SIMULATION MODEL FOR THE PIPER PA-30 LIGHT MANEUVERABLE AIRCRAFT IN THE FINAL APPROACH

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TECHNICAL MEMORANDUM

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This report describes the Piper PA-30 "Twin Comanche" aircraft and a representative autopilot during the final approach configuration for simulation purposes. The aircraft is modeled by linearized six-degrees-of-freedom perturbation equations referenced to the aircraft stability axis. Other equations are presented which derive the body axis rates, velocities and accelerations, and ground referenced velocities (translation equations).

The autopilot is a representative system for automatic ILS approaches from initial localizer track down to decision height. The glideslope system is engaged by approaching the glidepath at constant altitude (usually in the altitude hold mode) on the localizer beam. The pilot must take over manually at the decision height since light aircraft are not normally equipped with automatic flare capability.

The aircraft autopilot model described herein has been used extensively in simulation studies at TSC and exhibits the expected behavior.
The contents of this report reflect the views of the Federal Aviation Administration which is responsible for the facts and the accuracy of the data presented herein. The contents do not necessarily reflect the official views or policy of the Department of Transportation. This report does not constitute a standard, specification or regulation.
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<td>nondimensional drag stability derivative</td>
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<tr>
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<td>$m$</td>
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<td>$\bar{q}$</td>
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<td>$r$</td>
<td>yawing angular rate of aircraft (about stability axis when no subscript)</td>
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<tr>
<td>r.p.m.</td>
<td>engine revolutions per minute</td>
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<td>wing area</td>
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<td>$T$</td>
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<td>$\Delta T$</td>
<td>change in thrust due to pilot throttle input</td>
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<td>$u()$</td>
<td>perturbed forward velocity of aircraft (along stability x-axis when no subscript)</td>
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</tr>
<tr>
<td>$U_0$</td>
<td>equilibrium or reference forward velocity of aircraft</td>
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<tr>
<td>$V()$</td>
<td>perturbed side velocity of aircraft (along stability Y-axis when no subscript)</td>
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</tr>
<tr>
<td>$V_T$</td>
<td>total velocity of aircraft</td>
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- $\text{ft/sec}^2$
- $\text{ft}$
- $\text{slug-ft}^2$
- slugs
- in. of Hg
- rad/sec
- lbs/ft$^2$
- rad/sec
- -
- ft$^2$
- lbs.
- lbs.
- ft/sec
- ft/sec
- knots
$w()$ 
perturbed normal velocity of aircraft 
( along stability Z axis when no subscript) 

$X_{\text{accel}}$ 
distance from center of gravity to 
accelerometer location measured along 
$X$ fuselage axis, positive forward 

$X_E$ 
$X$-axis in local vertical coordinate frame 

$Y_E$ 
$Y$-axis in local vertical coordinate frame 

$Z_E$ 
$Z$-axis in local vertical coordinate frame 

$Z_T$ 
pitching moment arm of the thrust vector 
positive downward 

$\alpha$ 
angle of attack $= \tan^{-1}(w/u)$ 
rad. 

$\alpha_r$ 
angle between $X$-stability axis and 
$X$-fuselage axis 
rad. 

$\alpha_T$ 
angle between $X$-stability axis and 
thrust axis 
rad. 

$\beta$ 
angle of sideslip $= \sin^{-1}(v/V_T)$ 
rad. 

$\epsilon$ 
glideslope error, localizer error 
rad. 

$\delta_t$ 
deflection of throttle position 
($\delta_t = 1$ = full throttle) 

$\delta()$ 
deflection of control surface 
rad. 

$\theta, \phi, \psi$ 
Euler angles referenced to stability axis 
rad. 

$\rho$ 
atmospheric air density 
slug/ft$^3$ 

$({}^\prime)$ 
derivative with respect to time 

Subscripts 

a 
aileron 

B 
fuselage reference frame 

E 
local vertical coordinate frame 

e 
elevator or stabilizer 

f 
flaps
c  equilibrium or reference condition
r  rudder
s  aircraft stability coordinate frame
t  throttle

$u, \alpha, q, \phi, r, \dot{\psi} \in_{\alpha', \psi}'$ as defined above
1.0 INTRODUCTION

The primary objective of this research effort is to derive a light, maneuverable aircraft-autopilot model as one extreme of aircraft type for final approach simulation studies. This model is to serve as a high priority vehicle in two FAA projects: developing requirements for a Scanning Beam Microwave Instrument Landing System and developing an all-encompassing generalized set of equations of an aircraft during approach and landing for all-weather landing system studies on the NAPEC hybrid computation facility.

In general, no such model existed for a common and representative light aircraft at the beginning of this effort because light aircraft are not generally designed by analysis and simulation. Similarly, light aircraft generally do not have automatic landing systems as standard equipment.

The definition and identification of a light maneuverable aircraft is treated in Reference (1). The Piper PA-30 "Twin Comanche" is selected as the light maneuverable aircraft for modeling primarily because of the availability of data from wind tunnel and actual flight tests and because of an existing, partially useful, simulation model at the NASA Edwards Flight Research Center. The final aircraft model is derived from this simulation model and NASA TN D4983. The flight condition is based upon a high wind environment (headwind and sidewind of approximately 24 feet per second). This condition was selected to represent an extreme case for the simulation studies.

The autopilot description is based on a report (Reference 6) prepared for TSC by Dr. Kohlman of the University of Kansas. The final configuration (i.e., gain values, gain scheduling and logic) was determined by simulation at TSC.
2.0 VEHICLE DESCRIPTION

The Piper PA-30 is a light twin engine low-wing monoplane. Figure 1 gives the principal dimensions. The airplane has a wing span of 35.98 ft., a wing area of 178 ft², an aspect ratio of 7.3, and a mean aerodynamic cord of 5 feet based on projection of the outboard leading edge of the wing through the fuselage. The wing airfoil section is a modified NACA64\textsubscript{2}A215 airfoil with the trailing-edge cusp faired out. The wing has 5° of dihedral with no twist and is at 2° positive incidence with respect to the fuselage reference line. The airplane has the standard three-control system. The horizontal tail is of the all-movable type with a control deflection range of 4° to -14°. The tail has a trailing-edge tab which moves in the same direction as the tail with a deflection ratio (tab deflection to tail deflection) of 1.5. The control deflection range on each aileron is from 14° to -18°. The rudder control deflection range is ±27°.

2.1 Vehicle Model

A final approach model is presented for the Piper PA-30 aircraft based on data available from the NASA Edwards Flight Research Center simulation model and NASA TND 4983. The model consists of rigid body, six-degree-of-freedom aircraft equations of motion which are basically linear perturbation equations in the stability axis system (some of the nonlinear cross coupling terms have been included).

2.2 Axis Systems

The stability axis frame (s) is depicted in Figure 2. The definition of airplane angles and sign convention is described in Figure 3. The stability axis is fixed to the aircraft and rotates and translates with the aircraft. Its origin is the center of mass of the aircraft. The X-axis is in the direction of motion of the airplane in a reference condition of steady
All dimensions are in feet (meters).

Figure 1. Piper Dimensional Data
Figure 2. System of Axes and Positive Sense of Angles, Forces and Moments
Figure 3. Definition of Airplane Angles and Sign Convention
symmetric flight. The Y-axis is normal to the aircraft's plane of symmetry (positive to the right), and the Z-axis is in the plane of symmetry (positive downward) and orthogonal to the X- and Z-axes. The aerodynamic stability derivatives are all referenced to this axis system.

A fuselage referenced body coordinate frame (B), is defined for determining the angular rates and velocities in body axis, and the aircraft normal acceleration. This axis system is similar to the stability axis system except that the X-axis is directed along the fuselage. The angle $\alpha$ relates the two axis systems. A local vertical coordinate frame (E) is defined for determining the velocities of the aircraft with respect to the air mass. This coordinate frame has its origin at the center of mass of the aircraft with the X-axis pointing North, and Y-axis pointing East, and the Z-axis pointing down. The velocities from this coordinate frame can be converted to ground velocity by adding the various components of the steady wind.

2.3 Assumptions in Using Aircraft Equations

The derivation of the aircraft equations involved the following 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.
5. The aircraft is initially in equilibrium flight with no linear or angular accelerations, no angular rates, and no initial roll angle or lateral velocity.
6. Small disturbance (perturbation) theory is used. Motions and forces are referred to the equilibrium flight condition.
7. Hinge moments are insignificant.
2.4 Equations of Motion

Drag Equation

\[ \frac{m}{qS} \ddot{u} + \frac{2C_D}{U_o} u - \left( C_L - C_D \right) \dot{u} + \frac{mg}{qS} \dot{\gamma} - \frac{mU_o}{qS} \dot{\theta} = \left( \frac{\cos \alpha}{qS} \right) \Delta T \]

Lift Equation

\[ \frac{2C_L}{U_o} u + \left( \frac{mU_o}{qS} + \frac{c}{2U_o} C_L \right) \dot{\alpha} + \left( C_L + C_D \right) \alpha + \left( \frac{c}{2U_o} C_L \right) \cdot \frac{mU_o}{qS} \] \[ \dot{\theta} + \left( \frac{mg \theta}{qS} \right) \dot{\theta} + \frac{mU_o}{qS} \delta \dot{\phi} - \frac{m}{qS} \delta \dot{q} = - C_L \delta e - \left( \frac{\sin \alpha}{qS} \right) \Delta T \]

Pitching Moment Equation

\[ - \frac{C_m}{U_o} u - \frac{c}{2U_o} C_m \dot{\alpha} - C_m \alpha + \frac{I_{yy}}{qSc} \dot{\phi} - \frac{c}{2U_o} C_m \dot{q} = C_m \delta e + \frac{z_T}{qSc} \Delta T \]

Sideforce Equation

\[ \frac{mU_o}{qS} \dot{\delta} - C_Y \beta - \frac{b}{2U_o} C_Y \dot{\phi} - \frac{mg}{qS} \phi + \frac{mU_o}{qS} r - \frac{b}{2U_o} C_Y r + \frac{m}{qS} \delta u = \] \[ C_Y \delta r + C_Y \delta a \]

Rolling Moment Equation

\[ - C_{n \beta} + \frac{I_{xz_s}}{qSb} \dot{\phi} - \frac{b}{2U_o} C_{n \beta} \dot{p} - \frac{I_{xz_s}}{qSb} \dot{r} - \frac{b}{2U_o} C_{n \beta} \dot{r} = C_{n \beta} \dot{r} + C_{n \beta} \delta a \]

Yawing Moment Equation

\[ - C_{n \beta} + \frac{I_{xz_s}}{qSb} \dot{\phi} - \frac{b}{2U_o} C_{n \beta} \dot{p} + \frac{I_{xz_s}}{qSb} \dot{r} - \frac{b}{2U_o} C_{n \beta} \dot{r} = C_{n \beta} \dot{r} + C_{n \beta} \delta a \]
Body Axis Rates and Velocities

\[ q_B = q \]
\[ p_B = p \cos(\alpha_r) - r \sin(\alpha_r) \]
\[ r_B = r \cos(\alpha_r) + p \sin(\alpha_r) \]
\[ u_B = (U_o + u) \cos(\alpha_r) - U_o \alpha \sin(\alpha_r) \]
\[ w_B = U_o \alpha \cos(\alpha_r) + (U_o + u) \sin(\alpha_r) \]
\[ v_B = U_o \beta \]

Accelerations

Vertical acceleration at center of gravity
\[ \dot{w}_B = U_o \dot{\alpha} \cos(\alpha_r) + \dot{u} \sin(\alpha_r) \]

Vertical acceleration at accelerometer
\[ \dot{w}_{\text{accel}} = \dot{w}_B - x_{\text{accel}} \dot{q} \]

Measured Normal Acceleration
\[ a_n = \dot{w}_{\text{accel}} + p_B v_B - q_B U_B \]

Euler Angle Rates
\[ \dot{\phi} = \dot{q} \cos \theta - r \sin \theta \]
\[ \dot{\theta} = \dot{p} + (q \sin \phi + r \cos \phi) \tan \theta \]
\[ \dot{\psi} = (q \sin \phi + r \cos \phi) \sec \theta \]
Translation Equations With Respect to Moving Frame

\[
\begin{align*}
X_E &= (U_0 + u) \cos \theta \cos \psi + U_0 \beta (\sin \phi \sin \theta \cos \psi - \cos \phi \\
&\quad \sin \psi + U_0 \alpha (\cos \phi \sin \theta \cos \psi + \sin \phi \sin \psi) \\
Y_E &= (U_0 + u) \cos \theta \sin \psi + U_0 \beta (\sin \phi \sin \theta \sin \psi + \cos \psi) \\
&\quad \cos \phi + U_0 \alpha (\cos \phi \sin \theta \sin \psi - \sin \phi \cos \psi) \\
Z_E &= (U_0 + u)(-\sin \theta) + U_0 \beta \sin \phi \sin \phi + U_0 \alpha \cos \phi \cos \phi.
\end{align*}
\]

2.5 Vehicle Data

Geometry

\begin{align*}
b &= 35.98 \text{ ft.} \\
c &= 5. \text{ ft.} \\
S &= 178. \text{ ft}^2 \\
Z_T &= -.75 \text{ ft.} \\
\alpha_T &= 0 \text{ deg.}
\end{align*}

Weight and Inertias

\begin{align*}
m &= 111.9 \text{ slugs} \\
I_{xxB} &= 2800 \text{ slug-ft}^2 \\
I_{xxS} &= 2801.7 \text{ slug-ft}^2 \\
I_{yyB} &= 1900 \text{ slug-ft}^2 \\
I_{yyS} &= 1900 \text{ slug-ft}^2 \\
I_{zzB} &= 4500 \text{ slug-ft}^2 \\
I_{zzS} &= 4513.7 \text{ slug-ft}^2 \\
I_{xzB} &= 80 \text{ slug-ft}^2 \\
I_{xzS} &= -7.9 \text{ slug-ft}^2
\end{align*}

Trim Flight Condition - Final Approach (High Wind Environment)

\begin{align*}
U_0 &= 176. \text{ ft/sec} \\
C_{LO} &= .55 \\
C_{DO} &= .034 \\
\beta_0 &= 0^\circ \\
\gamma_0 &= 0^\circ \\
\delta_e \text{ (trim elevator setting)} &= 0.4^\circ \\
\delta_f \text{ (trim flap setting)} &= 0.0^\circ \\
o &= 0.002378 \text{ slugs/ft}^3 \\
\bar{q} &= 36.8 \text{ lbs/ft}^2 \\
\alpha_x &= 0.0515 \text{ rad (2.95°)} \\
\alpha_T \text{ (thrust coeff.)} &= 0.034 \\
\text{center of gravity at 10% MAC} \\
\text{gear down}
\end{align*}
Nor-Dimensional Derivatives – Final Approach

\[
\begin{align*}
C_D = 0. & \quad C_L = 0. & \quad C_m = 0. \\
C_D = .275 & \quad C_L = 5.04 & \quad C_m = -1.147 \\
C_D = 0. & \quad C_L = 5.3 & \quad C_m = -14.55 \\
C_D = 0. & \quad C_L = 9.12 & \quad C_m = -25.0 \\
C_D = 0. & \quad C_L = 1.05 & \quad C_m = -2.87 \\
C_D = -.086 & \quad C_n = .0756 & \quad C_y = -.494 \\
C_D = .11 & \quad C_n = -.16 & \quad C_y = 0. \\
C_D = -.50 & \quad C_n = -.063 & \quad C_y = 0. \\
C_D = .01147 & \quad C_n = -.0573 & \quad C_y = .143 \\
C_D = -.0803 & \quad C_n = .00573 & \quad C_y = -.00916
\end{align*}
\]

2.6 Control Wheel and Pedal Characteristics

Aileron: Gearing constant at zero control deflection, 0.80 radian/ft.; wheel deflection, ±90°; max. force at end of wheel, ±15 lb.

Elevator: Gearing constant at zero control deflection, 0.42 radian/ft.; wheel throw, 4 inches forward, 5 inches aft; force differential 40 lbs. (detent position is 4 inches from firewall)

Rudder: Gearing constant at zero control deflection, 0.93 radian/ft.; pedal deflection, 4 1/2 inches; maximum force at full deflection, 120 lbs.
3. ENGINE THRUST DATA

Two 160 h.p. Lycoming IO-320-B four-cylinder air-cooled engines power the PA-30. Figure 4 presents the maximum available manifold pressure from each engine as a function of standard temperature altitude. This curve should be mechanized so that the pilot's indicator shows the maximum value as a function of altitude at full throttle.

Figure 5 presents the power available per engine as a function of r.p.m., MAP, and altitude. This curve should be mechanized so that power can be obtained from the r.p.m., MAP, and altitude.

Once the power is obtained, thrust can be calculated from the following equation:

\[
T \text{ (lbs.)} = \frac{325 \ N_p \ BHP}{V_T \text{ (knots)}}
\]

where \( N_p \) (power efficiency) = 0.74

Alternatively, a simplified algebraic expression for BHP has been derived from Figure 5 for the final approach configuration and can be used in place of Figure 5. The expression is given by

\[
BHP = 0.0024(h) + 0.0028(r.p.m.) \left[ \frac{29.2 - 0.000989(h)}{\delta_t} \right] - 8.0
\]

where \( h \) = altitude in feet
\( \delta_t = 1 = \text{full throttle} \)

An engine lag of about 0.1 second can be used in completing the engine transfer function (i.e. throttle+thrust).
Figure 4. Maximum Available Manifold Pressure
TO FIND ACTUAL HORSEPOWER FROM ALTITUDE, r.p.m., AND MANIFOLD PRESSURE:
1. LOCATE A ON FULL THROTTLE ALTITUDE CURVE FOR GIVEN r.p.m. MANIFOLD PRESSURE.
2. LOCATE B ON SEA LEVEL CURVE FOR r.p.m. & MANIFOLD PRESSURE & TRANSFER TO C.
3. CONNECT A & C BY STRAIGHT LINE AND READ HORSEPOWER AT GIVEN ALTITUDE D.

Figure 5. Power Characteristics Per Engine
4. AUTOPILOT

This section describes a representative PA-30 three-axis autopilot with stabilization system and with an automatic ILS approach mode available down to decision height.

Block diagrams of the longitudinal and lateral axis are shown in Figures 6 and 7 respectively. Damping is provided about all three axes, and inputs to the system come from a vertical gyro ($\theta$, $\phi$), yaw rate gyro ($r$), altitude sensor, and navigation receiver. Limiters are installed in the lateral channel to prevent excessive roll angles in response to large error signals. Gain scheduling is provided in the longitudinal channel to desensitize the system to glidepath errors as the runway threshold is approached. Automatic ILS approaches are possible down to decision height. The pilot must take over manually at that point because there is no automatic flare capability.

The auto-trim system, which actuates the elevator trim tab to unload the elevator servo, is not included in the block diagrams since it has no measurable influence on the performance or dynamic response of the airplane-autopilot system. It is a time-delayed, slow, integrating actuator and as such does not respond to transients.

The output of the autopilot system is expressed in control surface deflections. Because of the dependence of deflections on dynamic pressure, the final gain of $1/\bar{q}$ represents the aerodynamic gain of the control surfaces.

The glideslope system is engaged by approaching the glidepath at constant altitude (usually in the altitude hold mode) on the localizer beam. Initially the glideslope error signal will be a strong nose up command but switch B will be open. The "glideslope engage logic" circuit continuously monitors the signal at point B, which will gradually decrease in strength as the glidepath is approached. When the signal at point B reaches a 0.044 rad. (2.50°) nose up command, switch B is closed, and the altitude hold mode is
Figure 6. Longitudinal Control System
Figure 7. Lateral Control System
disengaged. To prevent a sudden pitchup at this point a 0.044 rad. nose down command is biased into the attitude circuit, to cancel the input from switch B. Thus, after engagement, the airplane will continue in level flight. As it moves closer to the glidepath, the signal at B will continue to decrease, and the bias signal will gradually pitch the nose down. The gains and logic switches are set so that the airplane gradually approaches the correct glideslope with minimal overshoot. Speed is controlled manually at all times in the representative autopilot system since auto-throttles are not typically provided. However, a preliminary auto-throttle was derived at TSC by simulation and can be used for completely automatic approach studies. The auto-throttle system is shown in Figure 8.
Figure 8. Auto-throttle
5. **INSTRUMENT PANEL**

The instrument panel layout is shown in Figure 9. It is designed to accommodate the customary advanced flight instruments on the left side in front of the pilot and engine instruments on the right side.
Figure 9. Instrument Panel Layout
REFERENCES


