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

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TECHNICAL NOTE

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Prepared for
DEPARTMENT OF TRANSPORTATION
FEDERAL AVIATION ADMINISTRATION
WASHINGTON, D.C. 20590
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1. Report No. | DOT-TSC-FAA-71-14
---|---
2. Government Accession No. | 3. Recipient’s Catalog No.  
4. Title and Subtitle | REAL-TIME SIMULATION PROGRAM FOR DE HAVILLAND (CANADA) "BUFFALO" AND "TWIN OTTER" STOL TRANSPORTS
5. Report Date | June 25, 1971
6. Performing Organization Code | 
7. Author(s) | B. A. MAC DONALD, MEL GARELICK, J. HAAS
9. Performing Organization Name and Address | Department of Transportation  
Transportation Systems Center  
55 Broadway, Cambridge, Mass. 02142
10. Work Unit No. | R-2901
11. Contract or Grant No. | FA-18
12. Sponsoring Agency Name and Address | Department of Transportation  
Federal Aviation Administration  
Washington, D.C. 20590
13. Type of Report and Period Covered | Technical Note
15. Supplementary Notes | *Service Technology Corporation  
Cambridge, Ma. 02142
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.
17. Key Words | Aircraft Math Model, STOL Aircraft Stability and Control, Aircraft Simulation
18. Distribution Statement | Availability is Unlimited. Document may be Released To the National Technical Information Service, Springfield, Virginia 22151, for Sale to the Public.
19. Security Classif. (of this report) | Unclassified
20. Security Classif. (of this page) | Unclassified
21. No. of Pages | 57
22. Price | $3.00
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<tr>
<td>a</td>
<td>Aircraft lift curve slope</td>
<td>rad(^{-1})</td>
</tr>
<tr>
<td>AR</td>
<td>Aspect ratio of wing = (b^2/s)</td>
<td></td>
</tr>
<tr>
<td>B(_{nm})</td>
<td>Elements of A-frame to L-frame transformation matrix</td>
<td></td>
</tr>
<tr>
<td>b</td>
<td>Wing span</td>
<td>ft</td>
</tr>
<tr>
<td>c</td>
<td>Mean chord of wing</td>
<td>ft</td>
</tr>
<tr>
<td>CD</td>
<td>Aircraft drag coefficient</td>
<td></td>
</tr>
<tr>
<td>C(_{DF})</td>
<td>Aircraft parasite drag coefficient</td>
<td></td>
</tr>
<tr>
<td>ΔCD</td>
<td>Aircraft drag coefficient less wing drag coefficient</td>
<td></td>
</tr>
<tr>
<td>CL</td>
<td>Aircraft lift coefficient</td>
<td></td>
</tr>
<tr>
<td>C(_{LO})</td>
<td>Trimmed aircraft lift coefficient</td>
<td></td>
</tr>
<tr>
<td>C(<em>{m})</em>(\delta ) _t _m _q _\alpha _\delta _) _p _\beta _\delta _a</td>
<td>Pitching moment coefficient which may be made variable to shape trim (\delta _e ) _vs _V_R _curve (C_m _t = 0 _in _this _report)</td>
<td></td>
</tr>
<tr>
<td>C(<em>{m})</em>(\alpha )</td>
<td>Pitching moment coefficient due to angle of attack</td>
<td></td>
</tr>
<tr>
<td>C(<em>{m})</em>(\delta _\alpha )</td>
<td>Pitching moment coefficient due to angle of attack rate</td>
<td></td>
</tr>
<tr>
<td>C(<em>{m})</em>(\delta _)</td>
<td>Pitching moment coefficient due to elevator deflection</td>
<td></td>
</tr>
<tr>
<td>C(_{p})</td>
<td>Rolling moment coefficient due to roll rate</td>
<td></td>
</tr>
<tr>
<td>C(_{\beta})</td>
<td>Rolling moment coefficient due to sideslip angle</td>
<td></td>
</tr>
<tr>
<td>C(_{\delta _a})</td>
<td>Rolling moment coefficient due to aileron deflection</td>
<td></td>
</tr>
</tbody>
</table>
\( C_{LR} \)  
CLR rolling moment coefficient due to yaw rate

\( C_{LRFIN} \)  
fin contribution to rolling moment coefficient due to yaw rate

\( C_{n_P} \)  
yawing moment coefficient due to roll rate

\( C_{n_{FIN}} \)  
fin contribution to yawing moment coefficient due to yaw rate

\( C_{n_r} \)  
yawing moment coefficient due to yaw rate

\( C_{nRFIN} \)  
fin contribution to yawing moment coefficient due to yaw rate

\( C_{n_P} \)  
yawing moment coefficient due to roll rate

\( C_{n_{FIN}} \)  
fin contribution to yawing moment coefficient due to yaw rate

\( C_{n_B} \)  
yawing moment coefficient due to sideslip angle

\( C_{NOLR} \)  
yawing moment coefficient due to rudder deflection

\( C_{y_P} \)  
side force coefficient due to roll rate

\( C_{yR} \)  
side force coefficient due to yaw rate

\( C_{y_B} \)  
side force coefficient due to sideslip angle

\( C_{T_1} \)  
empirical coefficient in thrust equation

\( C_{T_2} \)  
empirical coefficient in thrust equation

\( D \)  
aircraft drag

\( e \)  
aircraft efficiency factor

\( g \)  
gravitational constant \( = 32.2 \) ft/sec\(^2\)

\( h \)  
alitude \( = -z_L \) ft

\( h_{ATM} \)  
characteristic density altitude of atmosphere ft
\( i \), \( j \), \( k \) - unit vectors along the \( x, y, \) and \( z \), axes of the \( () \) coordinate frame, respectively.

**IAS** - indicated airspeed, **mph**

**I_x, I_y, I_z** - aircraft rolling, pitching, and yawing moment of inertia, respectively, **slug-ft**\(^2\)**

**J_{xz}** - product of inertia \( = \int xz \, dm \), **slug-ft**\(^2\)**

**L** - 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** - rolling moment due to roll rate, **ft-lbs/\( \text{rad/} \)sec**

**L_R** - rolling moment due to yaw rate, **ft-lbs/\( \text{rad/} \)sec**

**L_V** - rolling moment due to sideslip velocity, **ft-lbs/fps**

**L_\delta_a** - rolling moment due to aileron deflection, **ft-lbs/\( \text{rad/} \)sec**

**l/m** - 1/aircraft mass, **slugs\(^{-1}\)**

**N_P** - yawing moment due to roll rate, **ft-lbs/\( \text{rad/} \)sec**

**N_R** - yawing moment due to yaw rate, **ft-lbs/\( \text{rad/} \)sec**

**N_V** - yawing moment due to sideslip velocity, **ft-lbs/fps**

**N_\delta_r** - yawing moment due to rudder deflection, **ft-lbs/\( \text{rad/} \)sec**

**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** - dynamic pressure, **lbs/ft**\(^2\)**

**S** - wing area, **ft**\(^2\)**
<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
<th>Unit</th>
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<tbody>
<tr>
<td>T</td>
<td>THRUST</td>
<td>lbs</td>
</tr>
<tr>
<td>-</td>
<td>TMDLE</td>
<td>sec</td>
</tr>
<tr>
<td>-</td>
<td>TMTHR</td>
<td>sec</td>
</tr>
<tr>
<td>T_{static}</td>
<td>TSTAT</td>
<td>lbs</td>
</tr>
<tr>
<td>U,V,W</td>
<td>U,V,W</td>
<td>fps</td>
</tr>
<tr>
<td>U_{w},V_{w},W_{w}</td>
<td>UW,VW,WW</td>
<td>fps</td>
</tr>
<tr>
<td>V_{R}</td>
<td>VR</td>
<td>fps</td>
</tr>
<tr>
<td>W</td>
<td>WEIGHT</td>
<td>lbs</td>
</tr>
<tr>
<td>X,Y,Z</td>
<td></td>
<td></td>
</tr>
<tr>
<td>X(),Y(),Z()</td>
<td></td>
<td></td>
</tr>
<tr>
<td>X_{L},Y_{L},Z_{L}</td>
<td>X,Y,-H</td>
<td>ft</td>
</tr>
<tr>
<td>X_{L},Y_{L},Z_{L}</td>
<td>X DOT, Y DOT, -H DOT</td>
<td>fps</td>
</tr>
<tr>
<td>x_{w_{L}},y_{w_{L}},z_{w_{L}}</td>
<td>XSS,XSS, ZSS</td>
<td>fps</td>
</tr>
<tr>
<td>Y_P</td>
<td>YP</td>
<td>lbs/\text{rad}</td>
</tr>
<tr>
<td>Y_r</td>
<td>YR</td>
<td>lbs/\text{rad}</td>
</tr>
<tr>
<td>Y_V</td>
<td>YV</td>
<td>lbs/\text{fps}</td>
</tr>
</tbody>
</table>
\( \alpha \)  ALF  angle from the remote wind vector \( V_R \) to the \( X_A \) axis  rad

\( \dot{\alpha} \)  DALF  \( \alpha/dt \)  rad/sec

\( \alpha_B \)  -  angle from the remote wind vector \( V_R \) to the \( X_B \) axis  rad

\( \alpha_{BO} \)  ALFBO  angle between the body-fixed \( X_A \) and \( X_B \) axes  rad

\( \alpha_{BOL} \)  ALFBOL  value of \( \alpha_B \) for which no lift is developed by the aircraft  rad

\( \beta \)  BETA  aircraft sideslip angle  rad

\( \gamma \)  -  angle from the horizontal reference line to the remote wind vector \( V_R \): \( \gamma = \theta - \alpha \)  rad

\( \delta_a \)  DLA  aileron deflection  rad

\( \delta_e \)  DLE  elevator deflection  rad

\( \delta_r \)  DLR  rudder deflection  rad

\( \Theta \)  THETA  (See definition of Euler angles \( \psi, \theta, \phi \) below)

\( \Theta_B \)  -  angle from the horizontal reference line to the \( X_B \) axis  rad

\( \xi \)  THROT  pilot throttle input as fraction of maximum input

\( \rho \)  RHO  atmospheric air density  slugs/ft\(^3\)

\( \rho_o \)  RHSEA  atmospheric air density at sea level, std day  slugs/ft\(^3\)

\( \sigma \)  SIG  \( \rho/\rho_o \)

\( \psi, \theta, \phi \)  PSI, THETA  Euler angles relating \( L, C, \) and \( A \) coordinate frames (further defined in Figure 3)  rad
### SUBSCRIPTS

<table>
<thead>
<tr>
<th>Subscript</th>
<th>Description</th>
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<tbody>
<tr>
<td>A</td>
<td>aircraft body coordinate frame</td>
</tr>
<tr>
<td>B</td>
<td>body reference coordinate frame</td>
</tr>
<tr>
<td>C</td>
<td>Earth-aircraft control coordinate frame</td>
</tr>
<tr>
<td>L</td>
<td>Earth local-vertical coordinate frame</td>
</tr>
<tr>
<td>cr</td>
<td>design economy cruise condition</td>
</tr>
<tr>
<td>o</td>
<td>equilibrium or reference condition</td>
</tr>
<tr>
<td>OL</td>
<td>zero lift value</td>
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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 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 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 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 $\alpha(0) = \alpha_0$ is zero. (In this report, $\theta$ and $\alpha$ 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.

The Earth-Aircraft Control Coordinate Frame (C) 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 from $X_L$ to $X_C$;

2. Pitch (θ): angle of rotation about Y from $X_C$ to $X_A$;

3. Roll (φ): angle of rotation about X from $Y_C$ to $Y_A$.

These Euler angle rotations are shown in Figure 3.

The Body Reference Coordinate Frame (B) 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 $\alpha_{BO}$. The $Y_B$ axis coincides with the $Y_A$ axis, and the $Z_B$ axis (positive downward) forms an orthogonal set.

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

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

$$\theta_B = \theta + \alpha_{BO}$$

*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

$$\theta_{B_0} = \alpha_{B_0}$$

The trim angle $\alpha_{B_0}$ 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,

$$C_{L_{EO}} = a (\alpha_{B_0} - \alpha_{B_{OL}})$$

Next, assume that wing incidence has been chosen by the aircraft manufacturer to produce a level fuselage attitude ($\alpha_{B_0} = 0$) 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 $\alpha$ at which lift coefficient 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 $\alpha_{BO}$ at any other trim lift coefficient:

$$\alpha_{BO} = \frac{C_{L0}}{a} + \alpha_{B_{OL}}$$

(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, $a = 5.2/\text{rad}$, so

$$\alpha_{B_{OL}} = -.085 = -4.8^\circ \quad \text{(Buffalo)}$$
$$\alpha_{B_{OL}} = -.092 = -5.3^\circ \quad \text{(Twin Otter)}$$

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 \( \mathbf{R} \) can be resolved into its A-frame or L-frame components:

$$\mathbf{R} = R_{X_A} i_A + R_{Y_A} j_A + R_{Z_A} k_A$$

$$= R_{X_L} i_L + R_{Y_L} j_L + R_{Z_L} k_L$$

where $i$, $j$, and $k$ are unit vectors in the indicated frames. L-frame components of $\mathbf{R}$ can be expressed in terms of A-frame components of $\mathbf{R}$ 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 \( B_{11} = \cos \psi \cos \Theta \)  
\( B_{12} = \cos \psi \sin \Theta \sin \phi - \sin \psi \cos \phi \)  
\( B_{13} = \cos \psi \sin \Theta \cos \phi + \sin \psi \sin \phi \)  
\( B_{21} = \sin \psi \cos \Theta \)  
\( B_{22} = \sin \psi \sin \Theta \sin \phi + \cos \psi \cos \phi \)  
\( B_{23} = \sin \psi \sin \Theta \cos \phi - \cos \psi \sin \phi \)  
\( B_{31} = -\sin \Theta \)  
\( B_{32} = \cos \Theta \sin \phi \)  
\( B_{33} = \cos \Theta \cos \phi \)

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

\[
\begin{bmatrix}
R_{x_A} \\
R_{y_A} \\
R_{z_A}
\end{bmatrix} =
\begin{bmatrix}
B_{11} & B_{12} & B_{13} \\
B_{21} & B_{22} & B_{23} \\
B_{31} & B_{32} & B_{33}
\end{bmatrix}
\begin{bmatrix}
R_{x_L} \\
R_{y_L} \\
R_{z_L}
\end{bmatrix}
\]

-8-
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:

$$\dot{x}_L = B_{11} U + B_{12} V + B_{13} W \quad \text{(fps)}$$  \hspace{1cm} (12)

$$\dot{y}_L = B_{21} U + B_{22} V + B_{23} W \quad \text{(fps)}$$  \hspace{1cm} (13)

$$\dot{z}_L = -B_{31} U - B_{32} V - B_{33} W \quad \text{(fps)}$$  \hspace{1cm} (14)

### IIC Provisions for Atmospheric Disturbances (Winds)

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

$$U_w = U - [B_{11} \dot{x}_W + B_{21} \dot{y}_W + B_{31} \dot{z}_W] \quad \text{(fps)}$$  \hspace{1cm} (15)

$$V_w = V - [B_{12} \dot{x}_W + B_{22} \dot{y}_W + B_{32} \dot{z}_W] \quad \text{(fps)}$$  \hspace{1cm} (16)

$$W_w = W - [B_{13} \dot{x}_W + B_{23} \dot{y}_W + B_{33} \dot{z}_W] \quad \text{(fps)}$$  \hspace{1cm} (17)

Material contained in this report is sufficient to allow introduction of steady state wind components. The desired winds are simply input as $\dot{x}_W$, $\dot{y}_W$, and $\dot{z}_W$. 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.
IIID Airframe Equations of Motion

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

\[ m [U + QW - RV + g \sin \theta] = X \]  
\[ m [V + RU - PW - g \cos \theta \sin \phi] = Y \]  
\[ m [W + PV - QU - g \cos \theta \cos \phi] = Z \]  
\[ I_x \ddot{P} + (I_z - I_y) \dot{Q} - J \dot{\gamma} = L \]  
\[ I_y \ddot{Q} + (I_x - I_z) \dot{R} - J \dot{\phi} = M \]  
\[ I_z \ddot{R} + (I_y - I_x) \dot{P} - J \dot{\theta} = N \]

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

\[ \dot{\phi} = P + \dot{\psi} \sin \theta \]  
\[ \dot{\theta} = Q \cos \phi - R \sin \phi \]  
\[ \dot{\psi} = Q \sin \phi + R \cos \phi \]

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

\[
\begin{align*}
\dot{U} &= RV - QW - g \sin \theta + \frac{X}{m} \quad \text{ft/sec}^2 \quad (21) \\
\dot{V} &= PW - RU + g \cos \theta \sin \phi + \frac{Y}{m} \quad \text{ft/sec}^2 \quad (22) \\
\dot{W} &= QU - PV + g \cos \theta \cos \phi + \frac{Z}{m} \quad \text{ft/sec}^2 \quad (23) \\
\dot{P} &= \frac{L}{I_x} \quad \text{rad/sec}^2 \quad (24) \\
\dot{Q} &= \frac{M}{I_y} \quad \text{rad/sec}^2 \quad (25) \\
\dot{R} &= \frac{N}{I_z} \quad \text{rad/sec}^2 \quad (26)
\end{align*}
\]

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,

\[
\begin{align*}
X &= T - D \cos \alpha + L \sin \alpha \quad \text{lbs} \quad (27) \\
Z &= -(L \cos \alpha + D \sin \alpha) \quad \text{lbs} \quad (28)
\end{align*}
\]
The terms $X_q$, $Z_q$, $Z_w$, and $Z_{\delta e}$ 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 $\xi$: 

\[ \text{Thrust} = \text{empirically-derived expression} \]
\[ T = \frac{\sigma T_{\text{static}}}{1 + C_{T_1} V_R + C_{T_2} V_R^2} \cdot \xi \quad \text{(lbs)} \quad (29) \]

where \( 0 \leq \xi \leq 1.0 \),

\[ \sigma = e^{-h/h_{\text{atm}}} \quad (-) \quad (30) \]

and

\[ V_R = \left[ U_w^2 + V_w^2 + W_w^2 \right]^{1/2} \quad \text{(fps)} \quad (31) \]

Lift and drag are calculated from the standard relationships:

\[ L = C_L qS \quad \text{(lbs)} \quad (32) \]

\[ D = C_D qS \quad \text{(lbs)} \quad (33) \]

where

\[ C_L = C_{L_0} + \alpha \alpha \quad (-) \quad (34) \]

\[ C_D = C_{D_0} + \frac{C_L^2}{\pi \epsilon A R} \quad (-) \quad (35) \]

\[ q = \frac{1}{2} \rho V_R^2 \quad \text{(lbs/ft}^2) \quad (36) \]

\[ \rho = \sigma \rho_o \quad \text{(sl/ft}^3) \quad (37) \]

and

\[ \alpha = \tan^{-1} \frac{W_w}{U_w} \quad \text{(rad)} \quad (38) \]

The expression for pitching moment used in the simulation is

\[ M = qS \left[ C_{m_t} + C_m \alpha + \frac{C_m}{2V_R} (C_m \dot{\alpha} + C_m \omega) + C_m \delta_e \right] \quad \text{(ft/lbs)} \quad (39) \]

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 \( \delta_e \) 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:

\[
\dot{\alpha} = \frac{d}{dt} \left( \tan^{-1} \left( \frac{W_w}{U_w} \right) \right)
= \frac{U_w \dot{W}_w - W_w \dot{U}_w}{U_w^2 + W_w^2}
\]

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

\[
\dot{\alpha} = (\dot{W}_w - \frac{U_w}{U_w} \dot{U}_w) \cos^2 \alpha \left( \frac{U_w}{U_w} \right) \quad \text{(rad/sec)}
\]

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:

\[
Y = Y_v V_w + Y_r R + Y_p P \quad \text{(lbs)}
\]

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

\[
N = N_v V_w + N_r R + N_p P + N_{\delta_r} \delta_r \quad \text{(ft-lbs)}
\]
The terms \( Y_\delta \), \( L_\delta \), and \( N_\delta \), sometimes included in the lateral equations, have been omitted in the present analysis because of their negligible effects.

The coefficients of these equations are

\[
Y_v = \frac{1}{2} \rho V_R S C_{Y_\beta} \quad \text{(lbs/fps)} \tag{44}
\]

\[
Y_r = \frac{1}{4} \rho V_R S_b C_{Y_r} \quad \text{(lbs/\text{rad/ sec})} \tag{45}
\]

\[
Y_p = \frac{1}{4} \rho V_R S_b C_{Y_p} \quad \text{(lbs/\text{rad/ sec})} \tag{46}
\]

\[
L_v = \frac{1}{2} \rho V_R S_b C_{r_\beta} \quad \text{(ft-lbs/fps)} \tag{47}
\]

\[
L_r = \frac{1}{4} \rho V_R S_b^2 C_{r_\beta} \quad \text{(ft-lbs/\text{rad/ sec})} \tag{48}
\]

\[
C_{r_{\beta}} = C_{r_{FIN}} + C_L/4 \quad \text{(-)} \tag{49}
\]

\[
L_p = \frac{1}{4} \rho V_R S_b^2 C_{r_p} \quad \text{(ft-lbs/\text{rad/ sec})} \tag{50}
\]

\[
L_{\delta a} = q S_b C_{r_{\delta a}} \quad \text{(ft-lbs/\text{rad})} \tag{51}
\]

\[
N_v = \frac{1}{2} \rho V_R S_b C_{n_\beta} \quad \text{(ft-lbs/fps)} \tag{52}
\]

\[
N_r = \frac{1}{4} \rho V_R S_b^2 C_{n_r} \quad \text{(ft-lbs/\text{rad/ sec})} \tag{53}
\]

\[
C_{r_{FIN}} = C_{n_{r_{FIN}}} - \frac{C_{d_{\text{wing}}}}{4} \quad \text{(-)} \tag{54}
\]

\[
N_p = \frac{1}{4} \rho V_R S_b^2 C_{n_p} \quad \text{(ft-lb/\text{rad/ sec})} \tag{55}
\]

\[
C_{n_{p_{FIN}}} = C_{n_p} - \frac{C_L}{4} \left(1 - \frac{a}{\pi A_R} \right) \quad \text{(-)} \tag{56}
\]

\[
N_{\delta r} = q S_b C_{n_{\delta r}} \quad \text{(ft-lbs/\text{rad})} \tag{57}
\]
The equation for sideslip angle is

\[ \beta = \tan^{-1} \frac{V}{U} \]  

(58)

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

\[ U = U(0) + \int_0^t U \, dt \]  

(fps)  

(59)

\[ V = V(0) + \int_0^t V \, dt \]  

(fps)  

(60)

\[ W = W(0) + \int_0^t W \, dt \]  

(fps)  

(61)

\[ P = \int_0^t P \, dt \]  

(rad/sec)  

(62)

\[ Q = \int_0^t Q \, dt \]  

(rad/sec)  

(63)

\[ R = \int_0^t R \, dt \]  

(rad/sec)  

(64)

\[ \psi = \int_0^t \psi \, dt \]  

(rad)  

(65)

\[ \theta = \int_0^t \theta \, dt \]  

(rad)  

(66)

\[ \phi = \int_0^t \phi \, dt \]  

(rad)  

(67)

\[ x_L = \int_0^t x_L \, dt \]  

(ft)  

(68)

\[ y_L = \int_0^t y_L \, dt \]  

(69)

\[ h = -z_L = h(0) + \int_0^t h \, dt \]  

(70)
**IIIE 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** \( \frac{h}{60} \) (fpm)
- **Turn Rate Indicator Output** \( 57.3 R \) (deg/sec)
- **Slip Indicator Output**
  \[ \frac{g \cos \theta \sin \phi - V - RU + PW}{g \cos \theta \cos \phi - \dot{W} - PV + QU} \] (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.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Buffalo</td>
<td>Twin Otter</td>
</tr>
<tr>
<td>a, rad⁻¹</td>
<td>5.2</td>
<td>5.2</td>
</tr>
<tr>
<td>AR</td>
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<td>10</td>
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<tr>
<td>b, ft</td>
<td>96</td>
<td>65</td>
</tr>
<tr>
<td>c, ft</td>
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<td>6.5</td>
</tr>
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<td>Cₐ₀</td>
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<td>.039</td>
</tr>
<tr>
<td>Cₐ₁</td>
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<td>.035</td>
</tr>
<tr>
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<td>0</td>
</tr>
<tr>
<td>Cₐ₃</td>
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<td>-24.6</td>
</tr>
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<td>Cₐ₄</td>
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<tr>
<td>Cₐ₅</td>
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<td>-6.15</td>
</tr>
<tr>
<td>Cₐ₆</td>
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</tr>
<tr>
<td>Cₐ₇</td>
<td>-.53</td>
<td>-.53</td>
</tr>
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- 18 -
<table>
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<tr>
<th>Parameter</th>
<th>Value</th>
<th>Value</th>
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</thead>
<tbody>
<tr>
<td></td>
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</tr>
<tr>
<td>$C_{l\beta}$</td>
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<td>$C_{n\text{r fin}}$</td>
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<tr>
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</tr>
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<td>$C_{n\delta r}$</td>
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<td>.107</td>
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<td>-.492</td>
</tr>
<tr>
<td>$C_{T_1, fps^{-1}} (1)$</td>
<td>.00370</td>
<td>.00378</td>
</tr>
<tr>
<td>$C_{T_2, fps^{-2}} (1)$</td>
<td>$6.51 \times 10^{-6}$</td>
<td>$9.07 \times 10^{-6}$</td>
</tr>
<tr>
<td>$e$</td>
<td>.75</td>
<td>.75</td>
</tr>
<tr>
<td>$h_{\text{ATM}, ft} (2)$</td>
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<td>32500</td>
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<tr>
<td>$I_x, \text{slug-ft}^2$</td>
<td>273000</td>
<td>243000</td>
</tr>
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<td>22000</td>
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<td>Parameter</td>
<td>Value</td>
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<tr>
<td>-------------------</td>
<td>---------------</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Buffalo</td>
<td>Twin Otter</td>
</tr>
<tr>
<td>$S, \text{ft}^2$</td>
<td>945</td>
<td>420</td>
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<tr>
<td>$T_{\text{static}}, \text{lbs}$ (1)</td>
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<td>5750</td>
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<tr>
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<td>12000</td>
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<tr>
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<tr>
<td>$\rho_0, \text{slugs/ft}^3$</td>
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<td>.002378</td>
</tr>
</tbody>
</table>

Notes:
1. From Appendix, this report.
2. Atmospheric density ratio calculated as $\sigma = e^{-h/32500}$ compares with standard atmosphere data as follows:

<table>
<thead>
<tr>
<th>$h$</th>
<th>standard</th>
<th>calculated</th>
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</thead>
<tbody>
<tr>
<td>0</td>
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<td>1</td>
</tr>
<tr>
<td>5000</td>
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<td>.858</td>
</tr>
<tr>
<td>10000</td>
<td>.738</td>
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<tr>
<td>15000</td>
<td>.629</td>
<td>.630</td>
</tr>
<tr>
<td>20000</td>
<td>.533</td>
<td>.540</td>
</tr>
</tbody>
</table>

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 displacement (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:

\[ \dot{U}, V, W, P, Q, R, \phi, \theta, \psi, \dot{\phi}, \dot{\theta}, \dot{\psi}, \]

\[ \psi, \theta, \phi, V_w, W_w, x_L, y_L, \alpha, \alpha, \beta \]

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

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

\[ \rho = \sigma \rho_0 \]

\[ q = \frac{1}{2} \rho V_R^2 \]

\[ U_w = V_R \]

\[ \ddot{x}_L = U = V_R + x_{wL} \]

\[ \ddot{y}_L = V = y_{wL} \]

\[ -\dot{h} = W = z_{wL} \]

\[ C_L = C_{L_0} = \frac{W}{qS} \]

\[ C_D = C_{Df} + C_L^2 / \pi eAR \]

\[ D = C_D qS \]

\[ \alpha_B = \alpha_{B_0} = C_{L_0} / \alpha + \alpha_{OL} \]

\[ \delta_e = 0 \]

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

\[ -22- \]
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 $\xi$, and TMDLE seconds for elevator input $\delta_e$. This scheme permits setting up an initial 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 $\theta$, angle of attack $\alpha$, altitude rate $\dot{h}$, and forward speed $U$ resulting from a $1^\circ$ step elevator input $\delta_e$ 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 $\beta$, roll rate $\dot{\phi}$, yaw rate $\dot{\psi}$, and yaw angle $\psi$ resulting from a $1^\circ$ step aileron input $\delta_a$. Figure 7 compares with Figure D7 of Reference 1.

Figure 8 shows the response in the same parameters resulting from a $1^\circ$ step rudder input $\delta_r$. This figure corresponds to Figure D8 of Reference 1.

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


TABLE I

SIMULATION PROGRAM LISTING

-27-
### TABLE I (Cont)

<table>
<thead>
<tr>
<th>NAMELIST</th>
<th>BLFREQ, PRNFOQ</th>
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<td>NAMELIST</td>
<td>GDELE, GDLA, GDLR, TNNPNT, SLPBL, PTCURHR, SELTRM</td>
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<tr>
<td>NAMELIST</td>
<td>XSS, YSS, ZSS, XSD, YSD, ZSD, XTAU, YTAU, ZTAU, JX, JY, JZ</td>
</tr>
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</table>

C

```fortran
CALL SETPST(32, 3000, 2000, 0000, 2000)
CALL SETPST(132, 3000, 2000, 0000, 2000)

C
12 NTYPE=0
C### CALL STANDBY
S   JX = 031010
C### 3 DEG/SEC/POINTER WIDTH FOR RATE OF TURN
   RNK = .19 + 1
C#### 4 DEG/BAR-WIDTH FOR SLIP
   SLPBL = 5.73
C#### 4 DEG/BAR-WIDTH FOR PITCH
   PTCURHR = 14.325
C#### 2 DEG/VOLT
   SELTRM = GDELE = 0349066
C#### 2 DEG/VOLT
   GDLR = 0116356
C#### 4 DEG/VOLT
   GDLA = 0232712
C
C### SET INITIAL CONDITIONS, IDLE L99P
C
### CALL COMPUTE
S   E55 = 031013
X   TEMP = BLFREQ / DEL
X   ITTH = TEMP
X   TEMP = PHTFREQ / DFL
X   ITTH = TEMP
X   IBLIP = IPRNE = 1
X   G8M = 32.2 / WEIGHT
D6 = 14.9
1   DELHV(1) = 0.
   SIG = EXP(-H/HATM)
   RH8 = SIG * RH0SEA
   CLO2 = WEIGHT / (RH8 * S * VR * VR)
   ALFRO = CLO2 / CLO / ALFROL
   TMETA = ALFRO
   IF (SENSESWITCH5) 20, 21
20 CONTINUE
   UVR = XSS
   V*YSS
   W*ZSS
```

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TABLE I (Cont)

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<th>GO TO 22</th>
</tr>
</thead>
<tbody>
<tr>
<td>21 CONTINUE</td>
</tr>
<tr>
<td>WIV=0.</td>
</tr>
<tr>
<td>U&gt;VR</td>
</tr>
<tr>
<td>22 CONTINUE</td>
</tr>
<tr>
<td>VVIV=0.</td>
</tr>
<tr>
<td>U&gt;VR</td>
</tr>
<tr>
<td>CO=CO+CL+CL+EL+TAN</td>
</tr>
<tr>
<td>DYN=5R+Rb+VR+VR</td>
</tr>
<tr>
<td>DRAG=CD+DYN+3</td>
</tr>
<tr>
<td>DRAI+DRAG</td>
</tr>
<tr>
<td>ALFI=ALF</td>
</tr>
<tr>
<td>VRI=VR</td>
</tr>
<tr>
<td>SIGT=SIG</td>
</tr>
<tr>
<td>T=DEL</td>
</tr>
<tr>
<td>RAN0=1+DEL+YTAU</td>
</tr>
<tr>
<td>RNA=1+DEL+YTAU</td>
</tr>
<tr>
<td>RNA=1+DEL+ZTAU</td>
</tr>
<tr>
<td>RNBD=XSD=SORT(2+DEL+YTAU)</td>
</tr>
<tr>
<td>RNA=ZSD=SORT(2+DEL+ZTAU)</td>
</tr>
<tr>
<td>SGUS=SORT(XSD+YSD+YSS)</td>
</tr>
<tr>
<td>CALL ARM(0)</td>
</tr>
<tr>
<td>CALL ENINT</td>
</tr>
<tr>
<td>CONNECT(40, AERD)</td>
</tr>
<tr>
<td>X IDLE COMP + TEST-CASE CALLS OF AERD</td>
</tr>
<tr>
<td>X 10 CONTINUE</td>
</tr>
<tr>
<td>X 11 CALL AERD</td>
</tr>
<tr>
<td>X IDLE LOOP + FAKE INTS00 USING Y/F/T/L/SDL, IF INT, SYSTEM D,A</td>
</tr>
<tr>
<td>X CONTINUE</td>
</tr>
<tr>
<td>X CALL AERD</td>
</tr>
<tr>
<td>X X IDLE LOOP</td>
</tr>
</tbody>
</table>
TABLE I (Cont)

C**** TIMING SIGNAL, SET F/F
S E04 030000
C RECTANGULAR INTEGRATION
T=T+DEL
DO 10 I=1/12
10 VINT(I)*VINT(I)+DERIV(I)*DEL
C ***** HERE
C**** DL= FROM -10 V DOWN TO +15 V UP
C**** DL= FROM -15 V RIGHT TO +15 V LEFT
C**** DL= FROM -30 V RIGHT TO +30 V LEFT
C**** EL= FROM -15 V DOWN TO +15 V UP
C**** THR= FROM -3.2 V IDLE TO 0 V FULL
CALC DDLZ(EDL=DLA*DLR DL=DLA*DLR
THR=THR+I.*SIG*(1.+CT1+VR1+CT2+VR1)/SIGI*STAT)
C TOTAL VELOCITY
VRSEC=UW+UW+VW+WV+WW
VR=SQRT(VRSEC)
C CALCULATE COEFFICIENTS
SIG=EXP(-H/HATM)
RHOG=RHOSE*SIG
IF(T=GE.HTHR)GB TO RA
IF(T=VEC+EQP)SIG=1
THR=THR+I.*SIG*(1.+CT1+VR1+CT2+VR1)/SIGI*STAT)
C DYNAMIC PRESSURE
DYNS=SIG*VRSEC
C SPH1=SI1(PHI1)
SPSI1=SI1(PSI)
STH=SI1(THETA)
CPS1=CO1(PHI)
CPSI=CO1(PSI)
CTH=CS1(THETA)
C BODY TO EARTH TRANS MATRIX
STC1=STC1(PSI)
SSC1=SSC1(PHI)
SSP1=SSP1(PSI)
BTE(1,1)=CTH*CPS1
BTE(1,2)=STC1*SPHI*SSCP
BTE(1,3)=STC1*CPS1*SSP1
BTE(2,1)=CTH*SPSI
BTE(2,2)=SSP1*STH*CPS1
BTE(2,3)=SCS1*STH*SPHI
BTE(3,1)=STH
BTE(3,2)=SPHI*CTH
BTE(3,3)=CPHI*CTH
C VEC=IND M9DEL
C F*S 212P 23 DOWN FOR Y рассматриным X Y Z RESP.
C S*S 5 SET FOR STEADY STATE
S SKS 233006
TABLE I (Cont)

S  BRU 11S
   CALL GUST(XGUS,RNAX,RNBX,JX)
   JX=2
S  BRU 12S
   XGUS=0*
S12 SKS 030007
S  BRU 13S
   CALL GUST(YGUS,RNAY,RNBY,ITY)
   ITY=2
S  BRU 14S
   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)=475
   CONTINUE
   WXX*YSS+(XGUS*YSS-YGUS*YSS)/SGUST
   WYY*YSS+(XGUS*YSS+YGUS*YSS)/SGUST
   WZZ*ZZS+ZGUS
   GO TO 6
5 WXX*XGUS
   WYY*YGUS
   WZZ*ZGUS
   CONTINUE

C *** BODY AXIS VELOCITIES INCLUDING WINDS
   UH=-(WXX*BT(1,1)+WYY*BT(1,2)-WZZ*BT(3,1))
   VM=v-(WXX*BT(1,2)+WYY*BT(2,2)-WZZ*BT(3,2))
   WM=-((WXX*BT(1,3)+WYY*BT(2,3)-WZZ*BT(3,3))
C ANGLE OF ATTACK, LIFT, DRAG
   ALF=ATAN2(WW, UW)
   CL=ALF*CLD
   CD=CF*CL*CL*EPARI
   QS=DYN+G
   LIFT=CL*QS
   DRAG=CD*QS

C SIDESLIP
   ETA=ATAN2(WV, UK)
C RV*S=RVS/2 RVS*B=RVS/2 RV*S=RVS/4 RV*S=RVS/4
   RV*S=RHV*VR*S
   YV=RVS*CV
   RV*S=RVS* Cy
   RV*S=RVS*CV
   YR=RVS*CV
   YP=RVS*CV
   LV=RVS*CLB
   CLH=CLF*H*25*CL
   RVBR=RVR*B
   LR=RVR*B
   LP=RVR*B
   CLDA*QS+BS*CLDA
   NV=RVS*CN8
   CNR=CNRF*H*25*(CD=FLCC)
   NR=RVR*B
   CNP=CNPF*H*25*CL+(1.-A*EPARI
   NP=RVR*B
   ND=RVR*B
C CALCULATE AERODYNAMIC FORCES

-31-
TABLE I (Cont)

CALF = COS(ALF)
SALF = SIN(ALF)

C+++++LONGITUDINAL EQUATIONS
C+++++IF NTYPF=2 FLY ONLY LATERAL
IF (NTYPE = 2) GO TO 31

UDOT = WDOT = 0

C 31 X FORCE
UDOT = 0

C 32 Y FORCE

C 33 ALFA

C+++++LATERAL EQUATIONS
C+++++IF NTYPF=1 FLY ONLY LONGITUDINAL
IF (NTYPE = 1) GO TO 32

C+++++MOMENT

C+++++EULER ANGLE TRANSFORMATION

C+++++X, Y, Z FORCE IN EARTH-FIXED COORDINATES

C+++++BLIPS FOR TIME ON STRIP CHART RECORDER
TABLE I (Cont)

D5=PHI=57.3
D6=ALF=143.25
D7=DLA=114.6

C

* NTYPE = 0 FOR COUPLED, 1 FOR LONGITUDINAL ONLY, 2 FOR LATERAL ONLY.

C

IF (NTYPE=1166,70,65)

70 D5=HDBT
D6=PHI=57.3
D7=ALF=143.25
GO TO 66

65 D1=PHI=57.3
D2=BETA=286.5
D3=DLR=57.3
D4=PSI=286.5
CONTINUE

66 CONTINUE

CALL DAL(20,D1,D2,D3,D4,D5,D6,D7,D8)

72 CONTINUE

TURNT=TRNPNT=7.5*PSI*HDBT

SLTPR=15*SLPBL*BETA

C

* HDBT IN FT/SEC, INDICATOR IN FT/MIN

C

-6.67 VELTS/1000 CCMB, +8.33 VOLTS/1000 DESCEND

C

IF (HDBT.GT.0.)R6FC = +HDBT
IF (HDBT.LE.0.)R6FC = +998*HDBT
C

VR IN FT/SEC, CONVERT TO 'RHO'S

VKTS=VR**59*SQRT(RHO/RH0SEA)

C

IF (VKTS=175.1)*200,210,210
200 IF (VKTS=125.1)*209,209,209

201 IF (VKTS=44.1)*208,208,208

202 AIRSPD=45
GO TO 211

208 AIRSPD= *22*(VKTS=44.1)
GO TO 211

209 AIRSPD= 17+8+092*(VKTS=125.1)
GO TO 211

210 AIRSPD= 22+4+092*(VKTS=175.1)

211 CONTINUE

ALT**02+H0
DIRGYR=PSI=916667
I=DIRGHR=+01
DIRGHR=DIRGYR=+100
ROLL=59.5*PHI
IF (ABS(ROLL).GT.99.9)*ROLL=SIGN(99.9*ROLL)
PITCH=5.*PITCHR*(THETA+ALF80)
RPM=3.+THRST+27.

C

D7A HERF

C

IF (PRNT=0) RETURN

C

C..... T-OUT AT FREQUENCY PNFREQ
X

IF (PRNT=0) RETURN TO 62

X

10 PRND=0

X

IF (GSSWCH=I4)61,62

X

61 WRITE (10,H1011,T,U,V,W,VR,H,HDBT,ALF=57.3,THETA=57.3,PHI=57.3,
X

THRT=THRST=THRST+LIFT,DRAG=DLA=57.3,3,BETA=57.3,
X

101 FORMA=423,T=49.18/23X*UDOT+F11+2*X*8U1+8F11+12X6V1+F11+4

X


X

2 100K5+ALF4=11.4*8*X*THETA+F11+4*X*PHI+F11+44*XPSI+F11+F11+47X*THRST+4,

X

20 F11+412X68+12X80+G11+412X88+F11+47X*THRST+4,

-33-
| X | 4 F11+4/2X,THRT=F11+2,X*LIFT=F11+9X,DRAG=F11+10X |
| X | 5 DRC=F11+212X,DUR=F11+54X,DRL=F11+211X,BETA=F11+4/2X |
| X | 6 DRC=F11+4/2X |

**CONTINUE**

**C**

**C**** TIMING SIGNAL, RESET FF**

**S**

**END**

```
RETURN
END
```
DHC-8 TWIN OTTER

Announced in August 1984, the Twin Otter is a STOL transport powered by two Pratt & Whitney (IFAO) PT6A-20 turboprop engines. Design work began in January 1984. 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, 1986.

At the beginning of 1987, a total of 27 Twin Otters had been delivered or were on order, with options for 11 more. They included eight for the Chuban Air Force, two for Trans-Australian Airlines, one for the Canadian Department of Lands and Forests, four for Pan American Airways, and others for Viking Airlines and Air Wisconsin, USA. Production was scheduled to be at the rate of six a month through 1987.

Under development for delivery in 1988 is a version of the Twin Otter with more powerful (840 eshp) Pratt & Whitney PT6A-27 turboprop engines, longer nose to provide more baggage space, and AVG of 12,500 lb (5,670 kg). The following data refer to the current production model.

**Type**: Twin-turboprop STOL transport.

**Wings**: High-wing monoplane, with single streamliner-section bracing struts on each side. Wing section NACA 64 series mean line, NACA 6010 (modified) thickness distribution. Aspect ratio 10, constant chord of 8 ft 8 in (2.64 m). Inclined 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-de-icing equipment optional.

**Fuselage**: Conventional all-metal semi-monocoque safe-life structure.

**Tail Unit**: Cantilever all-metal structure of high strength aluminium alloys. Pin integral with fuselage. Fixed incidence tailplane. Trims in rudder and port elevator, latter interconnected with flaps. Pneumatic de-icing boots on tailplane leading edge optional.


**Power Plant**: Two 259 eshp Pratt & Whitney (IFAO) PT6A-20 turboprop engines, each driving a Hamilton three-blade reversible-pitch fully-feathering metal propellers, diameter 8 ft 6 in (2.49 m). Fuel in two tanks (steel) under cabin floor; total capacity 919 Imp. gallons (4,318 litres). Two refueling points on port side of fuselage; fuel capacity 2 Imp. gallons (9 litres) per engine. Electric de-icing system for propellers and air-intakes optional.

**Accommodation**: Two seats side-by-side on flight deck. Seats for 12-18 passengers in main cabin. Cabin divided by bulkhead into main passenger or freight compartment and baggage or toilet compartments. Door on each side of main cabin, at rear. Baggage compartments in nose and all of cabin, each with upward-hinged door on port side.

**Reverses**: Hydraulic system, pressure 1,000 lb sq in (6904 kgf/m²) for flaps, brakes and nose-wheel steering. No hydraulic system. One 200A starter-generator on each engine.


| Dimensions, External | | |
|----------------------|------------------|
| Wing span | 45 ft 6 in (13.82 m) |
| Length overall | 49 ft 6 in (14.96 m) |
| Height overall | 18 ft 6 in (5.68 m) |
| Tailplane span | 21 ft 6 in (6.55 m) |
| Wheel track | 12 ft 6 in (3.81 m) |
| Wheelbase | 14 ft 9 in (4.50 m) |
| Passenger door (port side) | Height | 4 ft 2 in (1.27 m) |
| Width | 1 ft 6 in (0.46 m) |
| Height to sill | 2 ft 6 in (0.76 m) |
| Passenger door (starboard side) | Height | 3 ft 5 in (1.05 m) |

**Accommodation**: Total capacity 12-18 passengers, 30 passengers with extra seats fitted. All-purpouse cabin with extensions for a variety of operations. Classified as a STOL aircraft (low operating speed).

**Performance (at max T.O. weight)**:
- Max cruising speed at 10,000 ft (3,000 m) 181 mph (291 km/h)
- Max cruising speed at 10,000 ft (3,000 m) 181 mph (291 km/h)
- Max landing speed (100% power) 150 mph (241 km/h)
- Max take-off speed 150 mph (241 km/h)
- Rate of climb at S/L 1,550 ft (472 m) min
- Service ceiling 22,000 ft (6,706 m)
- Service ceiling, one engine out 12,000 ft (3,658 m)
- T.O. to 50 ft (15 m): 1,120 ft (341 m)
- STOL: 1,700 ft (518 m)
- Landing from 50 ft (15 m): 1,700 ft (518 m)
- CAR for 3 2,100 ft (635 m)
- Range with max fuel, 30 min reserve 920 miles (1,480 km)

**Cabin Dimensions, Internal**

- Cabin length | 45 ft 6 in (13.82 m)
- Cabin width | 18 ft 6 in (5.68 m)
- Cabin height | 10 ft 6 in (3.25 m)
- Cabin volume | 745 cu ft (21.10 m³)
- Baggage compartments (nose) volume | 22 cu ft (0.62 m³)
- Baggage compartment (rear) volume | 32 cu ft (0.91 m³)

**Takeoff**: 7,000 lb (3,175 kg).

**Cabin Configuration**: Three-class with 18 passengers. First class with air conditioning, reclining seats, and individual air vents. Business class with variable seats, also with air conditioning. Coach class with fixed seats, also with air conditioning. Overall length 45 ft 6 in (13.82 m).

**Weight**

- Empty weight | 7,000 lb (3,175 kg)
- Max T.O. weight | 11,000 lb (4,989 kg)
- Max landing weight | 9,500 lb (4,309 kg)

**Fuel Capacity**

- Fuel tank capacity | 919 Imp. gallons (4,318 litres)
- Refueling points | Two
- Refueling point capacity | 2 Imp. gallons (9 litres) per engine

**Engine**

- Engine type | Pratt & Whitney (IFAO) PT6A-20 turboprop
- Max thrust | 259 eshp (191 kW) each

**Power Plant**

- Powerplant configuration | Two engines
- Fuel type | Jet-A

**Dimensions, External**

- Length overall | 49 ft 6 in (14.96 m)
- Height overall | 18 ft 6 in (5.68 m)
- Wheelbase | 14 ft 9 in (4.50 m)

**Passenger Door (Port Side)**

- Height | 4 ft 2 in (1.27 m)
- Width | 1 ft 6 in (0.46 m)
- Height to sill | 2 ft 6 in (0.76 m)

**Passenger Door (Starboard Side)**

- Height | 3 ft 5 in (1.05 m)

**Source**: Reference 3

**FIGURE 1**

-35-
**DHC-5 BUCKAUO FO**

**Differences between the US and Canadian versions are as follows:**

**CV-7A**: US model, with 2,500 shp General Electric T64-GE-10 turbojets. Overall length 77 ft 6 in (23.67 m). Designation may be changed following transfer of responsibility for aircraft in this category from US Army to USAF.

**C-11A**: Canadian Defence Force model, with 3,000 shp General Electric T58B turbojets. Overall length 79 ft 0 in (24.05 m). Otherwise similar to CV-7A, with only small differences in performance.

**Wings**: Cantiler high-wing monoplane. Wing sections NACA 64A417-2 (mod at root, NACA 63A416 (mod at tip, Aspect ratio 9.75. Chord 11 ft 3 in (3.49 m) at root, 5 ft 11 in (1.80 m) at tip. Dihedral 6°, noseboard of carrello. 6°, outward. Incidence 3° 20'. Sweepback at quarter chord 1° 45'. Conventional fuselage multi-spar structure of high-strength aluminum alloys. Full span double-slotted aluminum alloy flaps, outward sections functioning as ailerons. Aluminum alloy slot-type spools, located at inboard flaps, are actuated by Jerry Hydraulics unit. Slippers coupled to manually-operated ailerons for lateral control, uncoupled for symmetric ground operation. Electrically-controlled trim-tab in starboard aileron. Leading edge tab on each aileron. Rudder-alternate interconnect tab on port aileron. Outer wing leading-edge fitted with electrically-controlled flush pneumatic rubber de-icer boots.

**Fuselage**: Fuselage structure of high-strength aluminum alloy. Cargo door supported by longitudinal keel members.

**Tail Unit**: Cantiler structure of high-strength aluminum alloy, with fixed incidence tailplane mounted at tip of fin. Elevator aerodynamic and mass-balanced. Flap and trailing-edge hinged rudders are powered by tandem jacks operated by two independent hydraulic systems manufactured by Jerry 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. Jerry Hydraulics also-pneumatic shock-absorbers. Goodrich main wheels and tires, size 37 x 13.60 x 6, pressure 14.5 lb/sq in (101 kgf). Goodrich nose wheels and tires, size 9 x 12.50, pressure 35 lb/sq in (241 kgf). Goodrich multi-disc brakes.

**POWER PLANT**: Two General Electric T64 turbo-props engines details under entries for individual versions, above, each driving a Hamilton Standard 3010-2-13 three-blade propeller, diameter 14 ft 6 in (4.42 m). Fuel one integral tank in each outer wing, capacity 335 Imp gallons (2,423 litres) and rubber bags tanks in each outer wing, capacity 335 Imp gallons (1,327 litres). Total fuel capacity 1,280 Imp gallons (5,100 litres). Refueling points above wings and on sides of fuselage for pressure refueling. Total oil capacity 120 Imp gallons (450 litres).

**DIMENSIONS, EXTERNAL**

| Wing span | 98 ft 6 in (30.05 m) |
| Length overall: | CV-7A | 77 ft 6 in (23.67 m) |
| CC-11A | 79 ft 0 in (24.06 m) |
| Height overall | 38 ft 0 in (11.60 m) |
| Tailplane span | 31 ft 0 in (9.45 m) |
| Wheel track | 39 ft 0 in (11.90 m) |
| Wheelbase | 21 ft 11 in (6.60 m) |
| Cabin doors (each side): | Height | 5 ft 6 in (1.68 m) |
| Width | 2 ft 0 in (0.60 m) |
| Height to sill | 3 ft 10 in (1.17 m) |
| Emergency exits (each side, below wing leading-edges): | Height | 3 ft 4 in (1.02 m) |
| Width | 2 ft 6 in (0.76 m) |
| Height to sill above | 3 ft 10 in (1.17 m) |
| Rear cargo loading door and ramps: | Height | 20 ft 0 in (6.10 m) |
| Width | 10 ft 0 in (3.05 m) |
| Height to ramp hinge | 3 ft 10 in (1.17 m) |

**DIMENSIONS, INTERNAL**

| Cabin, excluding flight deck: | Length, cargo floor | 31 ft 0 in (9.45 m) |
| Max width | 8 ft 0 in (2.44 m) |
| Max height | 6 ft 6 in (1.98 m) |
| Floor area | 143 sq ft (13.23 m²) |
| Volume | 1,200 cu ft (34.00 m³) |

**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 | 21,657 lb (9,834 kg) |
| Max payload | 13,843 lb (6,279 kg) |
| Max T-O weight | 41,000 lb (18,600 kg) |
| Max zero-fuel weight | 37,000 lb (16,783 kg) |
| Max landing weight | 36,000 lb (16,327 kg) |
| Max wing loading | 42.4 lb/sq ft (212 kgf/m²) |
| Max power loading | 7.2 lb/hp (3.27 kgf/kw) |

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

| Max level speed at 10,000 ft (3,000 m) | 271 mph (436 km/h) |
| Max permissible diving speed | 316 mph (507 km/h) |
| Max cruising speed at 10,000 ft (3,000 m) | 271 mph (436 km/h) |
| Econ cruising speed at 10,000 ft (3,000 m) | 268 mph (435 km/h) |
| Stall speed, 45° flaps at 10,000 ft (3,000 m) | 271 mph (436 km/h) |
| Stall speed, flaps up at max A.W. | 271 mph (436 km/h) |
| Rate of climb at 8/L | 1,800 ft (550 m) |
| Service ceiling | 30,000 ft (9,140 m) |
| Service ceiling, one engine out | 14,300 ft (4,350 m) |
| T-O on firm dry sod | 1,040 ft (317 m) |
| T-O to 30 ft (15 m) from firm dry sod | 1,240 ft (380 m) |
| Landing from 50 ft (15 m) on firm dry sod | 1,150 ft (350 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

\[ \psi = \text{ROTATION ABOUT } Z_L \text{ AXIS} \]
\[ \theta = \text{ROTATION ABOUT } Y_C \text{ AXIS} \]
\[ \phi = \text{ROTATION ABOUT } X_A \text{ AXIS} \]

Figure 3: Reference Coordinate Frames
(a) At equilibrium

(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 $1^\circ$ 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 $\eta_p$ is the propeller efficiency. Power supplied to the propeller is expressed in this report as

$$ P = \sigma P_o \xi $$

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

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

![Figure 3-17. Propeller efficiency (unconstrained).](image-url)
as a function of advance ratio $J$ and power coefficient $C_p$. By definition,

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

and

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

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 $n_p = .79$. 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

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

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

- A-2 -
\[ T_{\text{static}} = 22400 \text{ lbs} \]
\[ C_{T_1} = 0.00370 \text{ fps}^{-1} \]
\[ C_{T_2} = 6.51 \times 10^{-6} \text{ fps}^{-2} \]

The process is repeated for the "Twin Otter" (Figure 1). For this aircraft (with two PT6A-20 engines), \( N = 2200 \text{ rpm}, D = 8.5 \text{ ft}, \) and \( P_0 = 652 \text{ ESHP/engine}. \) The required constants are established as:

\[ T_{\text{static}} = 5750 \text{ lbs} \]
\[ C_{T_1} = 0.00378 \text{ fps}^{-1} \]
\[ C_{T_2} = 9.07 \times 10^{-6} \text{ fps}^{-2} \]

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