does not in itself prove the validity of either set of results. This proof can only be obtained by comparing the present results with data obtained from some other independent source. Unfortunately, however, specific data on "Buffalo" and "Twin Otter" responses from other sources are not currently available.

Accordingly, it is possible to say at this time only that this report is consistent with Reference 1 and that both sets of results are "reasonable". The time constants, frequencies, and damping ratios of the various modes presented in Appendix D of Reference 1 agree with results presented in this report. The values of these parameters are in the expected ranges, and show the normal variation with airspeed for each aircraft. Similarly, control power values appear to be within the expected ranges and in proper proportions.

Responses shown in this report are for the Cruise Flight Condition. For the "Buffalo" this is level flight at 400 fps and

10,000 ft altitude with a gross weight of 40,000 lbs. For the "Twin Otter", cruise is defined as level flight at 278 fps and 10,000 feet with a gross weight of 12,000 lbs.

Figure 5 shows the response in pitch rate Q, pitch angle Θ, angle of attack α, altitude rate $\dot{h}$, and forward speed U resulting from a 1° step elevator input δ_e for the "Buffalo". Lateral degrees of freedom were suppressed during this run. This figure compares with Figure D1 of Reference 1.

Figure 6 shows the same information for the "Twin Otter". This figure corresponds to Figure D13 of Reference 1.

Figures 7 and 8 present lateral responses for the "Buffalo". Here, longitudinal modes are suppressed. Figure 7 shows the response in sideslip angle β, roll rate P, roll angle Φ, yaw rate R, and yaw angle Ψ resulting from a 1° step aileron input δ_a. Figure 7 compares with Figure D7 of Reference 1.

Figure 8 shows the response in the same parameters resulting from a 1° step rudder input δ_r. This figure corresponds to Figure D8 of Reference 1.

Figures 9 and 10 present lateral responses for the "Twin Otter" for 1° aileron and rudder inputs, respectively. These figures correspond to Figures D19 and D20 of Reference 1.

References

1. O'Grady, J. W.; MacDonald, R. A.; and Garelick, M.: "Linearized Mathematical Models for DeHavilland Canada Buffalo and Twin Otter STOL Transports", Report No. DOT-TSC-FAA-71-8, Transportation Systems Center, Cambridge, Mass., 02142, June, 1971.

2. Perkins, C. E. and Hage, R. E., "Airplane Performance, Stability and Control", John Wiley & Sons, Inc., New York, 1963.

3. Jane's "All the World's Aircraft", 1967 Edition.

TABLE I

SIMULATION PROGRAM LISTING

```
      BLOCK DATA
C*****REVISED DATA FOR BUFFALOTER  MG 6/16/71
      REAL IX,IY,IZ
      COMMON/CONST/WEIGHT,RHOSEA,HATM,     A,B,C,S,EPIAR1,CT1,CT2,CMT,
     1 CMALF,CMDALF,CMQ,CMDLE,CYB,CYR,CYP,CNB,CNDLR,CNRFIN,DELCD,
     2 CNPFIN,CLB,CLRFIN,CLP,CLDLA,TSTAT,IX,IY,IZ,OOM ,CDF,NVEH,NTYPE,
     3  ALFBOL
      DATA NVEH/21/
      DATA WEIGHT,RHOSEA,HATM,ALFBOL/40000.0,.002378,32500.0,-.085/
      DATA A,B,C,S,EPIAR1/5.2,96.0,10.1,945.0,.0435/
      DATA CMT,CMALF,CMDALF,CMQ,CMDLE/0.0,-.78,-6.05,-35.6,2.12/
      DATA CYB,CYR,CYP,CNB,CNDLR,CLB/-.362,.368,-.055,.101,.107,-.125/
      DATA CNRFIN,CNPFIN,CLRFIN,CLP,CLDLA/-.169,.085,.038,-.53,.20/
      DATA CT1,CT2,DELCD,TSTAT,CDF/.0037,6.51E-6,.03,22400.,.032/
      DATA IX,IY,IZ/273000.,215000.,447000./
      END
      BLOCK DATA
C*****REVISED DATA FOR TWIN OTTER  MG 6/16/71
      REAL IX,IY,IZ
      COMMON/CONST/WEIGHT,RHOSEA,HATM,     A,B,C,S,EPIAR1,CT1,CT2,CMT,
     1 CMALF,CMDALF,CMQ,CMDLE,CYB,CYR,CYP,CNB,CNDLR,CNRFIN,DELCD,
     2 CNPFIN,CLB,CLRFIN,CLP,CLDLA,TSTAT,IX,IY,IZ,OOM ,CDF,NVEH,NTYPE,
     3  ALFBOL
      DATA NVEH/20/
      DATA WEIGHT,RHOSEA,HATM,ALFBOL/12000.,.002378,32500.0,-.092/
      DATA A,B,C,S,EPIAR1/5.2,65.0,6.5,420.0,.0425/
      DATA CMT,CMALF,CMDALF,CMQ,CMDLE/0.0,-.78,-6.15,-24.6,1.73/
      DATA CYB,CYR,CYP,CNB,CNDLR,CLB/-.492,.429,-.085,.121,.107,-.103/
      DATA CNRFIN,CNPFIN,CLRFIN,CLP,CLDLA/-.168,.033,.033,-.53,.38/
      DATA CT1,CT2,DELCD,TSTAT,CDF/.003774,9.07E-6,.035,5750.,.039/
      DATA IX,IY,IZ/24300.,22000.,41000./
      END
C     MAIN PROGRAM
      DIMENSION DERIV(12),VINT(12)
      REAL LIFT,LV,LR,LP,LDLA,NV,NR,NP,NDLR,IX,IY,IZ
X     COMMON/BLIPRN/BLPFRQ,ITTB,IBLIP,PRNFRQ,ITTP,IPRN
      COMMON/CONST/WEIGHT,RHOSEA,HATM,     A,B,C,S,EPIAR1,CT1,CT2,CMT,
     1 CMALF,CMDALF,CMQ,CMDLE,CYB,CYR,CYP,CNB,CNDLR,CNRFIN,DELCD,
     2 CNPFIN,CLB,CLRFIN,CLP,CLDLA,TSTAT,IX,IY,IZ,OOM ,CDF,NVEH,NTYPE,
     3  ALFBOL
      COMMON        XSS,YSS,ZSS,RNAX,RNBX,RNAY,RNBY,RNAZ,RNBZ,WXX,WYY,
     1  WZZ,XSD,YSD,ZSD,XTAU,YTAU,ZTAU,JX,JY,JZ,SGUST,UW,VW,WW
      COMMON        TMTHR,TMDLE,DRAGI,ALFI,VRI,SIGI
      COMMON        RHO,SIG,DYN,CL,CD,YV,YR,YP,LV,CLR,LR,LP,LDLA,NV,CNR,
     1 NR,CNP,NP,NDLR,CM,DEL,T
      COMMON        ELTRM,DLE,DLA,DLR,THROT,GDLE,GDLA,GDLR,GELTRM,
     1  TRNPNT,SLPBL,PTCHBR
      COMMON      ALF,VR,LIFT,DRAG,DALF,THRUST,UDOT,WDOT,QDOT,VDOT,PDOT,
     1 RDOT,THETDOT,PSIDOT,PHIDOT,XDOT,YDOT,HDOT,U,W,Q,V,P,R,THETA,PSI,
     2 PHI,X,Y,H,BETA
      COMMON CLO,ALFBO
      EQUIVALENCE (DERIV(1),UDOT),(DERIV(2),WDOT),(DERIV(3),QDOT),
     1 (DERIV(4),VDOT),(DERIV(5),PDOT),(DERIV(6),RDOT),(DERIV(7),THETDOT
     2),(DERIV(8),PSIDOT),(DERIV(9),PHIDOT),(DERIV(10),XDOT),(DERIV(11),
     3YDOT),(DERIV(12),HDOT),(VINT(1),U),(VINT(2),W),(VINT(3),Q),(VINT(4
     4),V),(VINT(5),P),(VINT(6),R),(VINT(7),THETA),(VINT(8),PSI),(VINT(9
     5),PHI),(VINT(10),X),(VINT(11),Y),(VINT(12),H)
      NAMELIST H,VR,X,Y,PSI,PHI,P,Q,R,V,DEL,NTYPE,WEIGHT
X     NAMELIST ELTRM,DLE,DLA,DLR,THROT
      NAMELIST TMTHR,TMDLE
```

TABLE I (Cont)

```
X       NAMELIST BLPFRQ,PRNFRQ
        NAMELIST GDLE,GDLA,GDLR,TRNPNT,SLPBL,PTCHBR,GELTRM
        NAMELIST XSS,YSS,ZSS,XSD,YSD,ZSD,XTAU,YTAU,ZTAU,JX,JY,JZ
C
C
        CALL SETPOT(32,3000,2000,0000,2000)
        CALL SETPOT(132,3000,2000,0000,2000)
C
   12   NTYPE=0
C***** CALL STANDBY
S       JM  031010
C*****   3 DEG/SEC/POINTER WIDTH FOR RATE OF TURN
        TRNPNT=19.1
C*****  10 DEG/BALL-WIDTH FOR SLIP
        SLPBL=5.73
C*****   4  DEG/BAR-WIDTH FOR PITCH
        PTCHBR=14.325
C*****  2 DEG/VOLT
        GELTRM=GDLE=.0349066
C*****  2/3 DEG/VOLT
        GDLR=.0116356
C*****  4/3 DEG/VOLT
        GDLA=-.0232712
C
C       SET INITIAL CONDITIONS,  IDLE LOOP
C
        H=6000.
        VR=200.
        Y=PSI=PHI=P=Q=R=  TMTHR=TMDLE=ZSS=0.
X       BLPFRQ=10.
X       PRNFRQ=1.
        JX=JY=JZ=1
        XSS=14.
        YSS=-14.
        XSD=2.3
        YSD=1.6
        ZSD=1.
        XTAU=YTAU=ZTAU=1.5
        DEL=.05
        CALL IFINITIA
        INPUT(105)
C***** CALL COMPUTE
S       EOM  031013
X       TEMP=BLPFRQ/DEL
X       ITTB=TEMP
X       TEMP=PRNFRQ/DEL
X       ITTP=TEMP
X       IBLIP=IPRN=-1
        OOM=32.2/WEIGHT
        DO 1 I=1,9
    1   DERIV(I)=0.
        SIG=EXP(-H/HATM)
        RHO=SIG*RHOSEA
        CLO=2.*WEIGHT/(RHO*S*VR*VR)
        ALFRO=CLO/A+ALFBOL
        THETA=ALF=0.
        IF(SENSESWITCH5)20,21
   20   CONTINUE
        U=VR+XSS
        V=YSS
        W=ZSS
```

TABLE I (Cont)

```
      GO TO 22
   21 CONTINUE
      W=V=0.
      U=VR
   22 CONTINUE
      VW=WW=0.
      UW=VR
      CD=CDF+CL*CL*EPIAR1
      DYN=.5*RHO*VR*VR
      DRAG=CD*DYN*S
      DRAGI=DRAG
      ALFI=ALF
      VRI=VR
      SIGI=SIG
      T=-DEL
      RNAX=1.-DEL/XTAU
      RNAY=1.-DEL/YTAU
      RNAZ=1.-DEL/ZTAU
      RNBX=XSD*SQRT(2.*DEL/XTAU)
      RNBY=YSD*SQRT(2.*DEL/YTAU)
      RNBZ=ZSD*SQRT(2.*DEL/ZTAU)
      SGUST=SQRT(XSS*XSS+YSS*YSS)
      CALL ARM(0)
      CALL ENINT
      CONNECT(40,AERO)
C*****IDLE LOOP + TEST-CASE CALLS OF AERO
   10 CONTINUE
X 11  CALL AERO
C***** IDLE LOOP + FAKE INT.OO USING F/F,T.L.,S.L. IF INT. SYSTEM DOWN
S     SKS  030004
S     BRU  $-1
S     EOM  030004
S     EOM  030200
      IF(SENSESWITCH1)12,10
      END
      SUBROUTINE AERO
      DIMENSION BTE(3,3)
      DIMENSION DERIV(12),VINT(12)
      REAL LIFT,LV,LR,LP,LDLA,NV,NR,NP,NDLR,IX,IY,IZ
X     COMMON/PLIPRN/BLPFRQ,ITTB,IBLIP,PRNFRQ,ITTP,IPRN
      COMMON/CONST/WEIGHT,RHOSEA,HATM,     A,B,C,S,EPIAR1,CT1,CT2,CMT,
     1 CMALF,CMDALF,CMQ,CMDLE,CYB,CYR,CYP,CNB,CNDLR,CNRFIN,DELCD,
     2 CNPFIN,CLB,CLRFIN,CLP,CLDLA,TSTAT,IX,IY,IZ,OOM ,CDF,NVEH,NTYPE,
     3  ALFBOL
      COMMON        XSS,YSS,ZSS,RNAX,RNBX,RNAY,RNBY,RNAZ,RNBZ,WXX,WYY,
     1  WZZ,XSD,YSD,ZSD,XTAU,YTAU,ZTAU,JX,JY,JZ,SGUST,UW,VW,WW
      COMMON        THTHR,TMDLE,DRAGI,ALFI,VRI,SIGI
      COMMON        RHO,SIG,DYN,CL,CD,YV,YR,YP,LV,CLR,LR,LP,LDLA,NV,CNR,
     1 NR,CNP,NP,NDLR,CM,DEL,T
      COMMON        FLTRM,DLE,DLA,DLR,THROT,GDLE,GDLA,GDLR,GELTRM,
     1  TRNPNT,SLPBL,PTCHBR
      COMMON       ALF,VR,LIFT,DRAG,DALF,THRUST,UDOT,WDOT,QDOT,VDOT,PDOT,
     1 RDOT,THETDOT,PSIDOT,PHIDOT,XDOT,YDOT,HDOT,U,W,Q,V,P,R,THETA,PSI,
     2 PHI,X,Y,H,BETA
      COMMON CLO,ALFBO
      EQUIVALENCE (DERIV(1),UDOT),(DERIV(2),WDOT),(DERIV(3),QDOT),
     1 (DERIV(4),VDOT),(DERIV(5),PDOT),(DERIV(6),RDOT),(DERIV(7),THETDOT
     2),(DERIV(8),PSIDOT),(DERIV(9),PHIDOT),(DERIV(10),XDOT),(DERIV(11),
     3YDOT),(DERIV(12),HDOT),(VINT(1),U),(VINT(2),W),(VINT(3),Q),(VINT(4
     4),V),(VINT(5),P),(VINT(6),R),(VINT(7),THETA),(VINT(8),PSI),(VINT(9
     5),PHI),(VINT(10),X),(VINT(11),Y),(VINT(12),H)
```

TABLE I (Cont)

```
C
C***** TIMING SIGNAL, SET F/F
S      EOM  030000
C      RECTANGULAR INTEGRATION
       T=T+DEL
       DO 10 I=1,12
   10  VINT(I)=VINT(I)+DERIV(I)*DEL
C      A/D HERE
C*****  DLE FROM -10 V. DOWN TO +15 V. UP
C*****  DLA FROM -15 V. RIGHT TO +15 V. LEFT
C* ***  DLR FROM -30 V. RIGHT TO +30 V. LEFT
C*****  ELTRM FROM -15 V. DOWN TO +15 V. UP
C*****  THROT FROM -3.2 V. IDLE TO 0 V. FULL
       CALL ADLT20,ELTRM,DLE,DLA,DLR,THROT)
       ELTRM=GELTRM*ELTRM
       DLA=GDLA*DLA
       DLR=GDLR*DLR
       DLE=GDLE*DLE+ELTRM
       THROT=1.+.3125*THROT
C      TOTAL VELOCITY
       VRSQ=UW*UW+VW*VW+WW*WW
       VR=SQRT(VRSQ)
C         CALCULATE COEFFICIENTS
       SIG=EXP(-H/HATM)
       RHO=RHOSEA*SIG
       IF(T.GE.TMTHR)GO TO 81
       IF(NVEH.EQ.2)SIGI=1
       THROT=DRAGI*(1.+CT1*VRI+CT2*VRI*VRI)/(SIGI*TSTAT)
   81  IF(T.GE.TMDLE)GO TO 82
       DLE=0.
   82  CONTINUE
C  CONVAIR IS SUPERCHARGED***USE SIG=1 FOR THRUST COMP
       IF(NVEH.EQ.2)SIG=1
       THRUST=THROT*SIG*TSTAT/(1.+CT1*VR+CT2*VRSQ)
C     DYNAMIC PRESSURE
       DYN=.5*RHO*VRSQ
C
       SPHI=SIN(PHI)
       SPSI=SIN(PSI)
       STH=SIN(THETA)
       CPHI=COS(PHI)
       CPSI=COS(PSI)
       CTH=COS(THETA)
C*****BODY-TO EARTH TRANS MATRIX
       STCS=STH*CPSI
       SSCP=SPSI*CPHI
       SSSP=SPSI*SPHI
       BTE(1,1)=CTH*CPSI
       BTE(1,2)=STCS*SPHI-SSCP
       BTE(1,3)=STCS*CPHI+SSSP
       BTE(2,1)=CTH*SPSI
       BTE(2,2)=SSSP*STH+CPHI*CPSI
       BTE(2,3)=SSCP*STH-SPHI*CPSI
       BTE(3,1)=STH
       BTE(3,2)=-SPHI*CTH
       BTE(3,3)=-CPHI*CTH
C     NEW WIND MODEL
C   F.S. 21,22,23 DOWN FOR GUST IN X,Y,Z RESP.
C      S.S. 5 SET FOR STEADY STATE
C
S      SKS  030006
```

TABLE I (Cont)

```
S       BRU  11S
        CALL GUST(XGUS,RNAX,RNBX,JX)
        JX=2
S       BRU  12S
  11    XGUS=0.
S12     SKS  030007
S       BRU  13S
        CALL GUST(YGUS,RNAY,RNBY,JY)
        JY=2
S       BRU  14S
  13    YGUS=0.
S14     SKS  030010
S       BRU  15S
        CALL GUST(ZGUS,RNAZ,RNBZ,JZ)
        JZ=2
        GO TO 3
   15   ZGUS=0.
   3    IF(SENSESWITCH5)4,5
    4   CONTINUE
        WXX=XSS+(XGUS*XSS-YGUS*YSS)/SGUST
        WYY=YSS+(XGUS*YSS+YGUS*XSS)/SGUST
        WZZ=ZSS+ZGUS
        GO TO 6
   5    WXX=XGUS
        WYY=YGUS
        WZZ=ZGUS
    6   CONTINUE
C***** BODY AXIS VELOCITIES INCLUDING WINDS
        UW=U-(WXX*BTE(1,1)+WYY*BTE(2,1)-WZZ*BTE(3,1))
        VW=V-(WXX*BTE(1,2)+WYY*BTE(2,2)-WZZ*BTE(3,2))
        WW=W-(WXX*BTE(1,3)+WYY*BTE(2,3)-WZZ*BTE(3,3))
C      ANGLE OF ATTACK,LIFT,DRAG
        ALF=ATAN2(WW,UW)
        CL=A*ALF+CLO
        CD=CDF+CL*CL*EPIAR1
        QS=DYN*S
        LIFT=CL*QS
        DRAG=CD*QS
C        SIDESLIP
        BETA=ATAN2(VW,UW)
C  RVS=RVS/2   RVSB=RVSB/2   RV4=RVSB/4   RV4B2=RVSBB/4
        RVS=.5*RHO*VR*S
        YV=RVS*CYB
        RVSB=RVS*B
        RV4=.5*RVSB
        YR=RV4*CYR
        YP=RV4*CYP
        LV=RVSB*CLB
        CLR=CLRFIN+.25*CL
        RV4B2=RV4*B
        LR=RV4B2*CLR
        LP=RV4B2*CLP
        LDLA=QS*B*CLDLA
        NV=RVSB*CNB
        CNR=CNRFIN-.25*(CD-DELCD)
        NR=RV4B2*CNR
        CNP=CNPFIN-.25*CL*(1.-A*EPIAR1)
        NP=RV4B2*CNP
        NDLR=QS*B*CNDLR
C
C         CALCULATE AERODYNAMIC FORCES
```

TABLE I (Cont)

```
      CALF=COS(ALF)
      SALF=SIN(ALF)
C*****LONGITUDINAL EQUATIONS
C*****IF NTYPE=2, FLY ONLY LATERAL
      IF(NTYPE.NE.2)GO TO 31
      UDOT=WDOT=QDOT=0.
      GO TO 32
C     X FORCE
   31 UDOT=OOM*(THRUST-DRAG*CALF+LIFT*SALF-WEIGHT*STH)-Q*W+R*V
C     Z FORCE
      WDOT=OOM*(-LIFT*CALF-DRAG*SALF+WEIGHT*CTH*CPHI)+Q*U-P*V
C     ALPHA DOT
      DALF=(WDOT-WW*UDOT/UW)*CALF*CALF/UW
C     M MOMENT
      CM=CMT+CMALF*ALF+C*.5*(CMDALF*DALF+CMQ*Q)/VR+CMDLE*DLE
      QDOT=CM*QS*C/IY
C
C*****LATERAL EQUATIONS
C*****IF NTYPE=1, FLY ONLY LONGITUDINAL
      IF(NTYPE.NE.1)GO TO 32
      VDOT=PDOT=RDOT=0.
      GO TO 33
C     Y FORCE
   32 VDOT=OOM*(YV*VW+YR*R+YP*P+WEIGHT*SPHI)-R*U+P*W
C     L MOMENT
      PDOT=(LV*VW+LR*R+LP*P+LDLA*DLA)/IX
C     N MOMENT
      RDOT=(NV*VW+NR*R+NP*P+NDLR*DLR)/IZ
C
C     EULER ANGLE TRANSFORMATION
   33 THETDOT=Q*CPHI-R*SPHI
      PSIDOT=(Q*SPHI+R*CPHI)/CTH
      PHIDOT=P+PSIDOT*STH
C
C
C***** XDOT,YDOT,HDOT IN EARTH-FIXED COORDINATES
      DO 35 I=1,3
   35 DERIV(I+9)=BTE(I,1)*U+BTE(I,2)*V+BTE(I,3)*W
C
      IF(NTYPE.NE.2)GO TO 34
      HDOT=THETDOT=0.
      PSIDOT=R
      PHIDOT=P
   34 CONTINUE
C
      IF(SENSESWITCH2)71,72
   71 D8=THROT*10.
C
C*****BLIPS FOR TIME ON STRIP CHART RECORDER
X     IBLIP=IBLIP+1
X     IPRN=IPRN+1
X     IF(IBLIP.EQ.ITTB)GO TO 63
X     D8=-25.
X     GO TO 64
X 63  IBLIP=0
X     D8=25.
X 64  CONTINUE
      D1=H*.005
      D2=VR*.1
      D3=THETA*143.25
      D4=DLE*143.25
```

```
      D5=PHI*57.3
      D6=R*286.5
      D7=DLA*114.6
C
C*****  NTYPE= 0 FOR COUPLED, 1 FOR LONGITUDINAL ONLY, 2 FOR LATERAL ONLY
C
      IF(NTYPE-1)66,70,65
   70 D5=HDOT
      D6=Q*57.3
      D7=ALF*143.25
      GO TO 66
   65 D1=P*57.3
      D2=BETA*286.5
      D3=DLR*57.3
      D4=PSI*28.65
   66 CONTINUE
      CALL DAL(20,D1,D2,D3,D4,D5,D6,D7,D8)
   72 CONTINUE
      TURN=-TRNPNT*7.5*PSIDOT
      SLIP=-15.*SLPBL*BETA
C*****      HDOT IN FT/SEC,  INDICATOR IN FT/MIN
C*****   -6.67 VOLTS/1000 CLIMB,  +8.33 VOLTS/1000 DESCEND
C
      IF(HDOT.GT.0.)ROFC= .4002*HDOT
      IF(HDOT.LE.0.)ROFC= .4998*HDOT
C        VR IN FT/SEC,     CONVERT TO KNOTS
      VKTS=VR*.592*SQRT(RHO/RHOSEA)
C
      IF(VKTS-175.)200,210,210
  200 IF(VKTS-125.)201,209,209
  201 IF(VKTS-44.)202,208,208
  202 AIRSPD=0.
      GO TO 211
  208 AIRSPD= .22*(VKTS-44.)
      GO TO 211
  209 AIRSPD= 17.8+.092*(VKTS-125.)
      GO TO 211
  210 AIRSPD= 22.4+.0296*(VKTS-175.)
  211 CONTINUE
        ALT=.02*H-80.
      DIRGYR=-PSI*15.916667
      I=DIRGYR*.01
      DIRGYR=DIRGYR-I*100.
      ROLL=95.5*PHI
      IF(ABS(ROLL).GT.99.9)ROLL=SIGN(99.9,ROLL)
      PITCH=5.*PTCHBR*(THETA+ALFBO)
      RPM=3.+THROT*27.
C     D/A HERE
      CALL DAL(28,
     1               TURN,SLIP,ROFC,AIRSPD,ALT,DIRGYR,ROLL,PITCH,RPM)
C*****PRINT-OUT AT FREQUENCY PRNFRQ
X     IF(IPRN.NE.ITYP)GO TO 62
X     IPRN=0
X     IF(SENSESWITCH3)61,62
X 61  WRITE(108,101)T,UDOT,U,V,W,VR,H,HDOT,ALF*57.3,THETA*57.3,PHI*57.3,
X    1 PSI*57.3,P*57.3,Q*57.3,R*57.3,THRUST,THROT,LIFT,DRAG,DLE*57.3,
X    2 DLA*57.3,DLR*57.3,BETA*57.3
X 101 FORMAT($ T=$,F8.2/3X,$UDOT=$,F11.4,12X,$U=$,F11.4,12X,$V=$,F11.4,
X    1 12X,$W=$,F11.4,11X,$VR=$,F11.4/6X,$H=$,F11.4,9X,$HDOT=$,F11.4,
X    2 10X,$ALF=$,F11.4,8X,$THETA=$,F11.4,10X,$PHI=$,F11.4,/4X,$PSI=$,
X    3 F11.4,12X,$P=$,F11.4,12X,$Q=$,F11.4,12X,$R=$,F11.4,7X,$THRUST=$,
```

TABLE I (Cont)

```
X     4 F11.4,/2X,$THROT=$,F11.2,11X$LIFT=$,F11.4,9X,$DRAG=$,F11.4,10X,
X     5 $DLE=$,F11.2,12X,$DLA=$,F11.2,74X,$DLR=$,F11.2,11X,$BETA=$,
X     6 F11.4///)
X 62  CONTINUE
C
C***** TIMING SIGNAL, RESET F/F
S     EOM  030001
      RETURN
      END
```

DHC-6 TWIN OTTER

Announced in August 1964, the Twin Otter is a STOL transport powered by two Pratt & Whitney (UAC) PT6A-20 turboprop engines. Design work began in January 1964. Construction of an initial batch of Twin Otters was started in November of the same year and the first of these flew on May 20, 1965.

At the beginning of 1967, a total of 52 Twin Otters had been delivered or were on order, with options on 11 more. They included eight for the Chilean Air Force, two for Trans-Australian Airlines, one for the Canadian Department of Lands and Forests, four for Aeralpi of Italy, one for Northern Consolidated Airlines, and others for Pilgrim Airlines and Air Wisconsin, USA. Production was scheduled to be at the rate of six a month through 1967.

Under development for delivery in 1968 is a version of the Twin Otter with more powerful (640 eshp) Pratt & Whitney PT6A-27 turboprop engines, longer nose to provide more baggage space, and AUW of 12,500 lb (5,670 kg). The following data refer to the current production model.

TYPE: Twin-turboprop STOL transport.

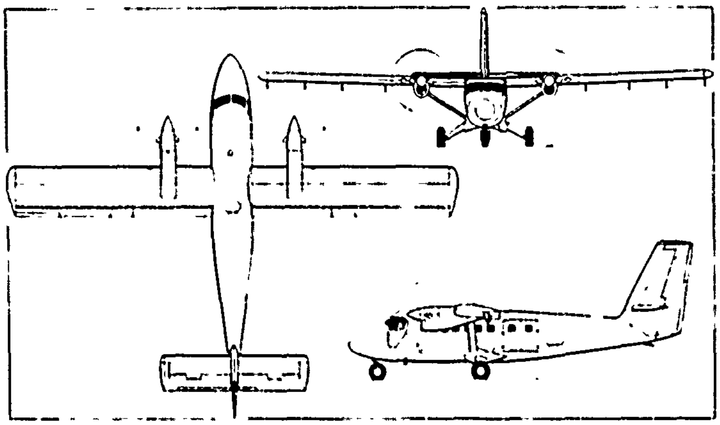
WINGS: Braced high-wing monoplane, with a single streamline-section bracing strut on each side. Wing section NACA 6A series mean line; NACA 0016 (modified) thickness distribution. Aspect ratio 10. Constant chord of 6 ft 6 in (1·98 m). Dihedral 2°. Incidence 2° 30'. No sweepback. All-metal safe-life structure. All-metal ailerons which also droop for use as flaps. Double slotted all-metal full span trailing-edge flaps. No spoilers. Trim-tabs in ailerons. Pneumatic-boot de-icing equipment optional.

FUSELAGE. Conventional all-metal semi monocoque safe-life structure.

TAIL UNIT: Cantilever all-metal structure of high strength aluminium alloys. Fin integral with fuselage. Fixed-incidence tailplane. Trim-tabs in rudder and port elevator, latter interconnected with flaps. Pneumatic de-icing boots on tailplane leading edge optional.

LANDING GEAR: Non-retractable tricycle type, with steerable nose-wheel. Rubber shock absorption on main units. Oleo pneumatic nose-wheel shock-absorber. Goodyear main wheel tyres size 11·00 × 12, pressure 32 lb/sq in (2 25 kg/cm²) Goodyear nose-wheel tyre size 8 90 × 12·50, pressure 31 lb/sq in (2·18 kg/cm²). Goodrich hydraulic brakes. Provision for alternative float and ski gear.

POWER PLANT Two 579 eshp Pratt & Whitney (UAC) PT6A-20 turboprop engines, each driving a Hartzell three-blade reversible-pitch fully-feathering metal propeller, diameter 8 ft 0 in (2 44 m) Fuel in two tanks (8 cells) under cabin floor; total capacity 919 Imp gallons (4,178 litres) Two refuelling points on port side of fuselage. Oil capacity 2 Imp gallons (9 litres) per engine. Electric de-icing system for propellers and air-intakes optional.



de Havilland Canada DHC-6 Twin Otter twin-turboprop transport

ACCOMMODATION: Two seats side-by-side on flight deck. Seats for 13-18 passengers in main cabin. Cabin divided by bulkhead into main passenger or freight compartment and baggage or toilet compartment. Door on each side of main cabin, at rear. Baggage compartments in nose and aft of cabin, each with upward-hinged door on port side.

SYSTEMS: Hydraulic system, pressure 1,500 lb/sq in (105 kg/cm²), for flaps, brakes and nose-wheel steering. No pneumatic system. One 200A starter-generator on each engine.

ELECTRONICS AND EQUIPMENT: Radio and radar to customer's specification. Blind-flying instrumentation standard.

DIMENSIONS, EXTERNAL:

Wing span	65 ft 0 in (19 81 m)
Length overall	49 ft 6 in (15 09 m)
Height overall	18 ft 7 in (5·66 m)
Tailplane span	21 ft 0 in (6 40 m)
Wheel track	12 ft 5 in (3·78 m)
Wheelbase	14 ft 9 in (4·50 m)
Passenger door (port side).	
Height	4 ft 2 in (1·27 m)
Width	2 ft 6 in (0 76 m)
Height to sill	3 ft 10 in (1·17 m)
Passenger door (starboard side):	
Height	3 ft 9½ in (1·15 m)
Width	2 ft 6 in (0 76 m)
Height to sill	3 ft 10 in (1·17 m)
Baggage compartment door (nose):	
Height to sill	3 ft 10 in (1·17 m)
Baggage compartment door (port, rear):	
Height	4 ft 2 in (1·27 m)
Width	4 ft 8 in (1·42 m)
Height to sill	3 ft 10 in (1 17 m)

DIMENSIONS, INTERNAL:

Cabin, excluding flight deck, galley and baggage or toilet compartment.	
Length	18 ft 6 in (5 64 m)
Max width	5 ft 3 in (1·60 m)
Max height	4 ft 11 in (1·50 m)
Floor area	80 2 sq ft (7·45 m²)
Volume	384 cu ft (10 87 m³)
Baggage compartment (nose) volume	22 cu ft (0 62 m³)
Baggage compartment (rear) volume	52 cu ft (1 47 m³)

AREAS:

Wings, gross	420 sq ft (39 02 m²)
Ailerons (total)	33·2 sq ft (3 08 m²)
Trailing-edge flaps (total)	112 2 sq ft (10 42 m²)
Fin	48 0 sq ft (4·46 m²)
Rudder, including tab	34 0 sq ft (3·16 m²)
Tailplane	100 sq ft (9 29 m²)
Elevators, including tab	35 sq ft (3·25 m²)

WEIGHTS:

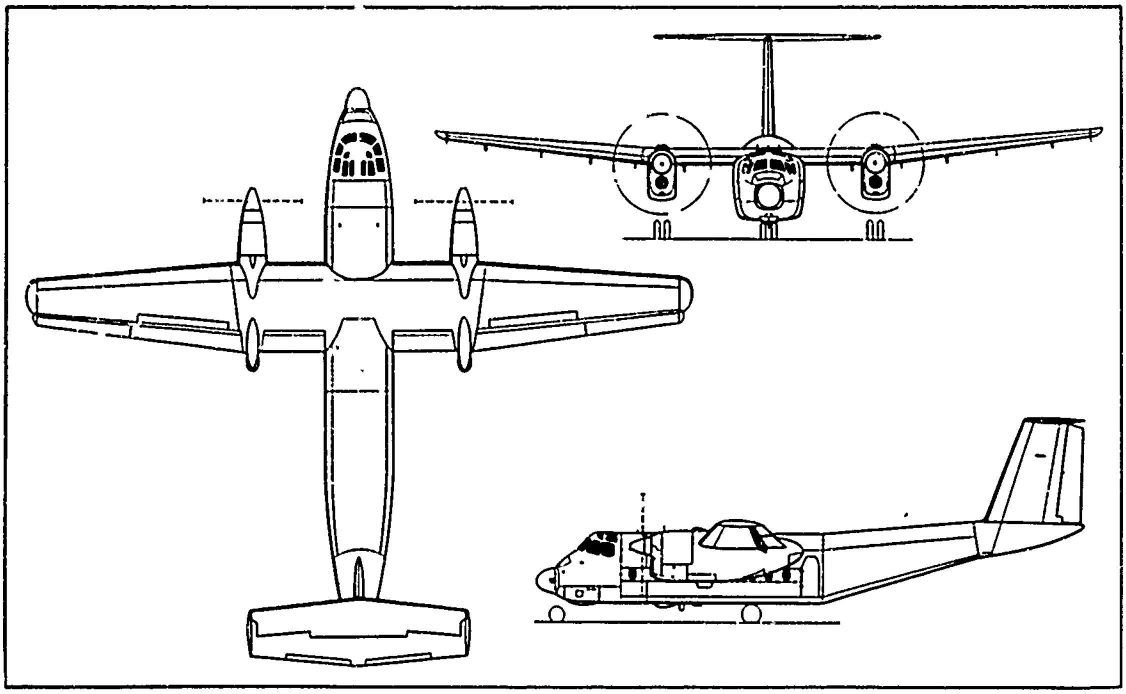
Basic operating weight, including pilot (170 lb = 77 kg), radio (100 lb = 45 kg) and full oil	6,170 lb (2,800 kg)
Max payload (for 100 mile = 160 km range)	4,430 lb (2,010 kg)
Max T-O weight	11,579 lb (5,252 kg)
Max landing weight	11,000 lb (4,990 kg)

PERFORMANCE (at max T-O weight).

Max cruising speed at 10,000 ft (3,050 m)	184 mph (297 kmh)
Econ cruising speed at 10,000 ft (3,050 m)	156 mph (251 kmh)
Landing speed	64 5 mph (104 kmh)
Rate of climb at S/L	1,550 ft (472 m) min
Service ceiling	25,500 ft (7,770 m)
Service ceiling, one engine out	8,500 ft (2,590 m)
T-O to 50 ft (15 m):	
STOL	1,120 ft (341 m)
CAR Pt 3	1,700 ft (518 m)
Landing from 50 ft (15 m):	
STOL	1,020 ft (311 m)
CAR Pt 3	2,160 ft (658 m)
Range with max fuel, 30 min reserve	920 miles (1,480 km)

Source: Reference 3

FIGURE 1



DHC-5 Buffalo twin-turboprop STOL utility transport

DHC-5 BUFFALO

Differences between the US and Canadian versions are as follows:

CV-7A. US model, with 2,850 eshp General Electric T64-GE-10 turboprops. Overall length 77 ft 4 in (23·57 m). Designation may be changed following transfer of responsibility for aircraft in this category from US Army to USAF.

CC-115. Canadian Defence Force model, with 3,055 eshp General Electric T64/P2 turboprops. Overall length 79 ft 0 in (24·08 m). Otherwise similar to CV-7A, with only small differences in performance.

WINGS: Cantilever high-wing monoplane. Wing section NACA 64,A417·5 (mod) at root, NACA 63,A615 (mod) at tip. Aspect ratio 9·75. Chord 11 ft 9½ in (3·59 m) at root, 5 ft 11 in (1·19 m) at tip. Dihedral 0° inboard of nacelles, 5° outboard. Incidence 2° 30′. Sweepback at quarter chord 1° 40′. Conventional fail-safe multi-spar structure of high-strength aluminium alloys. Full span double-slotted aluminium alloy flaps, outboard sections functioning as ailerons. Aluminium alloy slot-lip spoilers, forward of inboard flaps, are actuated by Jarry Hydraulics unit. Spoilers coupled to manually-operated ailerons for lateral control, uncoupled for symmetrical ground operation. Electrically-actuated trim-tab in starboard aileron. Geared tab in each aileron. Rudder-aileron interconnect tab on port aileron. Outer wing leading-edges fitted with electrically-controlled flush pneumatic rubber de-icer boots.

FUSELAGE: Fail-safe structure of high-strength aluminium alloy. Cargo floor supported by longitudinal keel members.

TAIL UNIT: Cantilever structure of high-strength aluminium alloy, with fixed-incidence tailplane mounted at tip of fin. Elevator aerodynamically and mass-balanced. Fore and trailing serially-hinged rudders are powered by tandem jacks operated by two independent hydraulic systems manufactured by Jarry Hydraulics. Trim-tab on port elevator, spring-tab on starboard elevator. Electrically-controlled flush pneumatic rubber de-icer boot on tailplane leading-edge.

LANDING GEAR: Retractable tricycle type. Hydraulic retraction, nose unit aft, main units forward. Jarry Hydraulics oleo-pneumatic shock-absorbers. Goodrich main wheels and tyres, size 37·00 × 15·00-12, pressure 45 lb/sq in (3·16 kg/cm²). Goodrich nose wheels and tyres size 8·90 × 12·50, pressure 38 lb/sq in (2·67 kg/cm²). Goodrich multi-disc brakes.

POWER PLANT: Two General Electric T64 turboprop engines (details under entries for individual versions, above), each driving a Hamilton Standard 63E50-13 three-blade propeller, diameter 14 ft 6 in (4·42 m). Fuel in one integral tank in each inner wing, capacity 533 Imp gallons (2,423 litres) and rubber bag tanks in each outer wing, capacity 336 Imp gallons (1,527 litres). Total fuel capacity 1,738 Imp gallons (7,900 litres). Refuelling points above wings and in side of fuselage for pressure refuelling. Total oil capacity 10 Imp gallons (45·5 litres).

DIMENSIONS, EXTERNAL:

Wing span	96 ft 0 in (29·26 m)
Length overall:	
CV-7A	77 ft 4 in (23·57 m)
CC-115	79 ft 0 in (24·08 m)
Height overall	28 ft 8 in (8·73 m)
Tailplane span	32 ft 0 in (9·75 m)
Wheel track	30 ft 6 in (9·29 m)
Wheelbase	27 ft 11 in (8·50 m)
Cabin doors (each side):	
Height	5 ft 6 in (1·68 m)
Width	2 ft 9 in (0·84 m)
Height to sill	3 ft 10 in (1·17 m)
Emergency exits (each side, below wing leading-edge):	
Height	3 ft 4 in (1·02 m)
Width	2 ft 2 in (0·66 m)
Height to sill approx	5 ft 0 in (1·52 m)
Rear cargo loading door and ramp:	
Height	20 ft 9 in (6·33 m)
Width	7 ft 8 in (2·33 m)
Height to ramp hinge	3 ft 10 in (1·17 m)

DIMENSIONS, INTERNAL:

Cabin, excluding flight deck:	
Length, cargo floor	31 ft 5 in (9·58 m)
Max width	8 ft 9 in (2·67 m)
Max height	6 ft 10 in (2·08 m)
Floor area	243·5 sq ft (22·63 m²)
Volume	1,715 cu ft (48·56 m³)

AREAS:

Wings, gross	945 sq ft (87·8 m²)
Ailerons (total)	39 sq ft (3·62 m²)
Trailing-edge flaps (total, including ailerons)	280 sq ft (26·01 m²)
Spoilers (total)	25·2 sq ft (2·34 m²)
Fin	92 sq ft (8·55 m²)
Rudder, including tab	60 sq ft (5·57 m²)
Tailplane	151·5 sq ft (14·07 m²)
Elevators, including tab	81·5 sq ft (7·57 m²)

WEIGHTS AND LOADINGS:

Operating weight empty, including 3 crew at 200 lb (91 kg) each, plus trapped fuel and oil and full cargo handling equipment	23,157 lb (10,505 kg)
Max payload	13,843 lb (6,279 kg)
Max T-O weight	41,000 lb (18,598 kg)
Max zero-fuel weight	37,000 lb (16,783 kg)
Max landing weight	39,000 lb (17,690 kg)
Max wing loading	43·4 lb/sq ft (212 kg/m²)
Max power loading	7·2 lb/eshp (3·27 kg/eshp)

PERFORMANCE (CV-7A, at max T-O weight):

Max level speed at 10,000 ft (3,050 m)	271 mph (435 kmh)
Max permissible diving speed	334 mph (537 kmh)
Max cruising speed at 10,000 ft (3,050 m)	271 mph (435 kmh)
Econ cruising speed at 10,000 ft (3,050 m)	208 mph (335 kmh)
Stalling speed, 40° flaps at 39,000 lb (17,690 kg) AUW	75 mph (120 kmh)
Stalling speed, flaps up at max AUW	105 mph (169 kmh)
Rate of climb at S/L	1,890 ft (575 m) min
Service ceiling	30,000 ft (9,150 m)
Service ceiling, one engine out	14,300 ft (4,360 m)
T-O run on firm dry sod	1,040 ft (317 m)
T-O to 50 ft (15 m) from firm dry sod	1,540 ft (470 m)
Landing from 50 ft (15 m) on firm dry sod	1,120 ft (342 m)
Landing run on firm dry sod	610 ft (186 m)

Source: Reference 3

Figure 2

L: EARTH LOCAL VERTICAL COORDINATE FRAME

C: EARTH-AIRCRAFT CONTROL COORDINATE FRAME

A: AIRCRAFT BODY COORDINATE FRAME

EULER ANGLES

Ψ = ROTATION ABOUT Z_L AXIS

Θ = ROTATION ABOUT Y_C AXIS

Φ = ROTATION ABOUT X_A AXIS

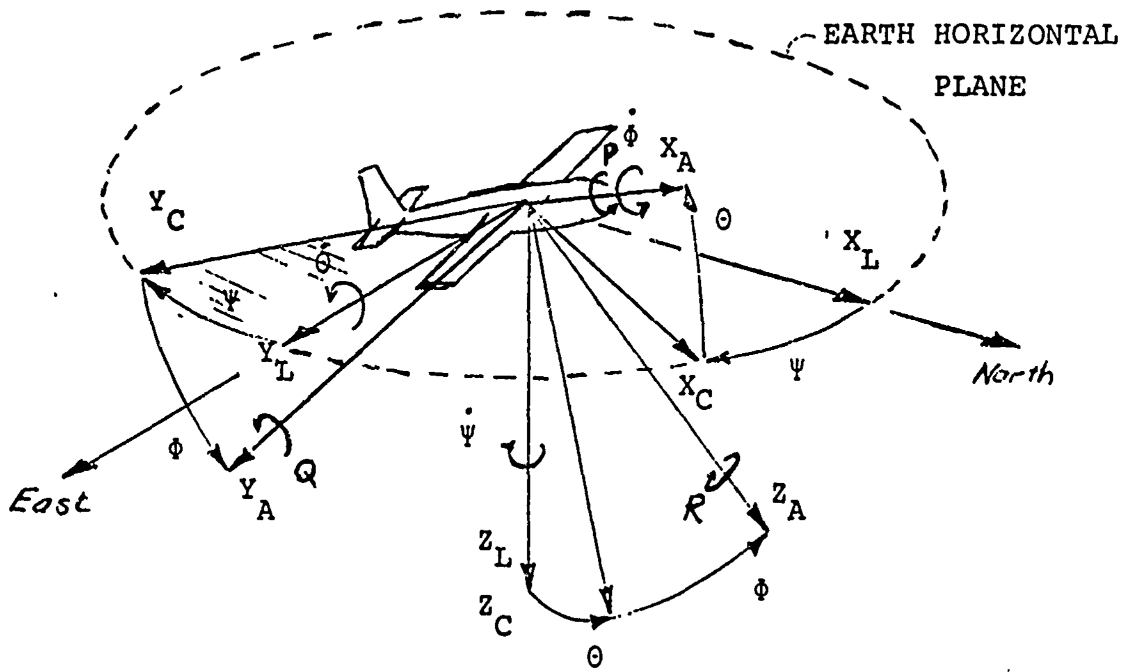
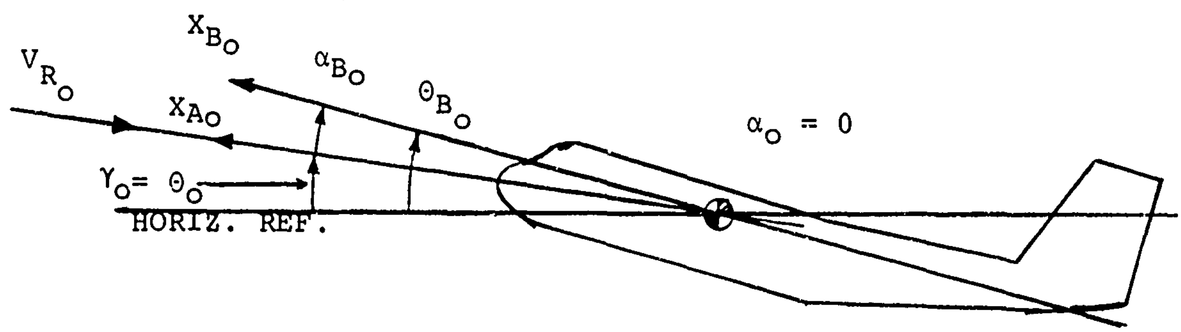


Figure 3: Reference Coordinate Frames

(a) At equilibruim



(b) Displaced from equilibrium

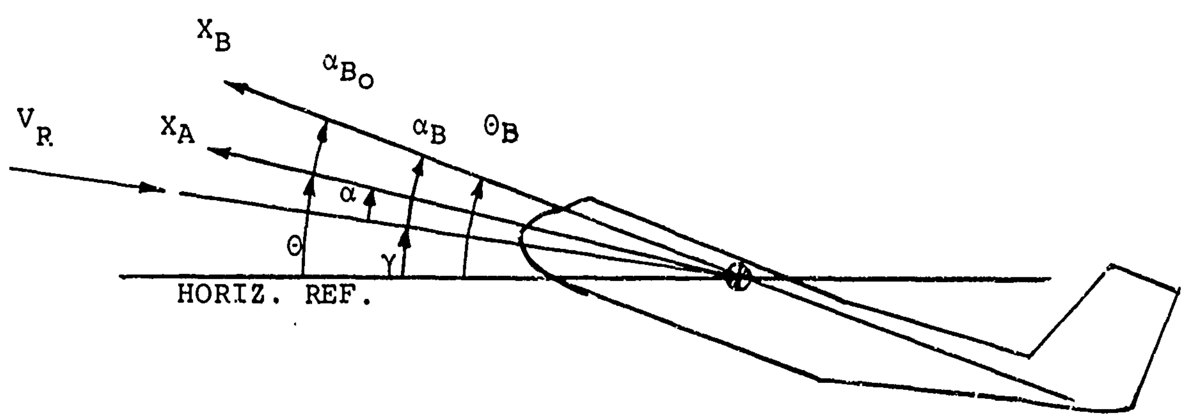


Figure 4: Sketches showing Relationship of A and B Frames

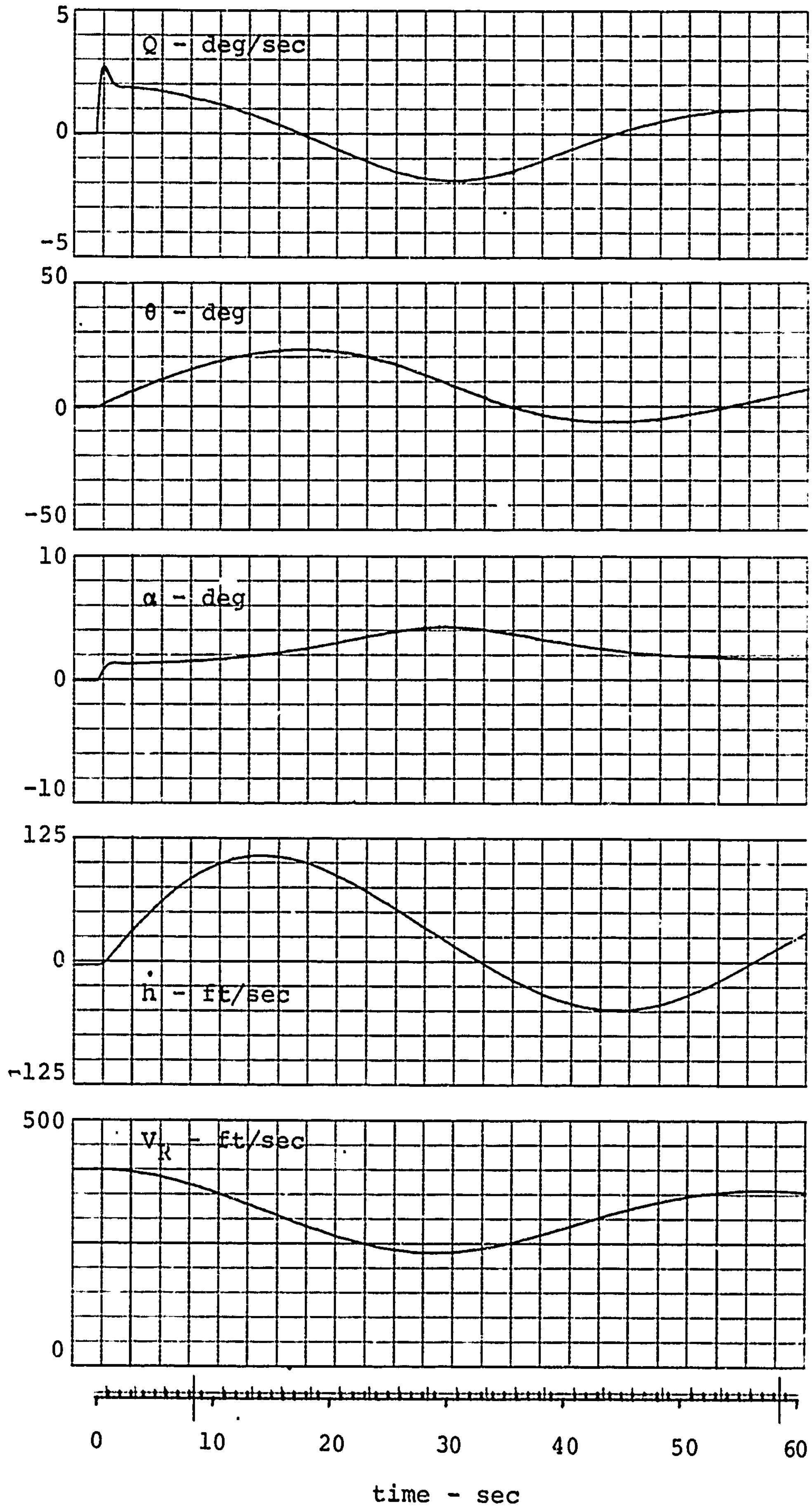


FIG. 5 Response to 1° Step Elevator Input (Buffalo, Cruise)

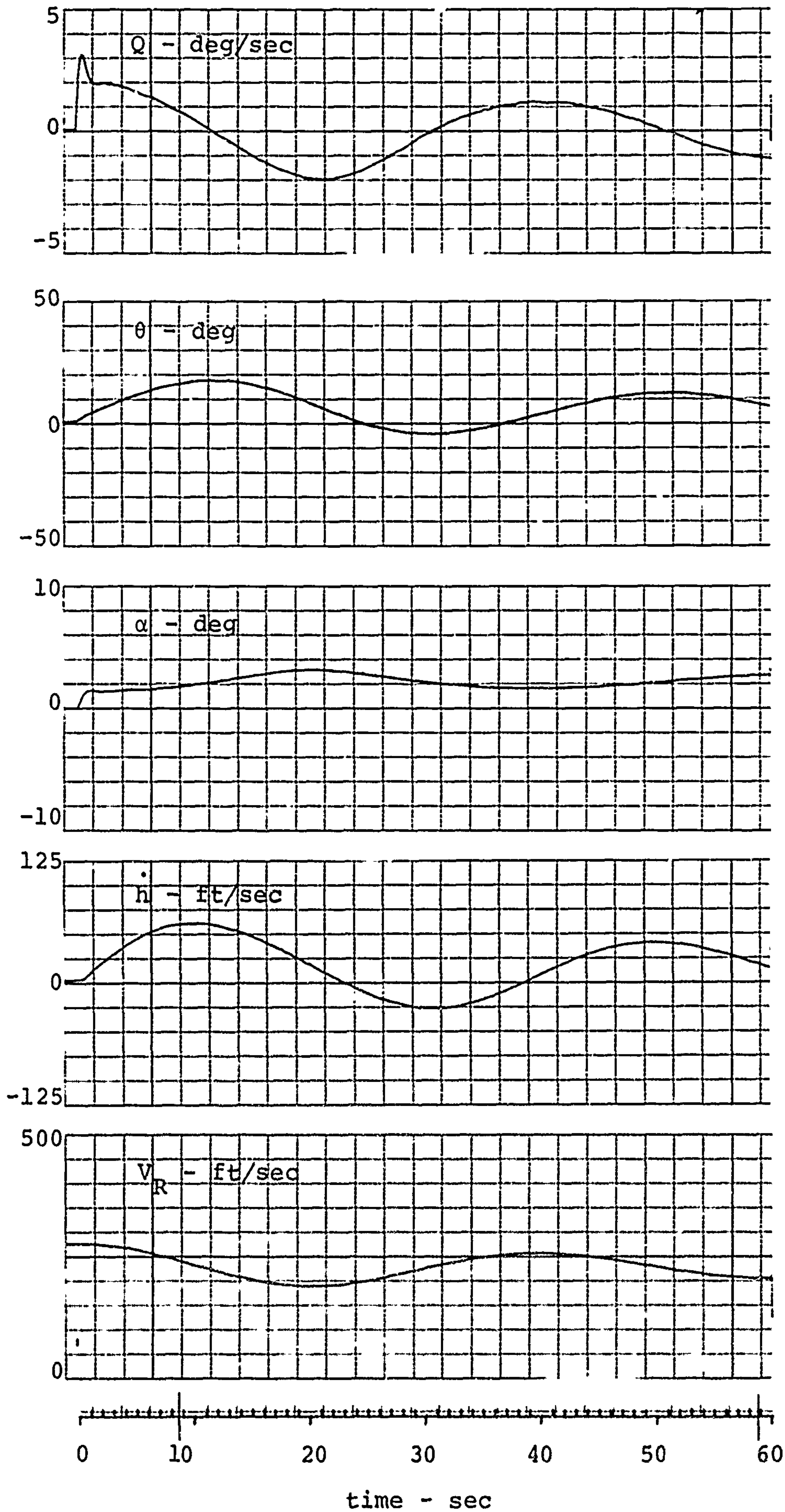


FIG.6 Response to 1° Step Elevator Input (Twin Otter, Cruise)

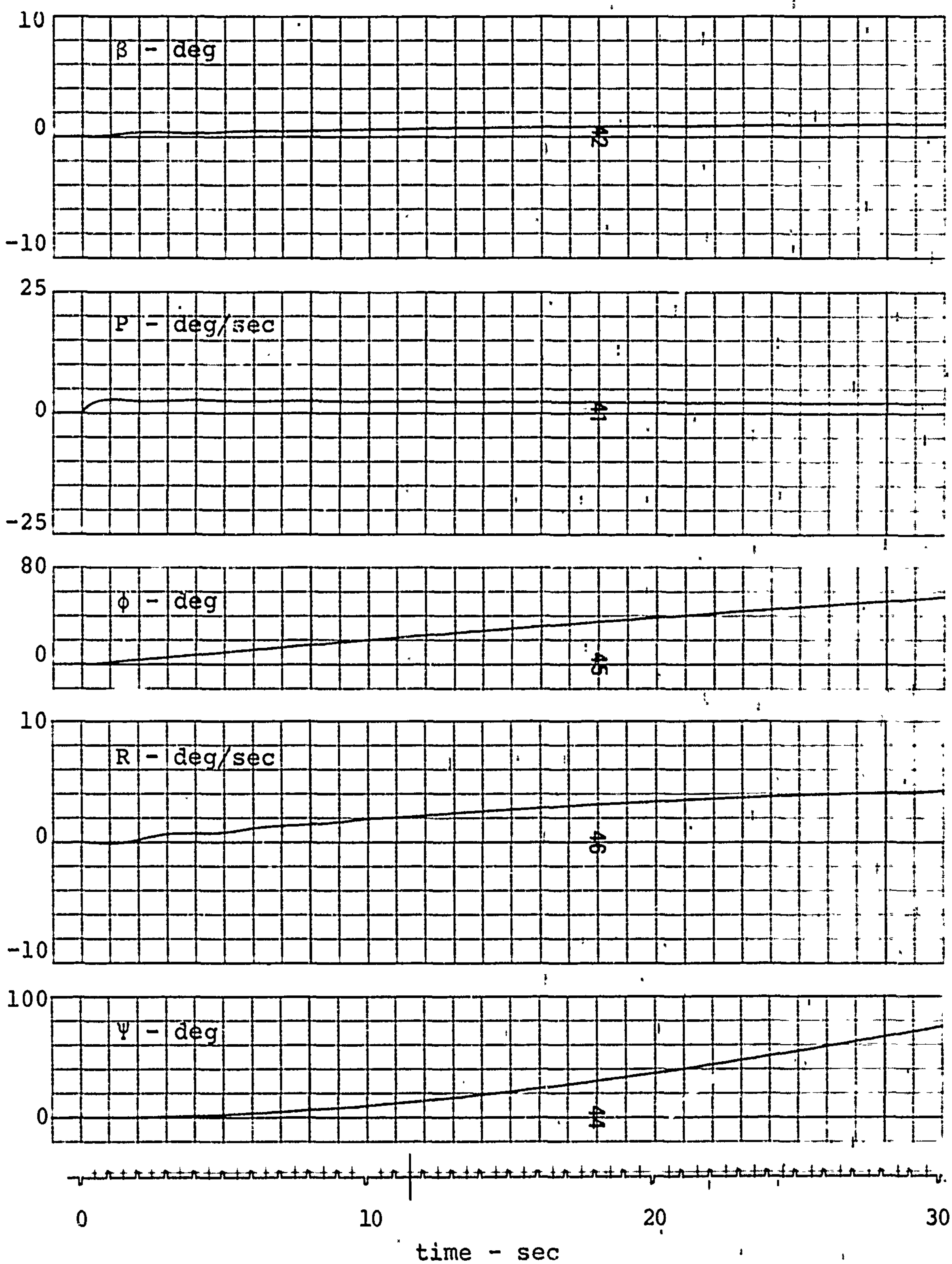


FIG. 7 Response to 1° Step Aileron Input (Buffalo, Cruise)

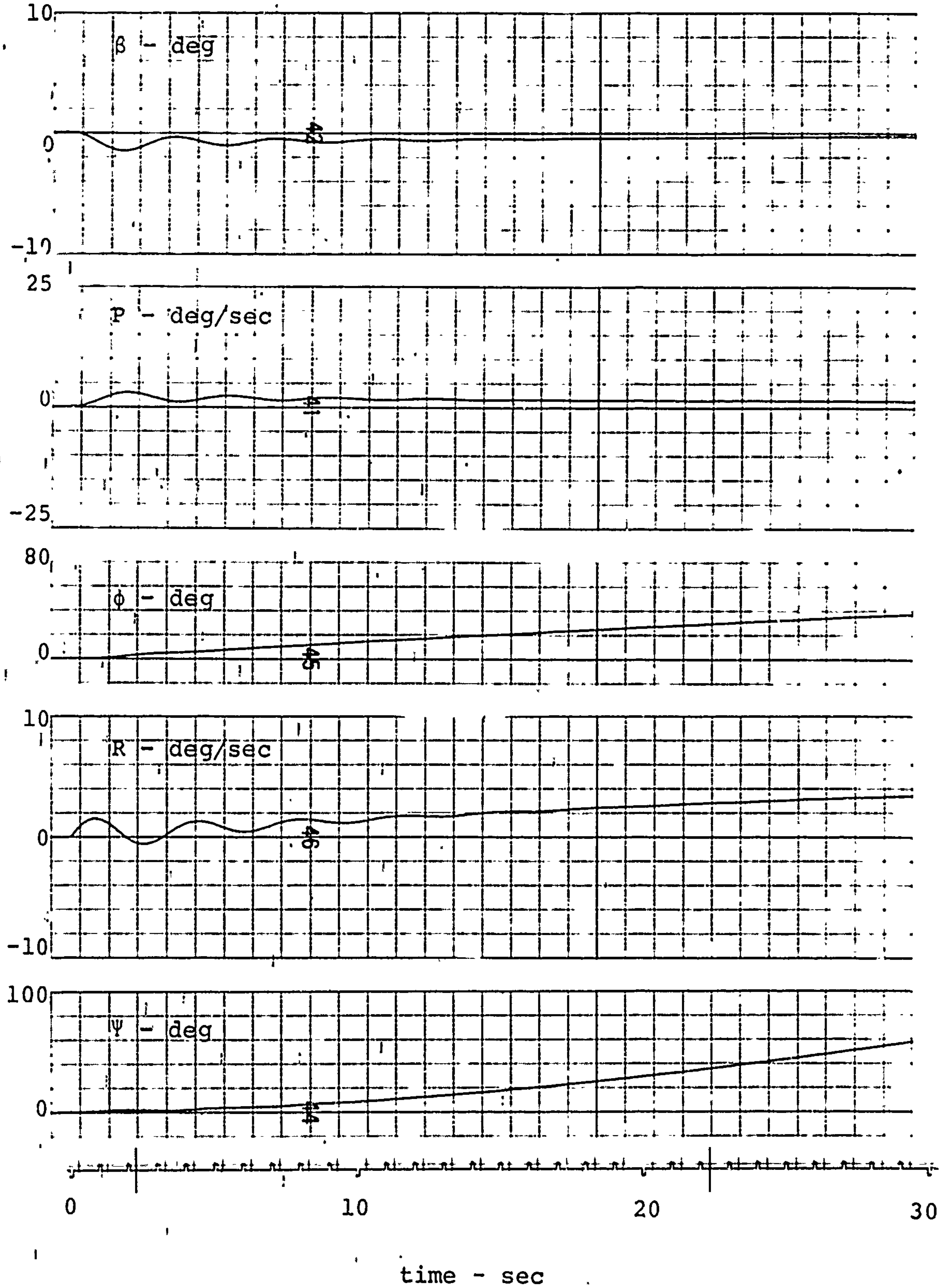


FIG. 8 Response to 1° Step Rudder Input (Buffalo, Cruise)

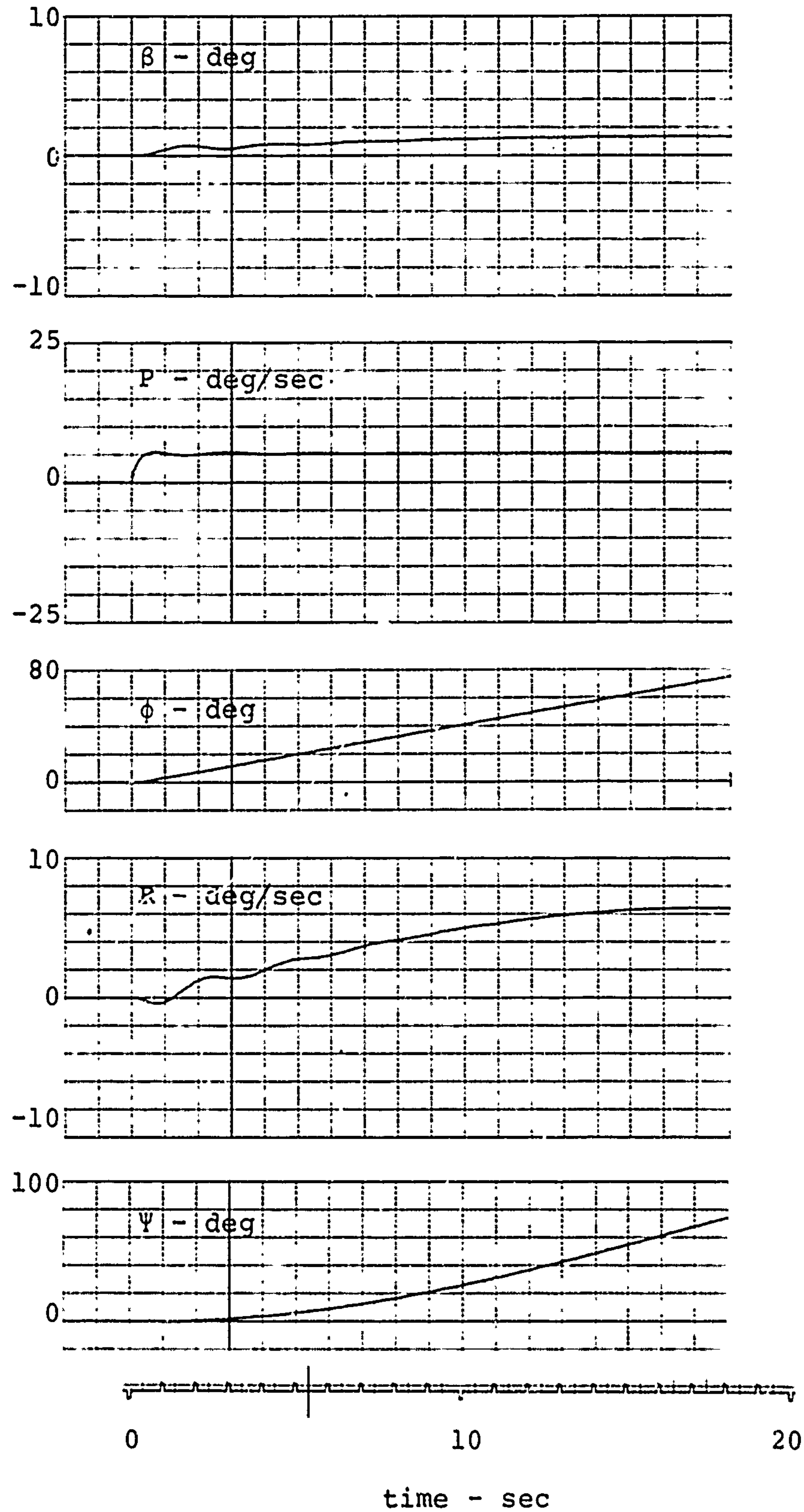


FIG. 9 Response to 1° Step Aileron Input (Twin Otter, Cruise)

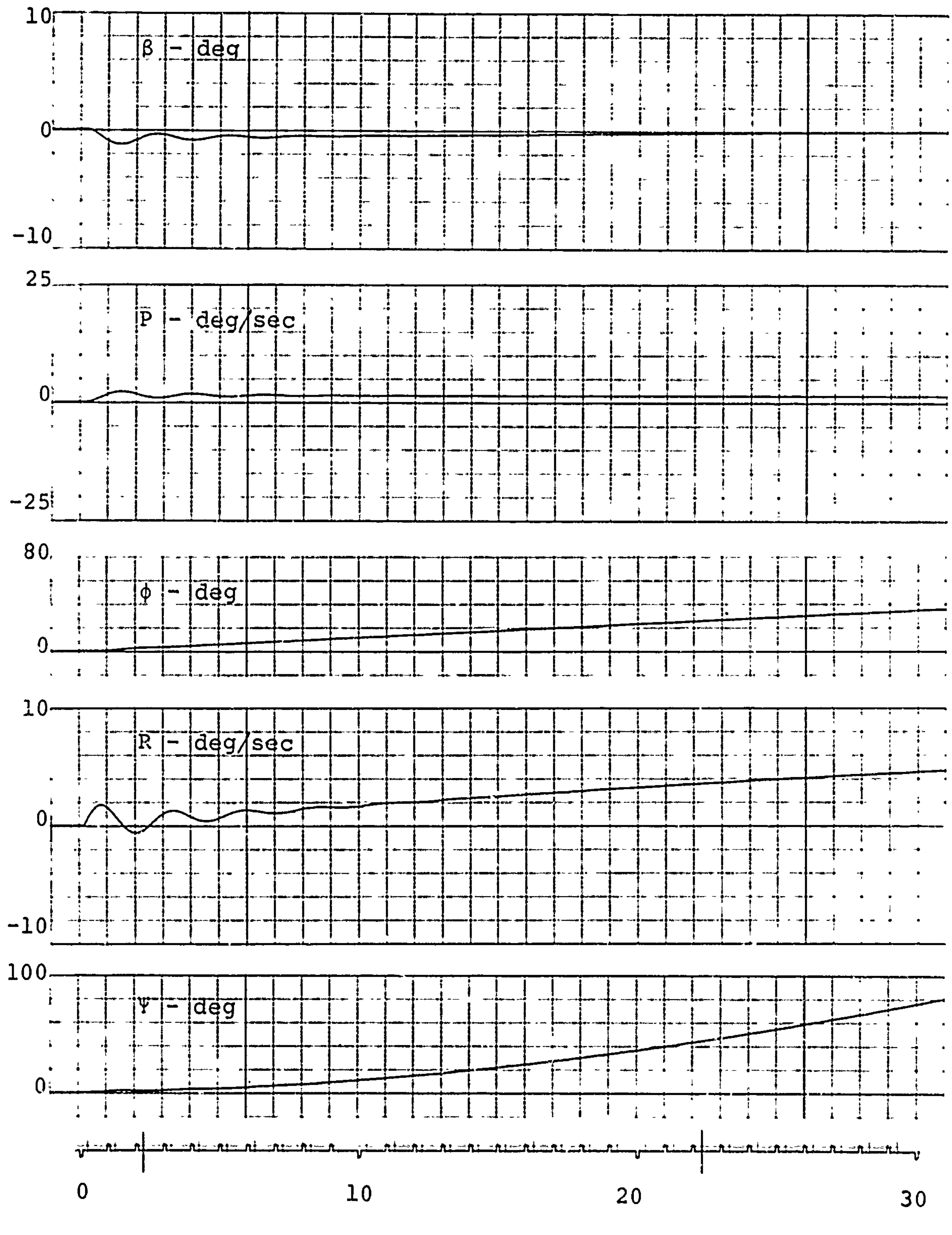


FIG. 10 Response to 1° Step Rudder Input (Twin Otter, Cruise)

Appendix: Development of Expression for Thrust

The longitudinal force equation of Section II includes the total thrust force T. Since directly applicable data on the propulsive system installation of the "Buffalo" and "Twin Otter" are not available, an expression for T is developed here for use in the simulation. Although the expression is adequate for the simulation documented in this report, it must be considered an approximate one.

Thrust developed by a propeller is

$$T = \eta_p \frac{P}{V}$$

where P is the power supplied to the propeller, V is the velocity of the propeller with respect to the air, and η_p is the propeller efficiency. Power supplied to the propeller is expressed in this report as

$$P = \sigma P_o \xi$$

where σ is the atmospheric density ratio, P_o is the rated power output of the engine at sea level, and ξ is the pilot's throttle deflection, expressed as a fraction of the deflection for rated power.

Propeller efficiency, η_p, is obtained from Figure 3-17 of Reference 2 (reproduced here)

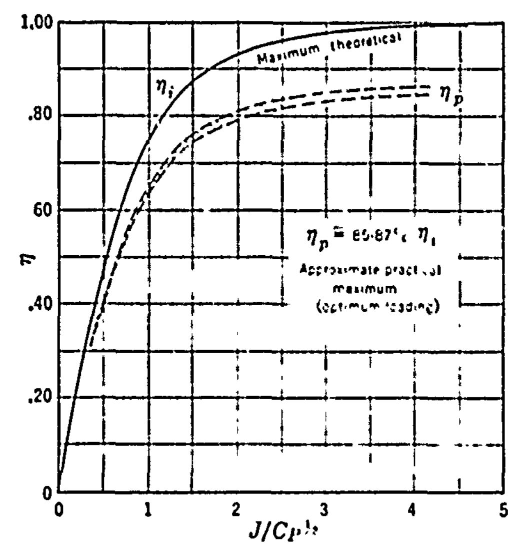


FIGURE 3-17. Propeller efficiency (incompressible).

as a function of advance ratio J and power coefficient C_p. By definition,

$$J = \frac{60V}{ND}$$

and

$$C_p = \frac{.5P/1000}{\sigma (N/1000)^3 (D/10)^5}$$

where V is in ft/sec, N is propeller speed in rpm, D is propeller diameter in feet, and P is power in horsepower units.

For the "Buffalo" (Figure 2) with its two T64-GE-10 engines, N = 1160 rpm, D = 14.7 ft, and P_o = 2850 ESHP/engine, so, at sea level,

$$C_p = .137$$

or

$$C_p^{1/3} = .515$$

Entering Figure 3-17 at $J/C_p^{1/3} = 2.0$ gives $\eta_p = .79$. This value of $J/C_p^{1/3}$ corresponds to J = 1.03 or V = 293 fps. Therefore

$$T = .79 \frac{(2850)(550)}{293} = 4220 \text{ lbs/engine}$$

or, for two engines, 8440 lbs. Repeating this calculation for other values of $J/C_p^{1/3}$ produces the required thrust vs speed relationship.

This thrust - speed curve can be represented by an equation of the form

$$T_{\substack{\text{rated power,} \\ \text{sea level}}} = \frac{T_{static}}{1 + C_{T_1} V_R + C_{T_2} V_R^2}$$

By curve-fitting techniques, it can be established that, for the "Buffalo",

T_{static} = 22400 lbs

C_{T_1} = .00370 fps^{-1}

C_{T_2} = 6.51×10^{-6} fps^{-2}

The process is repeated for the "Twin Otter" (Figure 1). For this aircraft (with two PT6A-20 engines), N = 2200 rpm, D = 8.5 ft, and P_o = 652 ESHP/engine. The required constants are established as:

T_{static} = 5750 lbs

C_{T_1} = .00378 fps^{-1}

C_{T_2} = 9.07×10^{-6} fps^{-2}

These values are tabulated in Section II where simulation input quantities are listed.