THE DESIGN AND SIMULATION OF A TAKEOFF STABILIZATION SYSTEM FOR AN AIRCRAFT WITH AN AIR CUSHION LANDING SYSTEM

THESIS

GE/EE/77D-43

Edward A. Kenney
Captain CAF

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THE DESIGN AND SIMULATION OF A TAKEOFF STABILIZATION SYSTEM FOR AN AIRCRAFT WITH AN AIR CUSHION LANDING SYSTEM

THESIS

Presented to the Faculty of the School of Engineering of the Air Force Institute of Technology

Air University

in Partial Fulfillment of the Requirements for the Degree of Master of Science

by

Edward A. Kenney, B.Sc.

Captain CAF

Graduate Electrical Engineering

December 1977

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Preface

When the Air Cushion System replaced the conventional takeoff and landing systems of the Jindivik remotely piloted vehicle, the possibility existed that instabilities in pitch, roll, and yaw could occur. As a result, this paper was intended as a design of a Takeoff Stabilization System for the Jindivik using existing autopilot sensors and incorporating an engine yaw thruster and vertical wing tip roll thrusters. When the design was completed, it was sufficiently general that the technique could be applied to any air cushion aircraft or VTOL aircraft. The Landing Stabilization System for the Jindivik using the same sensors and actuators is presently being designed by Captain Max Stafford as his thesis for the Air Force Institute of Technology (AFIT).

I wish to express my gratitude to my Thesis advisors, Dr. George Kurylowich of the Air Force Flight Dynamics Laboratory (AFFDL) and Major R. Potter of AFIT. Also, thanks are due to Captain James Negro of AFIT, Major Jack Randall and Mr. Jim Steiger of the Air Force Flight Dynamics Laboratory for their technical advice and assistance.

My wife, Jane, does not know how much she has contributed to this study, but her patience, understanding, and encouragement has definitely made the past eighteen months of work much easier.

Edward Kenney
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### List of Symbols

All symbols are in ft-lb-sec units unless indicated to the contrary.

#### Alphanumeric Symbols

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<td>A</td>
<td>Area</td>
</tr>
<tr>
<td>AR</td>
<td>Aspect Ratio</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>Horizontal distance between the inner and outer trunk attachment points</td>
</tr>
<tr>
<td>b</td>
<td>Wing span</td>
</tr>
<tr>
<td>c</td>
<td>Wing chord</td>
</tr>
<tr>
<td>$C_D$</td>
<td>Drag coefficient</td>
</tr>
<tr>
<td>$C_{D0}$</td>
<td>Drag coefficient for zero angle of attack and zero elevator angle</td>
</tr>
<tr>
<td>$C_{D\alpha}$</td>
<td>Variation of drag coefficient with angle of attack</td>
</tr>
<tr>
<td>CG</td>
<td>Centre of gravity (of aircraft)</td>
</tr>
<tr>
<td>$C_L$</td>
<td>Lift coefficient</td>
</tr>
<tr>
<td>Symbol</td>
<td>Definition</td>
</tr>
<tr>
<td>--------</td>
<td>------------</td>
</tr>
<tr>
<td>$C_{L0}$</td>
<td>Lift coefficient for zero angle of attack and zero elevator angle</td>
</tr>
<tr>
<td>$C_{Lq}$</td>
<td>$\frac{2U}{C} \frac{dC_l}{dq}$</td>
</tr>
<tr>
<td>$C_{Lt}$</td>
<td>Variation of lift coefficient with pitch rate</td>
</tr>
<tr>
<td>$C_{L\alpha}$</td>
<td>Coefficient of lift of the tail</td>
</tr>
<tr>
<td>$C_{L\alpha_F}$</td>
<td>Aircraft lift curve slope</td>
</tr>
<tr>
<td>$C_{L\alpha_H}$</td>
<td>Lift curve slope of the vertical stabilizer</td>
</tr>
<tr>
<td>$C_{L\alpha_{2-D}}$</td>
<td>Lift curve slope of the horizontal stabilizer</td>
</tr>
<tr>
<td>$C_{L_2}$</td>
<td>Theoretical two-dimensional lift curve slope of an airfoil at $0^\circ$ absolute</td>
</tr>
<tr>
<td>$C_e$</td>
<td>$\frac{\gamma}{\frac{1}{2} \rho S b}$</td>
</tr>
<tr>
<td>$C_{e\rho}$</td>
<td>Rolling moment coefficient</td>
</tr>
<tr>
<td>$C_{L\alpha}$</td>
<td>Variation of lift coefficient with rate of change of angle of attack</td>
</tr>
<tr>
<td>$C_{L\alpha}$</td>
<td>Variation of rolling moment coefficient with roll rate</td>
</tr>
<tr>
<td>Symbol</td>
<td>Definition</td>
</tr>
<tr>
<td>--------</td>
<td>------------</td>
</tr>
<tr>
<td>$C_{e_r}$</td>
<td>$\frac{2U}{n} \frac{dc_l}{d\alpha}$</td>
</tr>
<tr>
<td>$C_{e_\beta}$</td>
<td>$\frac{dc_l}{d\beta}$</td>
</tr>
<tr>
<td>$C_m$</td>
<td>$\frac{m}{\frac{q}{S}c}$</td>
</tr>
<tr>
<td>$C_{m_\alpha}$</td>
<td>$\frac{2U}{c} \frac{dc_m}{d\alpha}$</td>
</tr>
<tr>
<td>$C_{m_\beta}$</td>
<td>$\frac{2U}{c} \frac{dc_m}{d\beta}$</td>
</tr>
<tr>
<td>$C_n$</td>
<td>$\frac{n}{\frac{q}{S}b}$</td>
</tr>
<tr>
<td>$C_{n_\alpha}$</td>
<td>$\frac{2U}{b} \frac{dc_n}{d\alpha}$</td>
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<td>$C_{n_\beta}$</td>
<td>$\frac{dc_n}{d\beta}$</td>
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<tr>
<td>$C_Y$</td>
<td>$\frac{F_y}{\frac{q}{S}}$</td>
</tr>
<tr>
<td>$C_{Y_F}$</td>
<td></td>
</tr>
<tr>
<td>Symbol</td>
<td>Definition</td>
</tr>
<tr>
<td>--------</td>
<td>------------</td>
</tr>
<tr>
<td>$C_{yp}$</td>
<td>$= \frac{2\mu}{b} \frac{dC_y}{d\beta}$</td>
</tr>
<tr>
<td>$C_{yr}$</td>
<td>$= \frac{2\mu}{b} \frac{dC_y}{d\gamma}$</td>
</tr>
<tr>
<td>$C_{yp}'$</td>
<td>$= \frac{dC_y}{d\beta}$</td>
</tr>
<tr>
<td>$D$</td>
<td>Drag</td>
</tr>
<tr>
<td>$d$</td>
<td>Distance between trunk inner attachment points</td>
</tr>
<tr>
<td>$F$</td>
<td>Force</td>
</tr>
<tr>
<td>$F_{Ax}$</td>
<td>Aerodynamic force in the $x$ direction</td>
</tr>
<tr>
<td>$F_{Ay}$</td>
<td>Aerodynamic force in the $y$ direction</td>
</tr>
<tr>
<td>$F_{Az}$</td>
<td>Aerodynamic force in the $z$ direction</td>
</tr>
<tr>
<td>$F_{CT}$</td>
<td>Trunk damping force</td>
</tr>
<tr>
<td>$F_T$</td>
<td>Force from the roll thrusters</td>
</tr>
<tr>
<td>Symbol</td>
<td>Definition</td>
</tr>
<tr>
<td>--------</td>
<td>------------</td>
</tr>
<tr>
<td>( F_x )</td>
<td>Force in the x direction</td>
</tr>
<tr>
<td>( F_{x_{EXT}} )</td>
<td>External force in the x direction</td>
</tr>
<tr>
<td>( F_y )</td>
<td>Force in the y direction</td>
</tr>
<tr>
<td>( F_{y_{EXT}} )</td>
<td>External force in the y direction</td>
</tr>
<tr>
<td>( F_{y_T} )</td>
<td>Force from the yaw thrusters</td>
</tr>
<tr>
<td>( F_z )</td>
<td>Force in the z direction</td>
</tr>
<tr>
<td>( F_{z_{EXT}} )</td>
<td>External force in the z direction</td>
</tr>
<tr>
<td>( g )</td>
<td>Acceleration due to gravity</td>
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<tr>
<td>( H_w )</td>
<td>Distance between the ground tangent points on the sides of the trunk</td>
</tr>
<tr>
<td>( H_y )</td>
<td>Height of trunk cross section</td>
</tr>
<tr>
<td>( \theta )</td>
<td>Incidence angle of the horizontal stabilizer</td>
</tr>
<tr>
<td>Symbol</td>
<td>Definition</td>
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<td>------------------------------------------------</td>
</tr>
<tr>
<td>$I_{xx}$</td>
<td>Roll moment of inertia of the aircraft about the CG</td>
</tr>
<tr>
<td>$I_{xz}$</td>
<td>Product of inertia of the aircraft about the CG</td>
</tr>
<tr>
<td>$I_{yy}$</td>
<td>Pitch moment of inertia of the aircraft about the CG</td>
</tr>
<tr>
<td>$I_{zz}$</td>
<td>Yaw moment of inertia of the aircraft about the CG</td>
</tr>
<tr>
<td>$k$</td>
<td>$C_L^2$ coefficient from drag polar</td>
</tr>
<tr>
<td>$L$</td>
<td>Lift</td>
</tr>
<tr>
<td>$L_5$</td>
<td>Length of the straight part of the trunk</td>
</tr>
<tr>
<td>$\mathcal{L}$</td>
<td>Rolling moment</td>
</tr>
<tr>
<td>$\mathcal{L}_A$</td>
<td>Aerodynamic rolling moment</td>
</tr>
<tr>
<td>$\mathcal{L}_{EXT}$</td>
<td>External rolling moment</td>
</tr>
<tr>
<td>$\mathcal{L}_{THRUSTERS}, \mathcal{L}_T$</td>
<td>Rolling moment of the roll thrusters</td>
</tr>
<tr>
<td>Symbol</td>
<td>Definition</td>
</tr>
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<td>------------</td>
</tr>
<tr>
<td>( l_F )</td>
<td>Distance from CG to mean aerodynamic chord of vertical stabilizer</td>
</tr>
<tr>
<td>( l_P )</td>
<td>Peripheral distance from inner trunk attachment to first row of trunk orifices</td>
</tr>
<tr>
<td>( l_t )</td>
<td>Distance from CG to mean aerodynamic chord of horizontal stabilizer</td>
</tr>
<tr>
<td>( l_w = \frac{b}{2} )</td>
<td>Length of one wing</td>
</tr>
<tr>
<td>( M )</td>
<td>Number of straight trunk segments in one quarter of trunk periphery</td>
</tr>
<tr>
<td>( m )</td>
<td>Mass</td>
</tr>
<tr>
<td>( M )</td>
<td>Pitching moment</td>
</tr>
<tr>
<td>( M_A )</td>
<td>Aerodynamic pitching moment</td>
</tr>
<tr>
<td>( M_{ext} )</td>
<td>External pitching moment</td>
</tr>
<tr>
<td>( N )</td>
<td>Number of curved trunk segments in one quarter of trunk periphery</td>
</tr>
<tr>
<td>Symbol</td>
<td>Definition</td>
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<td>--------</td>
<td>------------</td>
</tr>
<tr>
<td>Nh</td>
<td>Number of trunk orifices per row</td>
</tr>
<tr>
<td>Nr</td>
<td>Number of rows of trunk orifices</td>
</tr>
<tr>
<td>η</td>
<td>Yawing moment</td>
</tr>
<tr>
<td>ηa</td>
<td>Aerodynamic yawing moment</td>
</tr>
<tr>
<td>ηext</td>
<td>External yawing moment</td>
</tr>
<tr>
<td>ηYT</td>
<td>Yawing moment produced by yaw thrusters</td>
</tr>
<tr>
<td>Px</td>
<td>Roll rate (about x axis)</td>
</tr>
<tr>
<td>Py</td>
<td>Pressure</td>
</tr>
<tr>
<td>Pa</td>
<td>Atmospheric pressure</td>
</tr>
<tr>
<td>PaV</td>
<td>Average pressure</td>
</tr>
<tr>
<td>Pch</td>
<td>Cushion pressure</td>
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<tr>
<td>Pt</td>
<td>Trunk pressure</td>
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<tr>
<td>Ω, q, g</td>
<td>Pitch rate (about y axis)</td>
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<td>Symbol</td>
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<td>$\bar{q}$</td>
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<tr>
<td>$R_\gamma$</td>
<td>Yaw rate (about z axis)</td>
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<td>$S$</td>
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</tr>
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</tr>
<tr>
<td>$S$</td>
<td>Horizontal stabilizer reference area</td>
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<td>Time</td>
</tr>
<tr>
<td>$t_0$</td>
<td>Initial time</td>
</tr>
<tr>
<td>$t^*$</td>
<td>Final time</td>
</tr>
<tr>
<td>$u_x, u$</td>
<td>Velocity of CG along x axis</td>
</tr>
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<td>Symbol</td>
<td>Definition</td>
</tr>
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<td>----------------</td>
<td>------------------------------------------------</td>
</tr>
<tr>
<td>$U(t)$</td>
<td>Control variable</td>
</tr>
<tr>
<td>$U_{\text{MAX}}$</td>
<td>Maximum roll control</td>
</tr>
<tr>
<td>$U_{YT\text{MAX}}$</td>
<td>Maximum yaw thruster control</td>
</tr>
<tr>
<td>$V_y, v$</td>
<td>Velocity of CG along y axis (aircraft)</td>
</tr>
<tr>
<td>$U_{\ell}$</td>
<td>Trunk vertical velocity</td>
</tr>
<tr>
<td>$W_z, w$</td>
<td>Velocity of CG along z axis (aircraft)</td>
</tr>
<tr>
<td>$X_{cx}$</td>
<td>Distance of trunk segment centre from CG along vehicle x axis</td>
</tr>
<tr>
<td>$y$</td>
<td>Distance along wing</td>
</tr>
<tr>
<td>$Z_{cx}$</td>
<td>Distance of trunk segment centre from CG along vehicle y axis</td>
</tr>
<tr>
<td>$\theta_F$</td>
<td>Mean height of the vertical stabilizer</td>
</tr>
<tr>
<td>Symbol</td>
<td>Definition</td>
</tr>
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</tr>
<tr>
<td>$\alpha$</td>
<td>Angle of attack of aircraft</td>
</tr>
<tr>
<td>$\alpha_F$</td>
<td>Angle of attack of vertical stabilizer</td>
</tr>
</tbody>
</table>
| $\beta$ | 1. Sideslip angle  
2. Angle subtended by curved trunk segment from trunk centre of curvature |
<p>| $\epsilon$ | Downwash angle |
| $\delta_e$ | Elevator angle |
| $\delta_f$ | Flap angle |
| $\delta(l)$ | Angle of trunk curved segment from centre of curvature |
| $\delta_s$ | Side excursion of trunk |
| $\delta_x$ | Width of straight trunk segment |
| $\lambda$ | Sweep angle of leading edge of wing (or horizontal stabilizer) |</p>
<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \Delta )</td>
<td>Mathematical symbol meaning a small change</td>
</tr>
<tr>
<td>( \rho )</td>
<td>Air density</td>
</tr>
<tr>
<td>( \theta )</td>
<td>Pitch attitude angle</td>
</tr>
<tr>
<td>( \theta_p )</td>
<td>Inclination angle of lift and drag vectors</td>
</tr>
<tr>
<td>( \phi )</td>
<td>Roll angle</td>
</tr>
<tr>
<td>( \psi )</td>
<td>Yaw angle</td>
</tr>
</tbody>
</table>
Abstract

The inherent instability in pitch and roll associated with an Air Cushion Landing System (ACLS) aircraft at low airspeeds was investigated, and a means to aid control in pitch and roll was developed. The control system required the use of vertical wing tip thrusters which provided thrust up or down depending on the control signal (similar to space vehicle thrusters). These thrusters could be activated alternately to control roll angle and roll rate with the use of a bang-bang optimal controller. As well, the thrusters would be set forward of the aircraft centre of gravity and could be activated in tandem to aid in pitch control.

The Jindivik Remotely Piloted Vehicle, an Australian target drone, was fitted with an ACLS and taxi tests showed the instability and need for a stabilization system. Subsequent use of Jindivik wind tunnel and taxi test data served as the basis for the development of the roll/pitch control system presented in this paper. Due to computational problems with the air cushion model of the computer program, the controller designs could not be completely verified; but expected trends in pitch, roll and yaw control were shown.
THE DESIGN OF A TAKEOFF STABILIZATION SYSTEM FOR AN AIRCRAFT WITH AN AIR CUSHION LANDING SYSTEM

Chapter I

Introduction

In the low speed range of a takeoff roll, the normal aircraft controls are not aerodynamically effective; hence, the pilot must control the heading by differential braking and wait until the ailerons become effective to control roll. During this time, the landing gear dampens most of the pitch and roll oscillations so that the pilot has few corrections to make in the latter part of the takeoff roll. However, when the conventional landing gear is replaced by an Air Cushion Takeoff System (ACTS) the pitch and roll damping is greatly reduced. This paper will use the Jindivik Remotely Piloted Vehicle as an example of an air cushion vehicle that can be controlled in the low speed range with the use of small jet thrusters on the wing tips and a thrust deflector on the tail section.

The Jindivik Remotely Piloted Vehicle (RPV) is an Australian target drone that can be launched and recovered on a runway. At present, the takeoff is accomplished with a takeoff dolly, as shown in Fig. 1, that provides a wing level attitude and directional control. At lift off the Jindivik separates from the dolly and the dolly brakes to a stop. Recovery of the Jindivik is done by landing on a single, four inch wide metal skid attached to the fuselage. Directional control during landing is maintained with the ailerons, the rolling moment thus produced
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Fig. 1. Jindivik on a Takeoff Dolly

makes the drone ride up onto an edge of the skid and turn in the direction of the roll. For the last twelve years the Australian Air Force has used the Jindivik in this configuration with considerable success.

A joint project by the Australian Air Force and the United States Air Force was initiated in 1972 to incorporate an Air Cushion Landing System on the Jindivik. The objectives of this project were to convert the Jindivik to an all-terrain RPV and to advance air cushion technology. The drone and air cushion are shown in Fig. 2. Initial low speed taxi tests, completed in Australia, show that the aircraft fitted with the air cushion is unstable in roll, pitch, and directional control (yaw) (Ref. 13). Therefore, a Stability Augmentation System (SAS) will have
Fig. 2. Jindivik and Air Cushion Landing System
to be designed and incorporated into the autopilot before the
RPV is airworthy.

Since the drone was designed to be launched from a directionally controlled dolly, it was not designed with a rudder. Implementation of a rudder during this project would require extensive structural changes and major changes to the autopilot and ground control units. Therefore, a yaw thruster was designed and fitted to the rear of the fuselage to direct the jet exhaust, thus providing a yawing moment. Roll and pitch control will be provided by a vertical roll thruster on the front tip of each wing pod. The roll thrusters will be activated alternately to control rolling moments and in tandem to control pitch. Since the roll thrusters are on the tip
Fig. 3. Block Diagram of Autopilot and SAS Control Units

of the wing pods, they are approximately six feet ahead of the centre of gravity of the drone and hence can produce a moment to control the pitch attitude to some degree. As well, the roll thrusters can be directed up or down to counteract positive or negative rolling and pitching moments.

At velocities near fifty knots, the ailerons and flaps become aerodynamically effective and the roll thrusters are phased out to ensure that the vehicle is not overcontrolled. An additional advantage of turning off the roll thrusters is that a smaller gas supply is required for the thrusters. Hence, they can be used with a gas bottle rather than bleed air from the engine. This arrangement will greatly reduce the airframe and engine modifications required for implementation.
The stability augmentation system will be designed to use the existing sensors in the autopilot to control the roll and pitch thrusters. The SAS unit will be placed in the feedback control loop between the autopilot and actuators, as shown in Fig. 3.

This thesis is organized in the following manner: Chapter II develops the equations of motion and aerodynamic stability derivatives, Chapter III describes the air cushion model, Chapter IV discusses the controller design, Chapter V contains a description of the computer program and the simulation results, and Chapter VI the conclusions and recommendations.
Chapter II

The Determination and Solution of the Jindivik Equations of Motion

The following six simultaneous non-linear differential equations fully describe the motion of the Jindivik RPV. The positive sense of the variables is in the direction of the arrows in Fig. 4.

\[ m(\dot{U} - VR + WG) = -mg \sin \theta + F_{Ax} + F_{x_{ext}} \]  
\[ m(\dot{V} + UR - WP) = mg \sin \phi \cos \theta + F_{Ay} + F_{y_{ext}} \]  
\[ m(\dot{W} - UQ + VP) = mg \cos \phi \cos \theta + F_{Az} + F_{z_{ext}} \]  
\[ I_{xx} \dot{P} - I_{xz} \dot{R} - I_{xz} PQ + (I_{zz} - I_{yy})RQ = L_A + L_{ext} \]  
\[ I_{yy} \dot{Q} + (I_{xx} - I_{zz})PR + I_{xz}(P^2 - R^2) = M_A + M_{ext} \]  
\[ I_{zz} \dot{R} - I_{xz} \dot{P} + (I_{yy} - I_{xx})PQ + I_{xz}QR = N_A + N_{ext} \]

The equations are first order in U, V, W, P, Q, and R with the added kinematic relationships.

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

The assumptions used in the derivation of these equations were:

1. the aircraft is a rigid body, 
2. the mass of the aircraft is constant for the duration of the analysis, 
3. gravity is constant, 
4. the earth is an inertial reference, 
5. and there is body axis symmetry about the x-z plane (i.e., \( I_{xy} = 0 = I_{yz} \)).

Equations (1) to (9) are included in the subroutines of the "EASY Dynamic Analysis Computer Program to Aircraft Modeling"
Fig. 4. Definitions of Vector Components in the Equations of Motion

(Ref. 5) which was used to simulate the takeoff motions of the Jindivik. The inputs required by EASY are the mass, inertias, geometry of the aircraft, and the aerodynamic and external forces acting on the drone. The air cushion system is considered to be a prime generator of the external forces and moments (aside from engine forces and moments) and is described in the next chapter. The mass, inertias and geometry are readily available from aircraft blue prints and reference manuals; and the aerodynamic forces and moments can be computed from wind tunnel model data and theoretical methods.

The aerodynamic forces and moments can be written as:

\[ L = C_L \frac{\rho}{2} SL \]  \hspace{1cm} (10)
\[ D = C_D \frac{\rho}{2} SL \]  \hspace{1cm} (11)
where
\[ C_L = f(u, \alpha, \dot{\alpha}, \beta, \dot{\beta}, \delta_e, \delta_f) \] (16)
\[ C_D = f(u, \alpha, \dot{\alpha}, \beta, \dot{\beta}, \delta_e, \delta_f) \] (17)
\[ C_Y = f(\beta, \dot{\beta}, \rho, \tau, S_A) \] (18)
\[ C_L = f(\beta, \dot{\beta}, \rho, \tau, S_A) \] (19)
\[ C_n = f(\beta, \dot{\beta}, \rho, \tau, S_A) \] (20)
\[ C_m = f(u, \alpha, \dot{\alpha}, \beta, \dot{\beta}, \delta_e, \delta_f) \] (21)

The coefficients \( C_L, C_D, C_Y, C_L, C_n, \) and \( C_m \) are non-dimensional. By determining the coefficients for every flight condition, the aerodynamic forces and moments can be calculated and added to the external forces and moments to produce the aircraft motions. These calculations are done by the EASY program but the program requires all the aerodynamic stability derivatives (all the functional relationships which determine the force and moment coefficients). The remainder of this chapter will deal with the derivation of the stability derivatives. These derivatives are derived in the stability axis system as defined by Blakelock (Ref. 2).
Reference 13 contains wind tunnel data for the Jindivik in various configurations, including one when fitted with the Air Cushion Recovery System (ACRS). The ACRS is the air cushion trunk with which the drone lands, but it also has an Air Cushion Takeoff System (ACTS) trunk with which it takes off. After takeoff, the ACTS is disengaged and drops to the ground. Both trunks are the same shape with the ACTS being about 21\% larger in all dimensions. Since no wind tunnel data was available for the ACTS, the data for the ACRS was extrapolated by percentages and assumed to be fairly accurate for the ACTS. An example of the estimation technique is that the increase in the frontal area of the aircraft due to the replacement of the ACRS by the ACTS was 6\%; therefore, the values of the coefficient of drag ($C_D$) were increased by 7\%. Since the trunk does not generate lift, the coefficient of lift ($C_L$) was not affected, nor was the coefficient of side force ($C_Y$); symmetry about the x-z axis meant that the coefficient of yawing moment ($C_n$) was not affected. The pitching moment coefficient ($C_m$) and rolling moment coefficient ($C_L$) were affected by the percentage that the increased drag affected those moments. Thus, $C_D$, $C_L$, $C_Y$, $C_m$, and $C_n$ can be empirically determined as functions of the angle of attack ($\alpha$), the sideslip ($\beta$), and the elevator deflection ($\delta_e$). In other words \[ \frac{\partial C_D}{\partial \alpha}, \quad \frac{\partial C_D}{\partial \beta}, \quad \frac{\partial C_L}{\partial \alpha}, \quad \frac{\partial C_L}{\partial \beta}, \quad \frac{\partial C_Y}{\partial \alpha}, \quad \frac{\partial C_Y}{\partial \beta}, \quad \frac{\partial C_m}{\partial \delta_e}, \quad \frac{\partial C_m}{\partial \delta_e}, \quad \text{and} \quad \frac{\partial C_n}{\partial \delta_e} \] can be found from the wind tunnel data.

Non-dimensional derivatives were calculated because they provided a means of checking typical values and signs with
Roskam (Ref. 17) and Blakelock (Ref. 2). Before entering the derivatives into the EASY program, they were dimensionalized.

At low airspeeds the heave motion of the air cushion can create angles of attack beyond the stall limit, but at these speeds aerodynamic contributions to the aircraft dynamics are small.

**Stability Derivative Derivation**

\[ C_L \]

From a curve of \( C_L \) vs. \( \alpha \) of the wind tunnel data it can be shown that

\[ C_L = C_{L0} + \frac{dC_L}{d\alpha} \alpha \quad (22) \]

\( C_{L0} \) and \( C_{L\alpha} = \frac{dC_L}{d\alpha} \) can be determined directly as the \( C_L \) intercept and slope of the curve.

\[ C_D \]

From a drag polar of \( C_L \) vs. \( C_D \) it can be shown that

\[ C_D = C_{D0} + KC_L^2 \quad (23) \]

where \( C_{D0} \) and \( K \) are determined by a curve fit of wind tunnel data of \( C_D \) and \( C_L \). Substituting for \( C_L \)

\[ C_D = C_{D0} + K(C_{L0} + C_{L\alpha} \alpha)^2 \]

\[ = C_{D0} + K(C_{L0}^2 + 2C_{L0}C_{L\alpha} \alpha + C_{L\alpha}^2 \alpha^2) \quad (24) \]

Differentiating

\[ \frac{dC_D}{d\alpha} = K(2C_{L0}C_{L\alpha} + 2C_{L\alpha}^2 \alpha) \quad (25) \]

so

\[ C_{D\alpha} = \frac{dC_D}{d\alpha} = 2KC_L \alpha \quad (26) \]

Roskam (Ref. 17, pgs 4.12, 1.18, 4.25) shows that for velocities below 300 ft/sec the variation of lift, pitch moment, and drag with velocity is zero. Thus,

\[ \frac{dC_D}{du} = 0 = \frac{dC_L}{du} = \frac{dC_m}{du} \quad (27) \]
The quantities $\frac{\partial C_0}{\partial \delta e}$, $\frac{\partial C_0}{\partial \delta f}$, $\frac{\partial C_l}{\partial \delta e}$, $\frac{\partial C_l}{\partial \delta f}$, $\frac{\partial C_m}{\partial \delta e}$, $\frac{\partial C_m}{\partial \delta f}$ were all computed from curve-fits of the wind tunnel data.

From Roskam (Ref. 17) it can be shown that the angle of attack of the tail in downwash is

$$\alpha_t = \alpha - \epsilon - \epsilon$$

and for a particular angle of attack

$$\Delta \alpha_t = -\Delta \epsilon$$

$$= \frac{\partial \epsilon}{\partial \alpha} \Delta \alpha$$

$$= \frac{\partial \epsilon}{\partial \alpha} \Delta t$$

$$= \frac{\partial \epsilon}{\partial \alpha} \frac{lt}{u}$$

(29)

now the change in lift coefficient on the tail due to downwash is

$$\Delta C_{lt} = C_{lt} \Delta \alpha_t$$

$$= C_{lt} \frac{lt}{u} \frac{\partial \epsilon}{\partial \alpha}$$

(30)

The change in aircraft lift is

$$\Delta C_L = \Delta C_{lt} \frac{S_t}{S}$$

(31)

$$\frac{\partial C_L}{\partial \alpha} = C_{lt} \frac{lt}{u} \frac{\partial \epsilon}{\partial \alpha} \frac{S_t}{S}$$

(32)

thus $$(C_L)_{T.o.t} \frac{\partial C_L}{\partial \alpha} \approx \frac{2}{C} C_{lt} \frac{lt}{u} \frac{S_t}{S} \frac{\partial \epsilon}{\partial \alpha}$$

(33)

where

$$C_{lt} = \frac{AR \cos \lambda \, C_{L_{2-D}}}{AR \sqrt{1 + (C_{L_{1-D}} \cos \lambda)^2 + C_{L_{2-D}} \cos \lambda}}$$

(34)

$$C_{L_{2-D}} = 5.73$$

(35)

$\lambda$ is sweep angle and $AR$ is aspect ratio (Ref. 7)
The wing contribution to $C_L^w$ is considerable but cannot be estimated (using Roskam, DATCOM, etc.). So a "typical" value from Roskam of $C_L^w \approx 0.5 \text{ rad}^{-1}$ was used; fortunately this derivative is of minor importance (Ref. 17, p. 4.114).

**$C_m^w$**

The contribution of the wing was neglected because it will be negligible with respect to the tail contribution. The correction to the pitching moment due to downwash on the tail is

$$\Delta C_m^\text{Tail} = -\Delta C_L t \frac{S_t}{S} \frac{U}{C} $$

$$= -C_L t \frac{\partial C_m}{\partial \alpha} \frac{U}{C} \frac{S_t}{S} $$

and

$$\frac{\partial C_m}{\partial \alpha} = -\frac{C_L t}{C} \frac{\partial C_m}{\partial \alpha} \frac{U}{C} \frac{S_t}{S} $$

and

$$C_m^w = \frac{2U}{C} \frac{\partial C_m}{\partial \alpha} = -\frac{2C_L t \frac{U}{C} \frac{S_t}{S} \frac{C_m}{\partial \alpha}}{\partial \alpha}$$

**$C_L q$**

$q$ changes the angle of attack on the tail by $\frac{q\Delta t}{U}$ radians (for quasistatic conditions)

so

$$\Delta \alpha_t = \frac{q\Delta t}{U}$$

and

$$\Delta C_L = \frac{S_t}{S} \Delta C_L t = \frac{S_t}{S} C_L t \frac{q\Delta t}{U}$$

Differentiating

$$(\frac{\partial C_l}{\partial q})^\text{Tail} = C_L t \frac{S_t}{S} \frac{\Delta t}{U}$$

and the contribution of the wing body is negligible in comparison to the tail (Ref. 17, p. 153)

Now

$$C_L q = \frac{2q}{C} \frac{\partial C_l}{\partial q} = \frac{2C_L t \frac{S_t \Delta t}{S}}{C}$$
The moment on the tail is

\[ M_t = -l_t C_{lt} q S_t \]

and the change in moment due to a change in angle of attack is

\[ \Delta M_t = -q l_t S_t C_{lt} \Delta \alpha \]

\[ = -q l_t S_t C_{lt} \Delta x \]

\[ = -\frac{q}{2} \frac{l_t^2 S_t}{u} C_{lt} q \]

and

\[ \Delta C_{mt} = -q \frac{l_t^2 C_{lt} S_t}{u C_s} \]

since the wing contribution is negligible with respect to the tail (Ref. 17, p. 153)

\[ C_{mq} = \frac{2 u}{c} \left( \frac{\partial C}{\partial q} \right)_{\text{TAN}} = -\frac{2 \frac{l_t^2 C_{lt} S_t}{c^2 S}}{c^2 S} \]

Wind tunnel data gave \( C_y, C_d, \) and \( C_m \) vs. \( \beta \) for values of \( |\beta| \leq 7^\circ \), but in any takeoff with crosswind the sideslip will normally exceed 7°. The normal takeoff procedure will be to initially line the aircraft into the relative wind at the centerline and change the heading as the aircraft gains speed. This procedure should keep \( \beta \) within \( \pm 30^\circ \) and the present data can be curve fitted and extrapolated to this value. Consequently, expressions can be obtained for \( C_{y\beta}, C_{d\beta}, \) and \( C_{mq} \) from the data. The \( \beta \) derivatives have been assumed to be zero (Ref. 17). For the \( \beta, p, \) and \( \lambda \) derivatives the effect of sidewash on the tail has been neglected.
Derivatives

The change in $C_n$ from the tail side force due to roll rate $p$ is

$$
\frac{\Delta C_n}{\Delta t} = \Delta C_{\alpha F} \frac{S_F l_F}{S_b} = -C_{\alpha F} \frac{p y S_F l_F}{u S_b} \tag{48}
$$

$$
\frac{\partial C_n}{\partial p} = -C_{\alpha F} \frac{\alpha_F S_F l_F}{u S_b} \tag{49}
$$

where $\alpha_F$ is the mean height of the fin, and the effect of sideward has been neglected.

So

$$
\frac{C_{n p}}{\partial t} = \frac{2 U}{b} \left( \frac{\partial C_n}{\partial p} \right)_{\text{tail}} = -\frac{2 C_{\alpha F} \alpha_F S_F l_F}{S_b} \tag{50}
$$

The wing contribution is in two parts due to lift and drag. For positive $p$, the angle of attack is increased on the right wing and decreased on the left; thus inclining and changing the lift and drag vectors of each wing section. The inclination angle is $\theta_p = \frac{p y}{u}$, where $y$ is the spanwise coordinate of the section. The change in lift is

$$
\Delta L_{\text{left}} = \bar{\gamma} C_{L \alpha} \Delta \alpha C dy = \bar{\gamma} C_{L \alpha} \frac{p y}{u} C dy \tag{51}
$$

and

$$
\Delta L_{\text{right}} = -\bar{\gamma} C_{L \alpha} \frac{p y}{u} C dy \tag{52}
$$

where $C dy$ is the area of the wing section.

So the change in the yawing moment due to lift is

$$
\Delta M_{\text{left}} = -y \Delta L_{\text{left}} \frac{p y}{u} \tag{53}
$$
and \[ \Delta N_{\text{right}} = y \Delta L_{\text{right}} \frac{p}{u} \] (54)

so \[ \Delta N_{\text{section}} = - \frac{2 \bar{g} cdy}{u^2} C_{Lx} p^2 y^3 \] (55)

and the total yawing moment of the wing is

\[ \Delta N_{\text{total}} = \int \Delta N_{\text{section}} dy \]

\[ = -2 \frac{\bar{g} c p^2}{u^2} \int \frac{1}{32} y^3 dy \]

\[ = -\frac{\bar{g} c C_{Lx} p^2 b^4}{32 u^2} \] (56)

and \[ \frac{\partial \alpha}{\partial p} = -\frac{\bar{g} c p b^4 C_{Lx}}{16 u^2} \] (57)

\[ (C_m)\text{wing lift} = \frac{2u}{S b^2} \frac{\partial N}{\partial p} = -\frac{C_{Lx} p b^2 c}{8 S u} \] (58)

the change in drag is

\[ \Delta D_{\text{aft}} = -\bar{g} C_{Dx} \Delta \alpha c dy \]

\[ = -\bar{g} C_{Dx} \frac{p y}{u} c dy \] (59)

and \[ \Delta D_{\text{right}} = \bar{g} C_{Dx} p y c dy \] (60)

the corresponding change in yawing moment is

\[ \Delta N_{\text{aft}} = y \bar{g} \frac{C_{Dx} p y}{u} dy \] (61)

and \[ \Delta N_{\text{right}} = \bar{g} \frac{C_{Dx} p y^2 c dy}{u} \] (62)

so \[ \Delta N_{\text{section}} = 2 \bar{g} \frac{C_{Dx} p y^2 c dy}{u} \] (63)
\[ \Delta \gamma_{\text{tot}} = \frac{2 \bar{c} \zeta C_{D\alpha} \rho}{u} S_0 \int y^2 dy \]

and
\[ \Delta \gamma_{\text{tot}} = \frac{\bar{c} \zeta C_{D\alpha} \rho b^3}{12u} \]  

(64)

now
\[ \frac{\partial \gamma}{\partial p} = \frac{\bar{c} \zeta C_{D\alpha} b^3}{12u} \]  

(65)

and
\[ (C_{m_p})_{\text{drag}} \equiv \frac{2u}{S_b b^2} \frac{\partial \gamma}{\partial p} = \frac{C_{D\alpha} b c}{6 S} \]  

(66)

summing all effects
\[ C_{m_p} = -\frac{2C_{\lambda\kappa} \theta_F S_F}{S b^2} - \frac{C_{\lambda\kappa} \rho b^3}{8u} + \frac{C_{D\alpha} b^3}{6 S} \]

\[ = -\frac{2C_{\lambda\kappa} \theta_F S_F}{S b^2} - \frac{C_{\lambda\kappa} \rho b}{8u} + \frac{C_{D\alpha}}{6} \]  

(67)

\[ C_{\lambda\kappa} \]

\[ C_{\lambda\kappa} \] is often negligible (Ref. 17, p. 170) and the tail is the major contributor. Let the mean change in angle of attack of the vertical stabilizer (fin) be
\[ \Delta \alpha_F = -\frac{p \alpha_F}{u} \]  

(68)

where \( \alpha_F \) is the mean height of the fin

now
\[ \Delta C_{\lambda\kappa} = C_{\lambda\kappa} \alpha_F \Delta \alpha_F \]

\[ = -C_{\lambda\kappa} \frac{p \alpha_F}{u} \]  

(69)

so the change on the side force coefficient of the aircraft is (sidewash is neglected)
\[ \Delta C_Y = \frac{S_F}{S} \Delta C_{\lambda\kappa} = -\frac{S_F p \alpha_F}{S u} C_{\lambda\kappa} \]  

(70)

so
\[ \frac{\partial C_Y}{\partial p} = -\frac{S_F \alpha_F C_{\lambda\kappa}}{S u} \]  

(71)

and
\[ C_{\lambda\kappa} = \frac{2u}{b} \frac{\partial C_Y}{\partial p} = -\frac{2S_F \alpha_F C_{\lambda\kappa}}{S b} \]  

(72)
The wing is the only major contributor (Ref. 17, p. 170). Etkin (Ref. 9) shows that the rolling velocity, $\dot{\rho}$, produces a change in the angle of attack of each wing section which is proportional to the span, i.e.,

$$\Delta \alpha = \frac{\dot{\rho} Y}{u}$$

(73)

where $Y$ is the spanwise coordinate of the wing section. Then the lift distribution on the wing due to rolling is estimated to be

Etkin (Ref. 9) changes the triangular lift distribution to a sinusoidal distribution to account for the loss of lift at the wing tips due to spanwise flow around the wing tips. However, since the Jindivik has large tip tanks the lateral airflow will be minimized and the lift distribution will be closer to the triangular distribution shown.

The change in lift will be

$$\Delta L = 2 \bar{g} C_{L\alpha} \Delta \alpha \, c \, dy$$

$$= 2 \bar{g} C_{L\alpha} \frac{c \rho Y}{u} \, dy$$

(74)

and the change in rolling moment due to this lift will be

$$\Delta \mathcal{L} = -\gamma \bar{g} C_{L\alpha} \frac{c \rho Y}{u} \, dy$$

(75)
thus \[ I = -\frac{\bar{F}}{\bar{g}} \frac{C_{L \alpha} \rho}{\rho} \int_0^b y^2 dy \]
\[ = -\frac{\bar{F}}{\bar{g}} \frac{C_{L \alpha} \rho b^3}{24 \rho} \]  
(76)

so \[ \frac{dI}{dp} = -\frac{\bar{F}}{\bar{g}} \frac{C_{L \alpha} \rho b^3}{24 \rho} \]  
(77)

and \[ \frac{dC_p}{dp} \approx \frac{1}{\bar{g} \bar{s} \rho b} \frac{dI}{dp} = -\frac{C_{L \alpha} b \rho}{\bar{g} \bar{s} 24 \rho} \]  
(78)

so \[ C_{L p} \approx \frac{2 U}{b} \frac{dC_p}{dp} = -\frac{C_{L \alpha} \rho b c}{12 s} = -\frac{C_{L \alpha}}{12} \]  
(79)

Derivatives

\( C_{y r} \)

The tail is the prime contributor (Ref. 17, p. 174). The change in fin angle of attack due to a yaw rate is

\[ \Delta \alpha_F = \frac{\tau F}{U} \]  
(80)

so the change in side force due to the tail is

\[ \Delta C_y = C_{L \alpha F} \frac{S_F \tau F}{S U} \]  
(81)

\[ \frac{dC_y}{dr} = C_{L \alpha F} \frac{S_F \tau F}{S U} \]  
(82)

\[ C_{y r} \approx \frac{2 U}{b} \frac{dC_y}{dr} = \frac{2 C_{L \alpha F} S_F \tau F}{S b} \]  
(83)

\( C_{e r} \)

Contributions are from the wing and tail. The side force on the tail acts at \( y_F \), the mean height of the fin and

\[ \Delta F_y = \Delta C_y \bar{g} S \]
\[ = \bar{g} S C_{L \alpha F} \frac{S_F \tau F}{S U} \]  
(84)
now

$$\Delta L = \Delta F_y \frac{C_l}{S}$$

$$= \frac{3}{2} S C_l \left( \frac{S_F r L F A_F}{S U} \right)$$

(85)

differentiating

$$\frac{dL}{dr} = \frac{3}{2} S C_l \left( \frac{S_F r L F A_F}{S U} \right)$$

(86)

and

$$C_{l_{r1}} = \frac{2U}{b} \frac{1}{\frac{S}{Gb}} \frac{dL}{dr}$$

$$= \frac{2C_l A_F S F L F A_F}{S b^2}$$

(87)

A positive $\Delta$ also increases lift on the left wing and decreases it on the right; the change in lift on each wing section is

$$\Delta L_{\text{left section}} = \Delta \frac{3}{2} C_l C d y$$

$$= \frac{1}{2} \rho (\Delta u)^2 C_l C d y$$

$$= \frac{1}{2} \rho (\Delta y)^2 C_l C d y$$

(88)

and

$$\Delta L_{\text{right section}} = -\frac{3}{2} \rho C_l C d y$$

$$= -\frac{1}{2} \rho (\Delta y)^2 C_l C d y$$

(89)

now the change in rolling moment due to two sections at $y$ is

$$\Delta L = \rho A^2 y^3 C_l C d y$$

(90)

and the total rolling moment change for the wing is

$$\Delta L = \rho A^2 C_l C S_b^b y^3 d y$$

$$= \frac{\rho A^2 C_l C b^4}{64}$$

$$= \frac{32}{3} \frac{r^2 C_l C b^4}{u^2}$$

(91)
now
\[
\frac{\partial L}{\partial r} = \frac{\bar{q} r c_c b^4}{16 u^2}
\]  \(92\)

\[\left( \frac{d C_L}{d \alpha} \right)_{\text{wing}}^{\alpha} \frac{1}{\bar{q} b} \frac{d L}{d r} = \frac{r C_L b^2}{16 u^2} \]  \(93\)

and
\[\left( C_L \right)_{\text{wing}} \frac{\partial}{\partial r} \frac{2u}{\bar{q} b} \frac{d C_l}{b} = \frac{r C_L b}{8 u} \]  \(94\)

summing the components
\[C_{\text{lr}} = \frac{2 C_{L_F} S_F l_F A_F}{S b^2} + \frac{r C_L b}{8 u} \]  \(95\)

\[C_{\alpha r} \]

The tail and wing contribute to \(C_{\alpha r} \). Knowing that the change in the fin angle of attack is
\[\Delta \alpha_F = \frac{-r l_F}{u} \]  \(96\)

and the moment arm of the tail is \(l_F\) then
\[\left( \Delta M \right)_{\text{tail}} = -\frac{l_F C_{L_F} S_F r l_F}{S} \]  \(97\)

so
\[\left( \frac{\partial M}{\partial r} \right)_{\text{tail}} = -\frac{S}{S} C_{L_F} \frac{S_F l_F^2}{S} \]  \(98\)

so
\[\left( C_{\nu r} \right)_{\text{tail}} \frac{\partial}{\partial r} \frac{2u}{\bar{q} b} \frac{1}{\bar{q} b} \frac{d M}{d r} \]

\[= -2 C_{L_F} \frac{S_F l_F^2}{S b^2} \]  \(99\)
A positive $\pi$ increases the drag on the left wing and decreases drag on the right wing. The change in drag on each wing section is

$$\Delta D_{\text{left section}} = \Delta \frac{1}{2} \rho C_D \frac{1}{2} \rho v^2 dy = \int_{-b}^{0} \frac{1}{2} \rho C_D \frac{1}{2} \rho v^2 dy$$

and

$$\Delta D_{\text{right section}} = -\frac{1}{2} \rho C_D \frac{1}{2} \rho v^2 dy$$

so the change in yawing moment is

$$\Delta N = \frac{1}{2} \rho C_D \frac{1}{2} \rho v^2 \frac{b^4}{64}$$

now

$$\frac{\Delta N}{\Delta \pi} = \frac{1}{32} \frac{1}{8} \frac{1}{b^4}$$

and

$$(C_n)_{\text{wing}} = 2 \pi \frac{1}{b} \frac{1}{5 \pi b} \frac{\Delta N}{\Delta \pi}$$

summing the components

$$C_n = \frac{2 C_l a e S_p l^2}{5 b^2} - \frac{C_D b}{8 u}$$

Once all the derivatives had been determined or estimated, Ref. 15 was used to convert to dimensional body-axis derivatives. The derivatives were written into the EASY program.
The air cushion model used in this analysis is a truncated version of an ACLS model that was designed by Foster-Miller Associates Inc. of Waltham, Massachusetts for the National Aeronautics and Space Administration (NASA) (Ref. 4).

The basic ACTS configuration is shown in Fig. 5. The model includes four primary subsystems: (1) the fan, (2) the feeding system, (3) the trunk, and (5) the cushion. Air from the fan flows through the ducts and plenum (feeding system) and enters the trunk. The trunk has several rows of orifices that exhaust both to the cushion and the atmosphere. Thus, the airflow from the trunk has two components, one entering the cushion and the other leading directly to the atmosphere. The cushion flow exhausts to the atmosphere through the clearance gap formed between the trunk and ground. In addition to the basic flows described above, two other flows have been included in the model. These are the plenum bleed flow and the direct cushion flow. Plenum bleeding causes some of the air to flow directly from plenum to atmosphere, and has been used in some designs to improve the dynamic characteristics of the air supply system. Direct flow from the plenum to the cushion can also improve dynamic response. A pressure relief valve is also included in the basic configuration. It allows additional flow to vent from the plenum whenever the pressure exceeds a preset level, and thus improves stability by reducing fan stall.

The support force acting on the aircraft is made up of two components. The first occurs due to the cushion pressure acting over the cushion area. The second, which comes about only during ground contact, is given by the
contact pressure acting over the trunk contact area. The support force, in general, also gives rise to a moment, given by the product of the force and its distance from the CG of the aircraft.

In plan, the cushion has an oval shape, made up of a rectangular section with semicircular ends. The lengths $a$ and $b$ are the horizontal and vertical spacing between the points of attachment of the trunk to the aircraft body. The initial (undeformed) trunk shape is defined by the above two parameters and the perimeter $l_p$ and height $H_y$ as shown. $S_h$ is the (uniform) spacing between the rows of peripherally distributed orifices. The number of the orifices is selected independently by the number of orifice rows $N_f$ and the number of orifices per row $N_h$. The cushion volume consists of two parts: an active (dynamically varying) region and a dead (static)
region. The active volume depends on the trunk shape and ground profile. The dead volume, which is a design variable, includes recesses in the cushion cavity as shown.

The forces transferred to the aircraft act through the cushion and trunk. To help calculate these forces, the trunk and cushion are divided into segments as shown in Fig. 6. Each straight section of the cushion and trunk is divided into $M$ rectangular segments, while each curved end is divided into $N$ pie-shaped segments. Thus, the total number of segments is $2(M + N)$. All cushion and trunk parameters are calculated first for each segment and then summed to give their total system values.

The dynamic analysis of the vehicle system is best derived with the help of two orthogonal coordinate frames of reference: a coordinate frame fixed in space (inertial frame), and a coordinate frame fixed to the vehicle (vehicle frame) with origin at the aircraft CG. The reason for two frames can be appreciated by recognizing that:

(a) Newton's law for translation motion requires that the CG acceleration be expressed relative to the inertial frame.

(b) The corresponding law for rotational motion, while valid in both inertial and vehicle frames, is applied more conveniently in the vehicle frame, because rotational inertia about any vehicle axis is constant, while the rotational inertia about any inertial (fixed) axis varies with aircraft position.

Accordingly, the two frames of reference have been defined as shown in Fig. 7. The vehicle frame with origin at the aircraft CG
Fig. 6. Division of Trunk Into Segments

has roll, yaw and pitch axes x, y and z, respectively, fixed to the aircraft body as shown. The inertial frame has corresponding axes X, Y and Z fixed in space. The two frames coincide only when the aircraft has not undergone any rotation from equilibrium.

In the analysis, the actual runway profile underneath the ACTS is approximated by segments that coincide in plan with those of the trunk and are parallel to the cushion hard surface as shown in Fig. 7. With this model, all pressure forces act parallel to the vehicle yaw axis so that the segment torque components about the aircraft CG can be easily computed by multiplying the segment force by the fore-and-aft and/or lateral separation between the segment and the CG.
Fig. 7. Inertial and Vehicle Coordinate Frames
The analytical model of the ACTS consists of a set of equations which when solved determines the pressures, flows, forces, and motion of the system for various aircraft and runway parameters. The overall ACTS system is divided into two interrelated systems: the flow system and the force system. These systems are shown in Fig. 8 and Fig. 9. The flow system establishes the pressure-flow relationship for various subsystems of the ACTS. The force system establishes the corresponding force-motion relationships. The interdependence of the two systems comes about because the trunk deflection obtained from the force system changes the volumes and orifice areas that form part of the flow system. Similarly, the cushion and trunk pressures found from the flow system give rise to forces and moments that form inputs to the force system.

The Trunk Model

The major component of the ACTS model is the trunk model because it determines the trunk shape parameters (volume, and orifice and contact areas), contact pressure distribution and damping that form inputs to the ACLS flow and force systems.

Trunk Shape. In past work, two analytical models have been developed for the trunk shape: the Membrane Trunk Model (Ref. 8) and the Frozen Trunk Model (Ref. 6). The shortcoming of both these analyses was that they modeled the side and end segments of the trunk in the same way while test data now confirm that the shorter curved end segments (front and rear) behave very differently from the longer, straight side segments. Fig. 10 shows the trunk cross section measured at the center of the side and end segments as the load on the cushion is increased. The entire side segment tends to bow outward and avoid ground contact, while the end segment remains virtually fixed, except for a flattening.
Fig. 8. ACTS Flow System
in the region that actually touches the ground. This difference in behavior occurs because the front segment is much smaller than the side segment and is curved. When the cushion pressure increases due to an increase in the load, the radially outward force causes the oval trunk planform to become more circular, as shown in Fig. 11. This causes a hoop tension force, \( T \), to act around the trunk periphery. In the side segments, this force acts substantially normal to the side excursion, \( \delta_s \), so that its component resisting the motion is negligible and the side segment can thus bow outwards relatively unrestrained. In the end segments the situation is different, since the curvature of the segment causes the hoop tension to have a much higher component opposing the motion so that outward motion of the trunk ends is much smaller.
Fig. 10. Measured Trunk Profile
Fig. 11. Outward Excursion of Trunk Segments
Since hoop tension has very little effect on side trunk motion, the side segments can be considered as simple, two-dimensional membranes, as done in the Membrane Trunk Model. On the other hand, the fact that hoop tension restrains ("freezes") the trunk ends suggests that these segments be modeled by the Frozen Trunk Model. Thus, the logical step in trunk model improvement is to combine the two existing models and form the Hybrid Trunk Model, in which the sides are represented by the Membrane Model and the ends by the Frozen Model.

The Hybrid Trunk Model is essentially a limiting case analysis of trunk deflection. In general, best results will be obtained at the middle of the respective segments, i.e., at the center of the side segments, where the trunk behaves very much like an ideal membrane, and at the center of the end segments, where the trunk shape is truly fixed. In the transition region (at and near where the segments meet), the trunk will exhibit properties of both the membrane and frozen trunk approximations.

**Contact Pressure.** In addition to trunk and cushion shape, the trunk model also determines the pressure distribution in the ground contact zone. The analysis for pressure distribution is complicated by the fact that two separate effects must be considered: direct trunk-ground contact caused by the trunk pressure forcing the trunk against the ground, and airflow through the trunk holes into the interstices that remain in the contact zone.

When two bodies in contact are acted upon by a force, \( F \), the actual contact occurs at a number of discrete regions rather than over the whole area, due to the inherent roughness of the contacting surfaces as shown in Fig. 12. Because the number of contact
Trunk Pressure, $P_t$

Atmospheric Pressure, $P_a$

Cushion Pressure, $P_c$

(a) Contact Geometry

Local Trunk-Ground Contact Pressure

Pressure due to Outflow from Trunk

Mean Contact Pressure

(b) Theoretical Pressure Profile

Mean Contact Pressure

Trunk Pressure

(c) Observed Pressure Distribution

Fig. 12. Pressure Distribution in Trunk Contact Zone
regions is large for a "smooth" surface, it was convenient to define an average contact pressure, $P_{av} = F/A$, acting as though the bodies were touching uniformly over the entire area, $A$. In fact, $P_{av}$ equals the trunk pressure, $P_t$. For purposes of trunk outflow calculation, the pressure profile in the non-contacting regions is approximated by a linearly decreasing relationship as shown in Fig. 12. The driving pressure for flow through any trunk hole is thus given by the difference between the trunk pressure and the gap, $P_{t} - P_{g}$. This pressure distribution model has been compared to experimental data and has been quite accurate (within 10%) (Ref. 4).

Trunk Damping. In dynamic operation, the trunk is deformed cyclically both in tension and flexure, and energy dissipation in the trunk material gives rise to a damping force which opposes the strain rate. Because the present trunk analysis does not solve for strain (and hence strain rate), a damping model that links trunk material properties directly to trunk damping forces cannot be developed. An alternate approach in which the damping characteristics are modeled by dimensional analysis (similarity) based on test data thus appears more appropriate. In keeping with the method of approach outlined earlier, the trunk is divided into segments (Fig. 13) and a series of dashpots, one for each segment, is included in the model such that the segment damping force is proportional to the vertical velocity of the trunk segment.

Each dashpot models the energy dissipation characteristic of the trunk segment. Although all parts of the trunk dissipate energy, the major contributions will come from those parts that undergo high stress reversals, since the strain rate is highest in these sections. Observations of a trunk in dynamic operation suggest that the high
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stress reversal regions lie along the periphery of the trunk-ground contact zone, because it is here that the rate of change of trunk slope (and hence stress) is high and constantly changing with the time as the contact area changes. As a first order approximation, the damping model derived here assumes that all the energy dissipation in the trunk is concentrated along the trunk-ground contact periphery so that the damping coefficient of each dashpot depends on the perimeter of the ground contact zone. This means that when a segment is not contacting the ground it has zero damping and when it is contacting the ground it has a damping coefficient proportional to the contact perimeter.

Model Synopsis

The Flow System

(a) The fan is characterized by a static pressure rise element for forward and back flow in series with an inertance (duct) and a capacitance (volume).

(b) The trunk and cushion volume are found from the Hybrid Trunk Model, which characterizes the side trunk segment as an ideal two-dimensional membrane and the end segment as a "frozen" trunk.

(c) The orifice areas between the trunk and cushion, trunk and atmosphere and cushion and atmosphere are found from the trunk shape as predicted by the Hybrid Trunk Model, along with the cushion orientation and ground profile.

(d) The pressure within the cushion, trunk and plenum is considered to be uniform.

(e) The pressure in the trunk/ground contact zone is found from the triangular profile given by the Hybrid Trunk Model.
(f) The flow through the plenum, trunk and cushion is governed by the unsteady state flow continuity equation in which the air is assumed to behave like a perfect gas and follow a polytropic expansion relationship.

(g) The flow through all orifices is found from the incompressible flow square-law orifice equation.

**The Force System**

(a) The mean contact pressure in the trunk/ground contact zone is equal to the trunk pressure.

(b) The trunk contact area and location relative to the aircraft CG is found from the trunk shape predicted by the Hybrid Trunk Model.

(c) The cushion area and location relative to the aircraft CG is found from the Hybrid Trunk Model. In width, the cushion extends between the lowest (ground tangent) points of the side trunk segments. In length, it extends between the ground tangent points of the end trunk segments, or, if in ground contact, between the inner edges of the contact zone.

(d) The total forces and moments acting on the aircraft occur due to the mean trunk contact pressure acting over the contact area, the cushion pressure acting over the cushion area, aerodynamic drag and trunk damping losses caused by aircraft heave motion, and trunk-ground friction.

(e) The forces and moments are found by dividing the cushion (and trunk) into segments, approximating the actual ground profile underneath the cushion by a similar set of segments parallel to the cushion, computing the cushion and contact pressure forces and moments for each segment, and then summing them to determine the total force.
(f) The heave motion of the aircraft is found by applying Newton's law in the vertical direction to the aircraft CG.

(g) Angular accelerations in pitch and roll are obtained by applying the theorem of moment of momentum about the aircraft pitch and roll axes.

(h) A coordinate transformation is carried out to express vehicle frame velocities and accelerations in terms of Euler angles and their derivatives.

(i) The moment of momentum equations, expressed in terms of Euler angles are integrated to give the angular position of the aircraft as a function of time.
Chapter IV
Controller Design

During the low speed portion of the takeoff roll (to approximately 50 knots), the controls available are a yaw thruster on the rear fuselage and vertical roll thrusters on each wing tip. The roll thrusters can be directed up or down. During the takeoff sequence, the most unstable mode of the aircraft is the roll mode. Therefore, it was decided to control this mode and observe the control that was applied to the pitch mode through the inertial cross-coupling. Also, during takeoff the yaw angle will be controlled by the yaw thruster on the rear of the fuselage. With roll and yaw controlled, Eqn (4), which is rewritten here, can be simplified.

\[
\mathbf{I}_{xx} \mathbf{\ddot{P}} - \mathbf{I}_{\alpha \beta} (\mathbf{\dot{R}} + \mathbf{PQ}) + (\mathbf{I}_{\alpha \gamma} - \mathbf{I}_{\beta \gamma}) \mathbf{RQ} = \mathbf{\mathcal{L}}
\]  

(106)

Controlling roll and yaw means that \( \dot{R} \) will be small and \( \mathbf{PQ} \) and \( \mathbf{RQ} \) (products of small numbers) will be small. \( \mathbf{Q} \) can be considered small because the takeoff starts with zero initial conditions on \( \mathbf{P}, \mathbf{Q}, \) and \( \mathbf{R} \). With the above simplification and assuming that any roll inputs from the ground profile are impulsive, then the roll moment generated by the roll thrusters can be written as

\[
\mathbf{I}_{xx} \mathbf{\ddot{\phi}} = \mathbf{\mathcal{L}}_{\text{thrusters}} = \ell \omega \mathbf{F}_T
\]

(107)

so

\[
\mathbf{\mathcal{L}}_{\text{thrusters}} = \mathbf{C} \mathbf{F}_T(t)
\]

(108)

where

\[
\mathbf{C} = \frac{\ell \omega}{\mathbf{I}_{xx}} = 8.28 \times 10^{-3}
\]
now let
\[ X_1 = \phi \quad (109) \]
\[ X_2 = \dot{\phi} \quad (110) \]
and
\[ u(t) = F_T(t) \quad (111) \]
so Eqn (108) can be rewritten as
\[ \dot{X}_2(t) = C u(t) \quad (112) \]
and
\[ \dot{X}_1(t) = X_2(t) \quad \text{from Eqn (110)} \quad (113) \]
so
\[ \dot{X}(t) = \begin{bmatrix} 0 & 1 \\ 0 & 0 \end{bmatrix} \phi(t) + \begin{bmatrix} 0 \\ 1 \end{bmatrix} u(t) \quad (114) \]

For a minimum time Performance Index and for \( |u(t)| \leq U_{\text{max}} \) an optimal control for this system can be shown to be a bang-bang control (Ref. 14, pgs 245-248). In other words, the control is a maximum (either positive or negative) whenever it is applied. Since the eigenvalues for \( A \) are both zero, Theorems 5.4-1 and 5.4-3 of Kirk (Ref. 14) show that an optimal control exists, is unique and has at most one switching.

Therefore, the control for a specified initial state must be
\[ u^*(t) = \begin{cases} +U_{\text{max}}, & t_0 \leq t < t^*, \text{ or} \\ -U_{\text{max}}, & t_0 \leq t < t^*, \text{ or} \\ +U_{\text{max}} \text{ for } t_0 \leq t < t_1, \text{ and } -U_{\text{max}} \text{ for } t_1 \leq t < t^*, \text{ or} \\ -U_{\text{max}} \text{ for } t_0 \leq t < t_1, \text{ and } +U_{\text{max}} \text{ for } t_1 \leq t < t^*, \text{ or} \\ 0, & \text{for } A(t) = 0 \end{cases} \quad (115) \]
Integrating Eqns (112) and (113) with \( U = \pm U_{\text{max}} \) gives

\[
\begin{align*}
\chi_z(t) &= C \int u(t) \, dt \\
&= \pm C U_{\text{max}} t + C C_2 \tag{116}
\end{align*}
\]

where \( C_2 \) is the value of \( \chi_z \) at \( t = t_0 \)

and

\[
\begin{align*}
\chi_1(t) &= \int \chi_z(t) \, dt \\
&= \int (\pm C U_{\text{max}} t + C C_2) \, dt \\
&= \frac{\pm C U_{\text{max}} t^2}{2} + C C_2 t + C_1 \tag{117}
\end{align*}
\]

where \( C_1 \) is the value of \( \chi_1 \) at \( t = t_0 \)

solving for \( t \) in (116)

\[
t = \frac{\chi_z(t) - C C_2}{\pm C U_{\text{max}}} \tag{118}
\]

substituting \( t \) into (117)

\[
\chi_1(t) = \frac{\pm C U_{\text{max}}}{2 C^2 U_{\text{max}}} \left( \chi_z^2(t) - 2 C C_2 \chi_z(t) + C^2 C_2 \right) \\
&+ \frac{C C_2 (\chi_z(t) - C C_2)}{\pm C U_{\text{max}}} + C_1 \tag{119}
\]

for \( U = \pm U_{\text{max}} \)

\[
\begin{align*}
\chi_1(t) &= \frac{1}{2 C U_{\text{max}}} \left( \chi_z^2(t) - C C_2 \chi_z(t) + \frac{C^2 C_2}{2 U_{\text{max}}} \right) + \frac{C C_2 (\chi_z(t) - C C_2)}{\pm C U_{\text{max}}} + C_1 \\
&= \frac{1}{2 C U_{\text{max}}} \chi_z^2(t) + C_3 \tag{120}
\end{align*}
\]

where

\[
C_3 = C_1 - \frac{C C_2^2}{2 U_{\text{max}}} \tag{121}
\]
for \( U = -U_{\text{max}} \)

\[
N_1(t) = -\frac{1}{2CU_{\text{max}}} X_2^2(t) + C_2 X_2(t) - \frac{C_2 C_2^2}{2U_{\text{max}}} X_2(t) + C_1 \frac{C_2^2 + C_1}{U_{\text{max}}}
\]

\[
= -\frac{1}{2CU_{\text{max}}} X_2^2(t) + C_1 + \frac{C_2 C_2^2}{2U_{\text{max}}}
\]

where \( C_4 = C_1 + \frac{C_2 C_2^2}{2U_{\text{max}}} \) \( (123) \)

the switching curve is

\[
N_1(t) = -\frac{1}{2CU_{\text{max}}} X_2^2(t) \| X_2(t) \| \quad (124)
\]

Let

\[
S_X = \frac{N(t) + \frac{1}{2CU_{\text{max}}} X_2^2(t) \| X_2(t) \|}{U_{\text{max}}}
\]

so

\[
U_*(t) = \begin{cases} 
- U_{\text{max}}, & S_X > 0 \\
U_{\text{max}}, & S_X < 0 
\end{cases}
\]

It can be noted that this controller design is almost completely independent of the aircraft type. In the low speed range where aerodynamic controls are not effective, this design will help stabilize the roll mode of any aircraft. The only relationship between the aircraft and controller is that the thruster force is a function of roll inertia and wing span. Thus, this design becomes very versatile
and applicable to stabilize the roll mode of any ACLS aircraft.

A somewhat similar analysis will be made for the controller of the yaw thruster. The criterion for directional control is to keep the aircraft on the runway centreline during takeoff; this can be accomplished by minimizing the lateral deviation, \( y \), and rate of deviation from the centreline, \( \dot{y} \). This deviation and rate will be minimized by yawing the aircraft in a direction to oppose the disturbance with the use of the yaw thruster.

Prior to the installation of the air cushion, the directional stability of the Jindivik was controlled by a batsman at the end of the runway. His job was to steer the dolly (Fig. 1) to keep the aircraft on the centreline. The same batsman will visually sense lateral deviation and deviation rate and control the yaw acceleration to indirectly control the lateral deviation and rate.

Assuming that the pitch and roll angles are kept small, then the lateral acceleration (to correct a lateral displacement) in the inertial frame of reference of the runway is a function of the thrust and yaw angle.

\[
\ddot{y} \approx \frac{\text{THRUST} \sin \Psi}{\text{MASS}}
\]  

(127)

and since \( \Psi \) will be small \((< 30^\circ)\) to keep the aircraft on the runway centreline, then

\[
\ddot{y} \approx \frac{\text{THRUST}}{\text{MASS}} \Psi
\]  

(128)

Equation 128 can be implemented as shown in Fig. 14. With these control loops the desired yaw angle to zero lateral displacement
can be determined. Referring to Fig. 14

\[ \psi_d = -K_1 \ddot{y} - K_2 y \]  

(129)

the inner loop open loop transfer function is

\[ (G_1 H)_1 = \frac{K K_1}{S} \]  

(130)

and the equivalent closed loop transfer function is

\[ G_1 = \frac{G H}{1 + G H} \]

\[ = \frac{K K_1}{S + K K_1} \]  

(131)

the outer loop open loop transfer function is therefore

\[ (G_2 H)_2 = \frac{G_1 K_2}{S} \]

\[ = \frac{K K_1 K_2}{S(S + K K_1)} \]  

(132)

this gives a root locus as shown in Fig. 15. For a damping ratio of 0.7 the closed loop roots are located at \((-\frac{K K_1}{2}, -\frac{K K_1}{2}\)) which gives a closed loop transfer function of

\[ G_2 = \frac{K K_1 K_2}{(S + \frac{K K_1}{2} \pm j K K_1)} \]

\[ = \frac{K K_1 K_2}{S^2 + K K_1 S + \frac{K^2 K_1}{2}} \]  

(133)

but

\[ G_2 = \frac{(G H)_2}{1 + (G H)_2} \]

\[ = \frac{K K_1 K_2}{S^2 + K K_1 S + K K_1 K_2} \]  

(134)
Fig. 14. Feedback Control Loops for Lateral Acceleration

Fig. 15. Root Locus for Fig. 14
therefore equating denominator coefficients gives

\[ KK_1K_2 = \frac{K^2K_1^2}{2} \]

or

\[ K_2 = \frac{KK_1}{2} \]

\[ = 9.1K_1 \] (135)

Since \( K \) is arbitrary, a value of 0.3 was selected, so

\[ K_1 = 0.3 \] (136)

and

\[ K_2 = 2.73 \] (137)

so the desired yaw angle is

\[ \psi_d = -0.3 \dot{\psi} - 2.73 \dot{y} \] (138)

The yaw angle is associated with the yaw thruster force by

\[ I_\theta \ddot{\psi} = \tau = F_{YT}l \omega \]

\[ \dot{\psi} = C_5 F_{YT} \] (139)

where

\[ C_5 = 2.73 \times 10^{-3} \] (140)

in matrix form

\[
\begin{bmatrix}
\dot{\psi} \\
\ddot{\psi}
\end{bmatrix} =
\begin{bmatrix}
0 & 1 \\
0 & 0
\end{bmatrix}
\begin{bmatrix}
\psi \\
\dot{\psi}
\end{bmatrix} +
\begin{bmatrix}
0 \\
C
\end{bmatrix} F_{YT}
\] (141)

This equation is the same as Eqn. 114 for the roll thruster, and similarly a bang-bang control exists for which the switching function
and the optimal control law is

\[
F_{yt} = \begin{cases} 
- F_{yT \text{MAX}} & SXYT > 0 \\
F_{yT \text{MAX}} & SXYT < 0 \\
F_{yT \text{MAX}} & SXYT = 0, \dot{\psi} > 0 \\
F_{yT \text{MAX}} & SXYT = 0, \dot{\psi} < 0 \\
0 & \psi = 0, \dot{\psi} = 0
\end{cases}
\]

(143)

The implementation of this control law would drive the yaw angle and yaw rate to zero in the minimum time, but the directional control problem requires that the yaw angle be equal to the desired angle, \( \psi_d \). This can be accomplished by shifting the switching curve by the amount \( \psi_d \).

\[
SXYT = \psi - \psi_d + \frac{\dot{\psi} | \dot{\psi} |}{2 C F_{yT \text{MAX}}}
\]

(144)

This change in the switching curve means that the yaw rate and the quantity \( (\psi - \psi_d) \) will be driven to zero in the minimum time, or \( \psi \) will equal \( \psi_d \).
Chapter V
The Computer Program and Simulation Results

General

The EASY Dynamic Analysis Program to Aircraft Modelling (Ref. 5) formed the major portion of the computer analysis and simulation and the air cushion system was modelled by the Foster-Miller model, as described in Chapter III. The computer program is listed in Appendix B. In brief, the EASY program provided the means for an analysis of the six degree of freedom rigid body dynamics of the Jindivik drone and the Foster-Miller model estimated the ground forces and moments transferred by the air cushion to the airframe. Additional FORTRAN was used to reflect the Jindivik autopilot and the designed roll, pitch and yaw controllers in the EASY program. Simulations were performed to obtain time history comparisons of the uncontrolled and controlled aircraft models with a crosswind driving function and an initial pitch angle to simulate flying off of a 2 inch step.

The Computer Program

Figures 16, 17, and 18 show the computer block diagrams of the aircraft dynamics, the longitudinal autopilot, and the lateral autopilot, respectively. Understanding the symbology used in these figures would require considerable referral to Reference 5, but a general description of the schematics will be given here.

In Figure 16, SD performs the six degree of freedom rigid body dynamics, AV calculates the aerodynamic variables, LO calculates the longitudinal force and moment sum, and LD calculates the lateral force and moment sum. The terms FX3S2, FZ3S2, TY3S2, TY3S2, TX3S2, and TZ3S2, shown feeding into LO and LD, are the sums of the engine and external
Fig. 17. Block Diagram of Longitudinal Autopilot
Fig. 18. Block Diagram of Lateral Autopilot
(i.e. air cushion system and controller) forces and moments.

The longitudinal and lateral autopilot functions shown in Figures 17 and 18 were developed from the elevator and aileron transfer functions given in Reference 3. The maximum deflection of the control surfaces, the gearing ratios between control surfaces and servos, and the maximum slewing rates for the servos were also programmed into the model by FORTRAN.

**Air Cushion Program**

The air cushion system was programmed with ten subroutines to the EASY program and the flow chart of the subroutines is shown in Figure 19. The functions of the ten subroutines are as follows: FM is the main subroutine which calls and interacts with the remaining subroutines; it also determines the appropriate fan curve and contains the integration routine. HC, TK, SE, CO, PR, CL, S1, and SP form a set of subroutines which need the aircraft position, cushion and trunk pressures and ground profile as input parameters and they calculate various areas and volumes associated with those parameters. HC calculates the value of side trunk height for a given cushion to trunk pressure ratio. Subroutine TK takes that height and calculates trunk cross section parameters. From these parameters SE updates the trunk division parameters. Subroutine CO transforms position vectors for each trunk centre, from the vehicle frame to the ground frame, and then calculates the distance between each of the trunk segments and the ground, and it also calculates the ground coordinates above which each of the segments lie. Subroutine PR determines ground elevations (input by user) corresponding to the
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Fig. 19. Air Cushion Subroutine Flow Chart

Set Initial Conditions
TK
SE
S1

DETERMINE STATIC FAN CURVE FROM RPM

Initialize Geometry
HC
TK
SE
CO
PR
CL
S1
SP

DYNAMIC SIMULATION

Calculate (1) rate of change of cushion volume due to geometry,
(2) rate of change of cushion volume due to pressure changes,
(3) rate of change of trunk volume
TK
SE
CO
PR
CL
S1
SP

Runge-Kutta Integration
ST

END
ground coordinates generated by CO, and subroutine CL determines the hard surface clearance for each segment using (a) the ground elevation value and (b) the distance of the trunk segment from the ground. Finally, subroutine SP calculates values of different areas and volumes for each segment and adds them together to give total areas and volumes.

In addition to these seven subroutines, FM also calls subroutine ST. ST determines the value of fan flow for a given value of fan pressure rise and also calculates the forces and torques for a given ACTS orientation. Subroutine ST also incorporates all the system differential equations so that the value of the state differentials can be updated each time it is called by the Runge-Kutta integration routine. The forces and torques calculated by ST are passed to the EASY program via FM.

Simulation Results

The results of the simulation to test the controller designs of Chapter IV are shown in Tables I and II. For initial conditions of -1° in pitch, 0° in roll, and a constant 40 ft/sec crosswind, the uncontrolled model induces a roll angle and pitch angle that are lightly damped in comparison to the controlled model. Also, the restoring yaw angle is less than the controlled yaw angle and the lateral deviation, y, is subsequently greater for the uncontrolled model. The 1.5 second simulation for the uncontrolled model required 15,000 seconds of computation time and consequently the simulation was not continued to the extent
Table I - Simulation Results Uncontrolled Model

<table>
<thead>
<tr>
<th>TIME</th>
<th>PITCH</th>
<th>ROLL</th>
<th>YAW</th>
<th>ALTITUDE</th>
<th>LATERAL DEVIATION</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.0</td>
<td>-1.000</td>
<td>0.000</td>
<td>0.00</td>
<td>2.60</td>
<td>0.00</td>
</tr>
<tr>
<td>0.5</td>
<td>0.305</td>
<td>0.113</td>
<td>-0.63</td>
<td>1.52</td>
<td>3.84</td>
</tr>
<tr>
<td>1.0</td>
<td>0.451</td>
<td>0.474</td>
<td>-1.94</td>
<td>1.02</td>
<td>8.78</td>
</tr>
<tr>
<td>1.5</td>
<td>0.377</td>
<td>0.142</td>
<td>-2.51</td>
<td>1.25</td>
<td>14.87</td>
</tr>
</tbody>
</table>

Table II - Simulation Results Controlled Model

<table>
<thead>
<tr>
<th>TIME</th>
<th>PITCH</th>
<th>ROLL</th>
<th>YAW</th>
<th>ALTITUDE</th>
<th>LATERAL DEVIATION</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.0</td>
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<td>0.000</td>
<td>0.00</td>
<td>2.60</td>
<td>0.00</td>
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<td>0.5</td>
<td>0.273</td>
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<td>1.71</td>
<td>3.83</td>
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<tr>
<td>1.0</td>
<td>0.282</td>
<td>0.072</td>
<td>-2.05</td>
<td>1.25</td>
<td>8.77</td>
</tr>
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<td>0.063</td>
<td>0.023</td>
<td>-3.82</td>
<td>2.32</td>
<td>14.68</td>
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<tr>
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<td>0.038</td>
<td>0.005</td>
<td>-5.99</td>
<td>2.31</td>
<td>21.23</td>
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<tr>
<td>2.5</td>
<td>0.045</td>
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<td>-8.52</td>
<td>2.27</td>
<td>27.87</td>
</tr>
<tr>
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<td>0.052</td>
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<td>2.32</td>
<td>43.74</td>
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<tr>
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<td></td>
<td>-21.10</td>
<td>2.32</td>
<td>46.22</td>
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<td>15.69</td>
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</table>
Chapter VI

Conclusions and Recommendations

The National Aeronautics and Space Administration (NASA) has accepted the Foster-Miller program as a valid air cushion model; however, in the course of the analysis of this thesis, several problems arose which prevent this program from becoming an effective design tool. The primary problem is the excessive computation time required for dynamic simulation when it is incorporated with the EASY Dynamic Analysis Program. The Fourth Order Runge-Kutta integration routine used in the air cushion model requires a time increment of 0.001 seconds for numerical stability. Since this integration routine is the prime reason for the excessive computation time, it is recommended that the routine be changed or augmented to reduce computation time.

The air cushion model assumes that the trunk is an elliptical shape rather than the actual shape, in which the aft end is 10% wider than the fore end. This discrepancy impinges on trunk and cushion volumes and areas, pitching and rolling moments, clearance and gap areas, etc. In other words, it requires considerable evaluation and extensive modification and verification of the program to change trunk shapes. It is recommended that Foster-Miller Associates be asked to modify their model to accommodate different trunk shapes as future designs may require.
The air cushion model has no provision to orient the trunk orifices other than perpendicular to the trunk surface. In fact, the Jindivik trunk orifices are drilled inward at a 45° angle to produce more cushion pressure in the region of trunk contact. Some adjustment should be made to the model to allow this orifice orientation as a design parameter. Also, the model uses a single curve to describe the fan characteristics of outflow vs. drive pressure, but the actual characteristics depend on more variables; hence, a fan "map" is required to replace the single curve and adequately describe the fan during all phases of its operation.

A weak part of the computer simulation is the evaluation of the Jindivik Stability Derivatives due to the fact that the static wind tunnel data was extrapolated from the Recovery Trunk Data and was suspect from the beginning of the analysis. Consequently, it is recommended that wind tunnel tests be conducted in a moving belt tunnel with the takeoff trunk and with measurement of the rate variables p, q, and r. Barring this option, the development of the derivatives should be reviewed and amended with the use of more sophisticated data reduction techniques. Once the computer program, shown in Appendix B, is changed to encompass the previous recommendations, it can be used to define the following parameters:

a) Operational limits and directions of crosswinds.

b) A "ground roughness" criteria above which the aircraft becomes unstable.
c) A flap deflection schedule to provide minimum takeoff distance within pitch stability. The present two flap settings would provide a step input to pitch and hence should be changed.

d) All of the above for different vertical thruster sizes and locations.

This thesis has integrated the EASY Dynamic Analysis Program and a truncated version of the Foster-Miller air cushion model to simulate an air cushion vehicle during takeoff. During the process of that integration and simulation, some major deficiencies in the Foster-Miller model have been highlighted. This thesis has also developed and demonstrated a technique to control bang-bang thrusters on the wing tips and a bang-bang thrust deflector on the tail section. Complete verification of the controller design was not possible due to the large computer resources that would have been required, but the results do show the control trends that are expected. The application of wing tip and yaw thrusters to other air cushion aircraft should provide comparable results. Also, these thrusters could be used on Vertical or Short Takeoff and Landing (V/STOL) aircraft because these aircraft also have marginal stability and require control enhancement in the low speed range.
AFIT/GE/EE/77D-43

Bibliography


Appendix A

Graphs of Aerodynamic Coefficients

Fig. A-1  Lift Coefficient Versus Angle of Attack
Fig. A-2  Pitching Moment Coefficient Versus Angle of Attack
Fig. A-3  Drag Coefficient Versus Angle of Attack
Fig. A-4  Side Force Coefficient Versus Sideslip Angle
Fig. A-5  Side Force Coefficient Versus Sideslip Angle – For Angle of Attack = 0.1 Deg
Fig. A-6  Roll Moment Coefficient Versus Sideslip Angle – For Angle of Attack = 2.2 Deg
Fig. A-7  Roll Moment Coefficient Versus Sideslip Angle – For Angle of Attack = 4.3 Deg
Fig. A-8  Roll Moment Coefficient Versus Sideslip Angle – For Angle of Attack = 6.4 Deg
Fig. A-9  Yaw Moment Coefficient Versus Sideslip Angle – For Angle of Attack = 0.1 Deg
Fig. A-10 Yaw Moment Coefficient Versus Sideslip Angle – For Angle of Attack = 2.2 Deg
Fig. A-11 Yaw Moment Coefficient Versus Sideslip Angle – For Angle of Attack = 4.3 Deg
Fig. A-12 Yaw Moment Coefficient Versus Sideslip Angle – For Angle of Attack = 6.4 Deg
FIG. A-1 LIFT COEFFICIENT VERSUS ANGLE OF ATTACK
FIG. A-2  PITCHING MOMENT COEFFICIENT VERSUS ANGLE OF ATTACK
FIG. A-3 DRAG COEFFICIENT VERSUS ANGLE OF ATTACK
FIG. A-4  SIDE FORCE COEFFICIENT VERSUS SIDESLIP ANGLE
FIG. A-5 ROLL MOMENT COEFFICIENT VERSUS SIDESLIP ANGLE
FIG. A-6  ROLL MOMENT COEFFICIENT VERSUS SIDESLIP ANGLE
FIG. A-7 ROLL MOMENT COEFFICIENT VERSUS SIDESLIP ANGLE

68
FIG. A-8 ROLL MOMENT COEFFICIENT VERSUS SIDESLIP ANGLE

ROLL MOMENT COEFFICIENT

--- HIND TUNNEL
- - - CURVE FIT

FOR ANGLE OF ATTACK = 0.4 DEG
FIG. A-9 YAW MOMENT COEFFICIENT VERSUS SIDESLIP ANGLE
FIG. A-11 YAW MOMENT COEFFICIENT VERSUS SIDESLIP ANGLE
FIG. A-12 YAW MOMENT COEFFICIENT VERSUS SIDESLIP ANGLE
Appendix B

Computer Program Listing
MODEL DESCRIPTION

INITIAL ACTS (CONTROLLED)

ADD STATES=FLFM,PTFM,POM,PGFM

ADD PARAMETERS=ALT,CL,XL,CS,R,匈,SPC,FF,R

ADD VARIABLES=HODS,HE SE

C

TBLN 1L 30

FORTRAN STATEMENTS

C ALL ANGLES IN DEGREES

C

SUBS=AV=100

C

SUBS=AV=506

C

SUBS=AV=506

C

SUBS=AV=506

C

SUBS=AV=506

C

SUBS=AV=506

C

SUBS=AV=506

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SUBS=AV=506

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SUBS=AV=506

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SUBS=AV=506

C

SUBS=AV=506

C

SUBS=AV=506

C

SUBS=AV=506

C

SUBS=AV=506

C

SUBS=AV=506
ANGLE OF ATTACK IS LESS THAN 3.0

180 GO TO CC14=-5.14E-4*ABS(BE)**2+2.37E-4*ABS(BE)**4+G 

CYAW=-4.52E-4*ABS(BE)**2+2.37E-4*ABS(BE)**4+G 

GO TO 2000

ANGLE OF ATTACK IS LESS THAN 5.0

190 GO TO CD14=-4.12E-1*ABS(BE)**4+2.32E-4*ABS(BE)**2-2.74E-6*ABS(BE)**4+G

XY(BR) 

IF(ABS(16.6) . LT. 1) GO TO 2000

IF(REL.47.1) GO TO 2000

CYW=-1.91E-4*ABS(BE)**4+2.32E-4*ABS(BE)**2-2.74E-6*ABS(BE)**4+G

GO TO 2000

ANGLE OF ATTACK IS GREATER THAN 5.0

195 GO TO CD14=-2.15E-3*ABS(BE)**2-1.72E-3*ABS(BE)**4+G 

XY(BR) 

IF(ABS(16.6) . LT. 1) GO TO 2000

IF(REL.47.1) GO TO 2000

CURL=1.14E-3*ABS(BE)**2+2.55E-3

GO TO 2000
AFIT/GE/EE/77D-43

2000. CONTINUE

C DERIVATION OF DIRECTIONAL DERIVATIVES (STABILITY AXIS)

XX = 0.5*(SIN(A1)-COS(A1))*COS(A2)
ZD = CL*QS*TS*AR - COS*TS*AR
MD = CH*C*S
M0 = T1*0*AR*2 - 2.43*0*AR*AR*AR + 4.03*0*AR*AR*AR*AR
3D = 1*AR*TS*AR
HA = M1*0*AR - AR + AR + 2 - AR + AR + AR + AR
V1 = YP*0*AR - YP*0*AR
YR = YP*0*AR - YP*0*AR
X0 = ZD*0*AR
L0 = AR - AR + AR - AR + AR + AR - AR + AR + AR
X1 = AR - AR + AR - AR + AR + AR - AR + AR + AR
X2 = AR - AR + AR - AR + AR + AR - AR + AR + AR
H0 = N0*0*AR*AR
X0 + (L0*0*AR*AR*AR - L0*0*AR*AR*AR + N0*0*AR*AR*AR - N0*0*AR*AR*AR)
X1 + (L0*0*AR*AR*AR - L0*0*AR*AR*AR + N0*0*AR*AR*AR - N0*0*AR*AR*AR)
X2 + (L0*0*AR*AR*AR - L0*0*AR*AR*AR + N0*0*AR*AR*AR - N0*0*AR*AR*AR)

MD = HQ*0*AR + NO

CONTINUE
2 PRINT 3
3 FORMAT(*OFF LOWER END OF FAN MAP*)
4 GO TO 199
5 PRINT 5
6 FORMAT(*OFF UPPER END OF FAN MAP*)
7 GO TO 199

C FAN CURVE FOR STATIC ITERATIONS

.14 QFAN=15.06 -1.343 *PFAN+2.345-3 *PFAN**2+1.69E-5 *PFAN
.14 QFAN=15.06 -1.343 *PFAN+2.345-3 *PFAN**2+1.69E-5 *PFAN
.12 QFAN=11.23 -1.7E-3 *PFAN+1.7E-3 *PFAN**2+1.7E-3 *PFAN
.11 QFAN=10.62 -1.194 *PFAN+1.7E-3 *PFAN**2+1.7E-3 *PFAN
.10 QFAN=9.72 -9.77 *PFAN+7.1E-4 *PFAN**2+7.1E-4 *PFAN
.09 QFAN=9.47 -9.77 *PFAN+7.1E-4 *PFAN**2+7.1E-4 *PFAN
.08 QFAN=12.39 -9.144 *PFAN+1.5E-3 *PFAN**2+1.5E-3 *PFAN
.07 QFAN=12.39 -9.144 *PFAN+1.5E-3 *PFAN**2+1.5E-3 *PFAN
.06 QFAN=12.39 -9.144 *PFAN+1.5E-3 *PFAN**2+1.5E-3 *PFAN
.05 QFAN=12.39 -9.144 *PFAN+1.5E-3 *PFAN**2+1.5E-3 *PFAN
.04 QFAN=12.39 -9.144 *PFAN+1.5E-3 *PFAN**2+1.5E-3 *PFAN
.03 QFAN=12.39 -9.144 *PFAN+1.5E-3 *PFAN**2+1.5E-3 *PFAN
.02 QFAN=12.39 -9.144 *PFAN+1.5E-3 *PFAN**2+1.5E-3 *PFAN
.01 QFAN=12.39 -9.144 *PFAN+1.5E-3 *PFAN**2+1.5E-3 *PFAN

C CHANGE FROM MASS FLOW TO VOLUME FLOW

110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3
110 QFAN=QFAN/32.3

C CONVERT DEGREES TO RADIANS

C OBTAIN INITIAL VALUE OF OCCUP AND INITIALIZE GEOMETRY

C MINIMUM TRUNK HEIGHT IS MAKING BLATTER HEIGHT

80
CALL TK1ISHK, PH2TK, RZ TK, R1 TK, PH1TK, L1 TK, L2 TK, HT, A, E, L(L5)

CALL TK1ISHK, PH2TK, RZ TK, R1 TK, PH1TK, L1 TK, L2 TK, HT, A, E, L(L5)

CALL STIFF ST, T2 ST, T1 ST, T0 ST, T1 TK, PTL, TPL, VTK2, DF X, DKV, PTK, DKV.

CALL STIFF ST, T2 ST, T1 ST, T0 ST, T1 TK, PTL, TPL, VTK2, DF X, DKV, PTK, DKV.

CALL STIFF ST, T2 ST, T1 ST, T0 ST, T1 TK, PTL, TPL, VTK2, DF X, DKV, PTK, DKV.

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CALL SPIVTK2,VMHS,ACM,2,PHTK,RZ TK,02IS1,8,5 SE A1 S1,A2 S1,OK S
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RZ TK,ITY5,0.5,M,MY,VM,VTSLS,ACIS1,LSJ
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CALL SPIVTK2,VMHS,ACM,2,PHTK,RZ TK,02IS1,8,5 SE A1 S1,A2 S1,OK S
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RZ TK,ITY5,0.5,M,MY,VM,VTSLS,ACIS1,LSJ
CALL SPIVTK2,VMHS,ACM,2,PHTK,RZ TK,02IS1,8,5 SE A1 S1,A2 S1,OK S
11,R2S1,SMTS1,PHTK,PR TK,0X SE,PE SE,HY R1 TK,PHTK,ICLFS,L1 TK,L
RZ TK,ITY5,0.5,M,MY,VM,VTSLS,ACIS1,LSJ
CALL SPIVTK2,VMHS,ACM,2,PHTK,RZ TK,02IS1,8,5 SE A1 S1,A2 S1,OK S
11,R2S1,SMTS1,PHTK,PR TK,0X SE,PE SE,HY R1 TK,PHTK,ICLFS,L1 TK,L
RZ TK,ITY5,0.5,M,MY,VM,VTSLS,ACIS1,LSJ
CALL SPIVTK2,VMHS,ACM,2,PHTK,RZ TK,02IS1,8,5 SE A1 S1,A2 S1,OK S
AFIT/GE/EE/77D-43

ISHAPE = ISMTK
IF (ISHAPE .EQ. 29) GO TO 199
CALL SECITY2, 1E, SE, DX, 27, TX, XCY, DCL5E, ACISE, ZCISE, ICLFM, RX, T
TK, PHMTK, OZISL, LS, M, N, CTG5S)
CALL COISL (O, KCS5E, ZC5E, Y, 6, AC5E, SIE, PH, THE, CG, FF, GG)
CALL PREY (90)
CALL CLGNC, ICM, PHE, THE, SLAC0, YG PR1
RMOFP=HQO
CALL S1OZE5I (41 S1, A2 S1, S1, OI S1, 0X5I, 0E51, 0I51, L251, R1II
15, RII1, N251, M251, PHMTK, 12 PK, TK, DX S1, FX SE, HY, RI TK, PHMTK, ICLFM, L5, TK1
12 TK, ITYSE, 3.4, 3.N, 4, VTIS1, ACIS1, LS)
CALL SPIKTSZ, ZCS5E, 3CH5E, PHMTK, 12 TK, OZIS1, 1E, SE, A1 S1, A2 S1, OX S
15, SE S11, ITYSE, YSH5, MY YS1, S1, 0X51, 0E51, 0I51, L251, R1II
1251, L251, R1II1, P1II1, P1I51, ZC5E, EN151, ICX52, JLS52, M, N, NLH1, HTZ5S
3H, VCC, R1 TK, AG52, AT52, AT52, AC52, AC52, AC52, AC52, JG55, WTIS1, ACIS1
,KACS2, AT151, AT52, P152, KCH52, 2CHS2, 2TS52, 2TS52, L1, TK)
DVK (VENKS2-VCHS2) / (FCHS2-KRM)
PR4=MAX (O, O, MIN (1, 12, (FCH/0.1)))

********* CALCULATE DVTK ************

CALL HCMT (PR4)

SHAPE=MTK

CALL TKIS5TK, PMT5K, R2 TK, PI TK, PMT5K, L1 TK12, TK, MY, A2, P.L, LS)

ISHAPE = ISMTK
IF (ISHAPE .EQ. 29) GO TO 199
CALL SECITY2, 1E, SE, DX, 27, TX, XCY, DCL5E, ACISE, ZCISE, ICLFM, RX, T
TK, PHMTK, OZISL, LS, M, N, CTG5S)
CALL COISL (O, KCS5E, ZC5E, Y, 6, AC5E, SIE, PH, THE, CG, FF, GG)
CALL PREY (90)
CALL CLGNC, ICM, PHE, THE, SLAC0, YG PR1
RMOFP=HQO
CALL S1OZE5I (41 S1, A2 S1, S1, OI S1, 0X5I, 0E51, 0I51, L251, R1II
15, RII1, N251, M251, PHMTK, 12 PK, TK, DX S1, FX SE, HY, RI TK, PHMTK, ICLFM, L5, TK1
12 TK, ITYSE, 3.4, 3.N, 4, VTIS1, ACIS1, LS)
CALL SPIKTSZ, ZCS5E, 3CH5E, PHMTK, 12 TK, OZIS1, 1E, SE, A1 S1, A2 S1, OX S
15, SE S11, ITYSE, YSH5, MY YS1, S1, 0X51, 0E51, 0I51, L251, R1II
1251, L251, R1II1, P1II1, P1I51, ZC5E, EN151, ICX52, JLS52, M, N, NLH1, HTZ5S
3H, VCC, R1 TK, AG52, AT52, AT52, AC52, AC52, AC52, AC52, JG55, WTIS1, ACIS1
,KACS2, AT151, AT52, P152, KCH52, 2CHS2, 2TS52, 2TS52, L1, TK)

199 CONTINUE

C CONVERT RADIANS TO DEGREES

THETA = THETA/RADIAN

PHI = PHI/RADIAN

DTHETA = DTHETA/RADIAN

RETURN XEN

SUBROUTINE SECITY, R4IA, 3ELX, TX, DCTRL, XCHT, THE, R2, PHI

DIVISION OF THE TRUNK INTO SEGMENTS

PEAL LS

DIMENSION 2CH (32), XCH (32), IYTP (32), OFLT (32), ISFG (32), XCHT (32), ZCM (32)

DATA PI, 3.1415926537

STOP = 32

C IF FIRST CALL, COMPUTE PARTIAL TERMS AND NUMBER SEGMENTS

IF (ICALL .NE. 0) GO TO 20

30 KLSM = 31 LS

C ACTA IS CURRENT SEGMENT ARC ANGLE

ACTA = I/2, PLOT

C DELX IS STRAIGHT SEGMENT LENGTH

DELX = LS/FLT (2)*

RETURN XEN

C NUMERICAL OF 2CH=HTS ACCORDING TO THEIR POSITION IN THE TRUNK

00 31 1=1, STOP

84

BEST AVAILABLE COPY
C STRAIGHT SEGMENT
6  IYYP(1)=0
   XCI(1)=XLH+(FLOAT(1-3*N+2*N-1)+0.5)*DELX
   ZCI(1)=ZCI(1)+0.5
   XCM(1)=XCI(1)
   ZCM(1)=ZCI(1)+0.5
   GO TO 8

C STRAIGHT SEGMENT
7  IYYP(1)=0
   XCI(1)=(FLOAT(1-3*N+2*N-1)+0.5)*DELX
   ZCI(1)=ZCI(1)+0.5
   XCM(1)=XCI(1)
   ZCM(1)=ZCI(1)+0.5
   GO TO 8

C CURVE SEGMENT
C IF NOT INITIAL CALL SKI CALCULATIONS
8  IF(ICALL) 9,880,9

880  IYYP(1)=1
   DELTA(I)=FLOAT(-T+H-L-N-I+1)+0.5
   SINC SINC DELTA(I)
   XCI(I)=XCI(I)+0.5*DELTA(I)
   ZCI(I)=ZCI(I)+0.5*DELTA(I)
   XCM(I)=XCI(I)
   ZCM(I)=ZCI(I)+0.5

9  CONTINUE
RETURN
END

C TRUNK GEOMETRY CALCULATIONS
REAL L,L1,L2,LS
RTOI=1
IF(NY(12)=0.0) GO TO 111

C***********************************************************************
C ITERATION FOR R
C COMPUTE INNER RADIUS IF CURVATURE
C***********************************************************************
C
C ITERATION LOOP FOR L2,L3,L4,L5
102  GO TO 102
   PH1=ASIN(1.0/SINH(L1))
   PH2=ASIN(1.0/SINH(L2))

C COMPUTE OUTER RADIUS OF CURVATURE
C
   X1=((4*PH1*PH2)**2+(9*NY(12))**2)**0.5
   YS=A-RA+2552*PH2
   IF (XS.LE.0.0) PH1=6.2831952-PH1
   L2=PH1/PH2

C R25 IC RESULTANT RADIUS FOR COMPUTE L2 IN ITERATION
   IF(L2>PH1) L2=PH1
C TEST IF TOLERANCE GT TOLERANCE
   IF((ABS(L2-R25))>TOLER1) GO TO 90
   R2=L2-R25
102  CONTINUE

C***********************************************************************
C ITERATED 50 TIMES WITHOUT SUCCESS,ERROR RETURN
C***********************************************************************
111  CONTINUE
   WRITE(6,9011)
   FORMAT(1X,8,E11.8)
   ISTATE=0
   RETURN

C TRUNK 0=RETURN
   RETURN
50   L1=L2=L2
   ISTATE=1
   RETURN
     
BEST AVAILABLE COPY
END

SUBROUTINE S1(D2, A1, A2, S, X234H1, OA, MA, BETA, D1, L1, L2, R1, R2,
K, SINHR, PHI, R2, D2, L1, L2, X1, X2, R1, R2, RC, H1, PHI, CAL1, L1, L2, L1, L2, D1, D2, KN, NV
L1, ACH1, L1)

C INITIAL ASSESSMENT OF AREAS, VOLUMES ASSUMING
C NO GROUND CONTACT
C
REAL L1, L2, L1, L2, LS

Q: DIMENSION ITYP(121), AK1(122), VT1(122), ACHI(32)

C COMPUTE GONNLEY TERMS

S: S = 2 * PH1 * 2 * PI2 * PI2

D2 = D2 + S * SINHR

U2 = D2 * L2

BD2 = 3 * ETA * D2 * 0.5

X = 2 * (A - SINHR) / (3 * PI2)

C COMPUTE AREAS OF TRUNK SECTORS

A1 = PI2 / 2 * A2 * 2 * 2

A2 = (R2 - H1) / 2.0 * SINHR

A = A1 / A2

A3 = PI2 / 2.0 * 2 * 2

A4 = (A - SINHR) / (3 * PI2)

X1 = SINHR + 0.0 * (SINHR / 2.5) * (R2 / 3.0 * PI2)

X2 = 0.5 * L2 * PI2

A5 = A1 * A1 / 2.0

X5 = SINHR + 0.0 * (SINHR / 2.5) * (R2 / 3.0 * PI2)

X6 = A1 * A1 / 2.0

Y = X1 + A2 * 2 * PI2 / 4 * PI2 * X5

IF (CAL1 .GT. 0) GO TO 20

C, SAVE TRUNK GEOMETRY TERMS FOR END TRUNK CALCULATIONS

RI = 1

R1 = 2

PHI1 = PHI1

PHI2 = PHI2

L1 = L1

L2 = L2

A11 = A11

A21 = A2

SINPH1 = SINPH1

SINPH2 = SINPH2

X1 = X1

X2 = X2

AHA1 = A1 + 2

D2 = D2

S = 5

BET2 = BETA + 1

X2 = X1 + 1 + 2 * A2 / 4 * PI2 / 4 * PI2

DPA1 = PI2 * (A11 + 1 + 2 * A2) / 4 * PI2 / 4 * PI2

RCPH1 = RCPH1

D2 = D2

BETA = BETA + 1

C, CONTINUE

C, COMPUTE TRUNK SEGMENT AREA, VOLUME, CUSHION AREA

DO 103 I = 1, NSTOP

IF (ITYP(I) .EQ. 1) GO TO 112

C, STRAIGHT PART OF TRUNK

111 AK1 = 11

VT1 = PI2 * (A11 + 11)

ACHI = ACHI + 30

GO TO 102

C, CURVATURE PART OF TRUNK

112 IF (ICALL .GT. 0) GO TO 102

AK1 = 11

X = XA / AK1(1)
SUBROUTINE HCII(K)
C DIMENSION 3L4(32),1.G(32),ZG(32),XG(32)
C CALCULATION OF TRUNK GROUND CLEARANCE FOR EACH SEGMENT
ICLNS=ICLN
COSCS=COS(IPIES)*COS(THETA)
C CALCULATE SEGMENT GAP
DO 161 I=1,NSTOP
YGH(I)=0.0
161 CONTINUE
ICLN=ICLNS
RETURN
END
SUBROUTINE CLYSM(I,ICLN,PHII,THI,SLY,SGY)
DIMENSION 3L4(32),1.G(32),ZG(32)
NSTOP=32
C CALCULATION OF TRUNK GROUND CLEARANCE FOR EACH SEGMENT
ICLNS=ICLN
COSCS=COS(IPIES)*COS(THETA)
C CALCULATE SEGMENT GAP
DO 161 I=1,NSTOP
YGH(I)=0.0
161 CONTINUE
ICLN=ICLNS
RETURN
END
SUBROUTINE CSM(L,ICLN,PHII,YCG,XYC,SGY,SGY2,THETA,FF,GG)
DIMENSION 3L4(32),1.G(32),ZG(32),XG(32)
C THIS SUBROUTINE CALCULATES X AND Z COORDINATES OF THE GROUND
C POINT CORRESPONDING TO EACH SEGMENT, FOR A PARTICULAR ACLS
C ORIENTATION
DIMENSION 3L4(32),1.G(32),ZG(32),XG(32),YGC(32)
NSTOP=32
C CALL SUBROUTINE SPECIAL TRANSFORMATION
C MATRIX TRANSFORMS 4 VECTORS FROM VEHICLE FRAME TO INERTIAL FRAME
C C
C C
C C
C CALCULATE TRANSCENDENTALS
CSII=COS(IPIE)
SPHII=COS(IPIE)*PHII
CMTAE=COS(THETA)
PIII=SIN(IPIE)
SFPHII=PHII
STHETA=SIN(THETA)
C
C COMPUTE TRANSFORMATION MATRIX ELEMENTS
B11=CSII*CMTAE*SPHII*PIII
B12=CSII*SMTAE*SPHII*PIII
B13=CSII*SPHII*PIII
B21=SPHII*CSII*CMTAE*PIII
B22=SPHII*SMTAE*PIII
B23=SPHII*PIII
B31=SMTAE*CMTAE
B32=SMTAE
B33=SMTAE*CMTAE
C
C DO LOOP OF ALL SEGMENTS TO GROUND POSITION
DO 104 I=1,NSTOP
XGCI4=EFC(I,THII)
C

BEST AVAILABLE COPY
ZCPPF(zCY(1)-FF)

C CALCULATE VECTOR DA FOR SEGMENT
SL4(I)+XCY(XCY(I)+SL4(I)+XCY(m)+SL4(I)+XCY(m)

C CALCULATE X-GROUND COORDINATE
XCY(I)+XCY(I)+SL4(I)+XCY(m)+XCY(I)+XCY(m)

C CALCULATE Z-GROUND COORDINATE
ZCY(I)+ZCY(I)+SL4(I)+ZCY(m)+ZCY(I)+ZCY(m)

SUBC CONTINUE RETURN

END

SUBROUTINE PRE(YG)

C USER SPECIFIED GROUND PROFILE.
C ELEVATION YG(I) IS EXPRESSED AS A FUNCTION OF X AND Z COORDINATES
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).

DO 105 I=1,NSSTOP

C SET FOR FLAT TerRAIN

SUBC CONTINUE RETURN

C USER SPECIFIED GROUND PROFILE.
C ELEVATION YG(I) IS EXPRESSED AS A FUNCTION OF X AND Z COORDINATES
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).
C OF GROUND POINT I.I.E., XG(I) AND ZG(I).

CALL CONINTG, YG(I),XG(I), ZG(I)

C CALCULATION OF AREAS AND VOLUMES ASSOCIATED WITH ACLS KNOWING ITS
C ORIENTATION
C ORIENTATION
C ORIENTATION
C ORIENTATION
C ORIENTATION
C ORIENTATION

REAL LS(LY),LS(LY),LS(LY)

SUBC COMPUTE PARTIAL TERMS
SIN=Z(SIN(2));
SINH=Z(SINH(2));

DO 200 I=1,NSSTOP

C TEST FOR TRUNK SEGMENT, WHETHER CURVED OR STRAIGHT
C TEST FOR TRUNK SEGMENT, WHETHER CURVED OR STRAIGHT
C TEST FOR TRUNK SEGMENT, WHETHER CURVED OR STRAIGHT
C TEST FOR TRUNK SEGMENT, WHETHER CURVED OR STRAIGHT
C TEST FOR TRUNK SEGMENT, WHETHER CURVED OR STRAIGHT

C STRAIGHT PART OF TRUNK
C STRAIGHT PART OF TRUNK
C STRAIGHT PART OF TRUNK
C STRAIGHT PART OF TRUNK
C STRAIGHT PART OF TRUNK

C CALCULATE CURVATURE SEGMENT INITIAL VOLUME
C CALCULATE CURVATURE SEGMENT INITIAL VOLUME
C CALCULATE CURVATURE SEGMENT INITIAL VOLUME
C CALCULATE CURVATURE SEGMENT INITIAL VOLUME
C CALCULATE CURVATURE SEGMENT INITIAL VOLUME

C CALCULATE CURVATURE SEGMENT GAP AREA
C CALCULATE CURVATURE SEGMENT GAP AREA
C CALCULATE CURVATURE SEGMENT GAP AREA
C CALCULATE CURVATURE SEGMENT GAP AREA
C CALCULATE CURVATURE SEGMENT GAP AREA

GO TO 10

89

BEST AVAILABLE COPY
C CURVED PART OF TRUNK

C

C CURVED PART OF TRUNK

C

C PART 2 R VALUE CALCULATIONS

C

C TEST FOR GROUND CONTACT AT EACH SEGMENT

C CONTINUE

C SEGMENT FOR CONTACT

C IF (TYTYP(I).EQ.1.AND.YGM(I).LE.MY) GO TO 14

C IF (TYTYP(I).EQ.0.AND.YGM(I).LE.MY) GO TO 23

C NO GROUND CONTACT

C SET CONTACT AND REMOVE TERMS TO ZERO

C ATKR(I)=0.0

C ACHR(I)=0.0

C VCKR(I)=0.0

C VCMR(I)=0.0

C ACHR(I)=0.0

C ATKR(I)=0.0

C PER(I)=0.0

C SET DISTANCES L7 TO FREE TRUNK VALUES

C XCH(I)=XCH(I)

C ZCH(I)=ZCH(I)

C ZTK(I)=ZCH(I)

C RTK(I)=XCH(I)

C COMPUTE TRUNK-CUSHION-ATMOSPHERE BLED AREAS

C IF (TYTYP(I).EQ.16.AND.1.6)

C CONTINUE

C NO CONTACT STRAIGHT SECTIONS

C ATKCH(I)=FLOAT(IPIX((LZ-LEX)/SH+S.0)*NH)*AD5

C ATKCH(I)=NH*AD5*ATKCH(I)

C GO TO 17

C CONTINUE

C NO CONTACT CURVED SECTIONS

C ATKCH(I)=FLOAT(IPIX((LZ-LEX)/SH+S.0)*NH)*AD5

C ATKCH(I)=NH*AD5*ATKCH(I)

C GO TO 17

C TRUNK GROUND CONTACT

C

C CURVED PART OF TRUNK

C CALCULATE DEFORMATION ANGLES FOR SEGMENT

C

C COMPUTE PARTIAL TERMS

C

C COMPUTE NORMALIZED TERMS

C

C COMPUTE NORMALIZED TERMS
AFIT/GE/Et/77D-43

C COMPUTE STICK CHORDS
X6=SINPHI1+.33333*(SIN(PHI1+.5)**2)*RZ1/PHI1
C7=SINPHI1-.33333*RZ1/SINPHI1
C8=SINPHI1+.33333*RZ1/SINPHI1
X10=X9

X102=PI2.0

IF(PHI1<4.0,110) GO TO 90
C IF PHI1 GREATER THAN 90 DEGREES, SET TO 90 DEGREE
PHI1=PI2

C COMPUTE TRUNK AREA CHANGE
ATRK11=ATK1+ATR1

C COMPUTE TRUNK VOLUME CHANGE
VTRE1=VTRE+ATR1
VTRE2=VTRE1+VTRE}

C COMPUTE TRUNK VEST AREAS
ACHM11=ACHM1+ACH1
ACHM12=ACHM2+ACH1

C COMPUTE CONTACT PERTURBER
ACHM11=ACHM1+ACH1

C COMPUTE CUSHION VOLUME CHANGE
XCHM11=XCHM1+XCH1

C COMPUTE GAP AREA CHANGE
AGAP11=AGAP11+AGAP1

C DISTANCE OF STICK CENTER TO CUSHION CENTER
X01=ZH1+6.11*SINPHI1
AP1=2*Z1+2*Z2*SINPHI1

C COMPUTE STICK CENTER, OGNI

C GO TO 10
GO TO 10

C CH1=LSH+MEDI CS
CH1=LSH+MEDI CS
CH1=LSH+MEDI CS
X1=XX2+X3+X4
X1=XX2+X3+X4
X1=XX2+X3+X4

GO TO 17
GO TO 17
GO TO 17

91
C \[ \text{TRUNK GROUND CONTACT} \]
C \[ \text{STRAIGHT PART OF TRUNK} \]
C \[ \text{CONTINUE} \]
C \[ \text{COMPUTE DEFORMATION ANGLES} \]
C \[ \text{DO TRANSCENDENTALS ONLY ONCE} \]
C \[ \text{COMPUTE SECTOR CENTROIDS} \]
C \[ \text{IF PHI IS GREATER THAN 90 DEGREES, SET TO 90 DEGREES} \]
C \[ \text{CONTINUE} \]
C \[ \text{COMPUTE TRUNK AREA CHANGE} \]
C \[ \text{COMPUTE TRUNK VOLUME CHANGE} \]
C \[ \text{COMPUTE CUS-101 VOLUME CHANGE} \]
C \[ \text{COMPUTE TRUNK LIFT AREAS} \]
AFIT/GE/EE/77D-43

C COMPUTE TRUNK CONTACT PERIMETER
C COMPUTE TRUNK CONTACT AREA
C COMPUTE GAP AREA CHANGE
C COMPUTE SEGMENT CONTACT CENTER OF PRESSURE FOR CUSHION AND TRUNK
DO 10 J=1,1,NSTOP
VTK=VTK* (VTK(J)-VTK(J))
AD=AD+ (AD(J)-AD(J))
ATKCH=ATKCH+ATKCH(J)1
ATKAT=ATKAT+ATKAT(J)
VCH=VCH* (VCH(J)-VCH(J))
AGAP=AGAP+ (AGAP(J)-AGAP(J))
ATKCC=ATKCC+ATKCC(J)
ATKCH=ATKCH+ATKCH(J)
CONTINUE
AGAP=AGAP* (AGAP(J)-AGAP(J))
AD=AD+ (AD(J)-AD(J))
VTK=VTK* (VTK(J)-VTK(J))
VCH=VCH* (VCH(J)-VCH(J))
ATKCC=ATKCC+ATKCC(J)
ATKAT=ATKAT+ATKAT(J)
ATKCH=ATKCH+ATKCH(J)
END

SUBROUTINE STFY.TZ.TK.DCRY.IPP.PPLM.VPLM.VTK.OFTK.OFTK.PTK.OFCH.
DYNAMIC FAN VERSION FOR FMA

REAL MASS

C FOLLOWING SUBROUTINES ARE CALLED TO UPDATE VALUES OF
C FORCES, FORCED RHEO AND FLOWS, GIVEN THE NEW VALUES OF THE
C STATE VARIABLES

DIMENSION JERY(32), LCHP(32), ACHP(32), AGBK(32), PERF(32)

CC=1.1S*18+S*326 2CM=6.6 TOTA=0.4 80VNT=0.

CCTC=0.334 3FF=0.  

HOC=1.  

APDK=598.  

SAIFAN=672.

CENFZ=0.  

ISTOP=32

C_SUBROUTINE TO FIND FLOW AND PRESSURE VALUES DURING DYNAMIC SIMULATION

TIRNO=2.0/DO

C PLENUM TO TRUNK FLOW

SIGN=1.0

IF(PPLH=P)LT.0.01 SIGN=1.0

DPLK=SIGN*CTC+10LPK*SORT IASLT/THO++)PPLH=P)K)

C TRUNK TO CUSHION FLOW

SIGN=1.0

IF(PKCH=P)LT.0.01 SIGN=1.0

OTKC=SIGN*CTC+5ORT AMSLT/THO+P)PKH=P)CH)

C TRUNK TO ATMOSPHERE FLOW

SIGN=1.0

IF(PTKLT.0.01 SIGN=1.0

OTKT=SIGN*CTC+3RT ITHO=3)PKH=P)KATK)

C CUSHION TO ATMOSPHERE FLOW

SIGN=1.0

IF(PCPMCHLT.0.01 SIGN=1.0

CHCM=SIGN*CTC+5RT ITTHO=3)PKH=P)CHCM)

C FORCES AND TORQUES ASSOCIATED WITH A PARTICULAR ACLS ORIENTATION

C ARE CALCULATED

C CALUCULATE TRANSPORTATION ONLY ONCE

C CUSTC=COS(18H) SIN(THELAE) SINHIE) COS(SINHIE) SINHIE) (-COS(SINHIE) SINHIE) (COS(18H) SIN(THELAE)

C=0.0*4159  

C CLEAR TOTAL FORCES AND TORQUES TO ZERO

FORC=0.0

TTPX=0.0

TT(?)=0.0

TCPZ=0.0

TCPX=0.0

TOPFZ=0.0

TOPQX=0.0

TOPQT=0.0

FORC=0.0  

STOPJ=0.  

SIGDUE=0.

C FORCES AND TORQUES INDEPENDENT OF SEGMENTS INDEPENDENTLY

C HEAVY FORCES CUSHION AND TRUNK

FCP=PCK=ACH

C COMPUTE VELOCITY FOR DRAG FORCE

VEL=CSS*C+5+SINKR*CPT

SIGN=1.0

IF(V(VT)0.0 SIGN=1.0

C HEAVY DRAG FORCE

FDF=0.5*H0*PHIPPA+PHD+V**V*SIGN

C DRAG TORQUE

TOPF=FCP+5

TOPF=FDF*PFZ

C FORCEs AND TORQUES DEPENDENT ON SEGMENTS INDEPENDENTLY

C SUM INDEPENDENT SEGMENTS TO FIND TOTALS

DO 101 I=1,ISTOP

C CUSHION PRESSURE TORQUES

94
TCP1*TCP2*(XCN11-CC)*RCH*(ACM11-ACHR11)
TCP1*TCP2*(XCN11-FF)*RCH*(ACM11-ACHR11)
TCP1*TCP2*(XCN11-CC)*RCH*(ATCN11-ATCHR11)
TCP1*TCP2*(XCN11-FF)*RCH*(ATCN11-ATCHR11)
IF(XCN11.G1.0,1.G2.ATCN11.G1.0) GO TO 111
GO TO 101

111 VELT=VENT+QMP+QERT1
FORC=FORC+FORC1
TORQ=TORQ+TORQ1
IF(VELT.EQ.0.0) GO TO 101
TORQ2=TORQ1*VC1+CPT1*QTK1*ATCN11*ATCHR11)

101 CONTINUE

C SUMMATION OF FORCE AND TORQUE COMPONENTS
C TOTAL HEAVE FORCE
FORC1=VC1+QPF1+QPF1+CPT1

C TOTAL TORQUE X AXIS
TORQ1=TORQ1+TORQ1+TORQ1+TORQ1

C TOTAL TORQUE Z AXIS
TORQ1=TORQ1+TORQ1+TORQ1+TORQ1

C STATE EQUATIONS
C STATE VARIABLES***
C 11PLP1, FLIGHT PRESSURE (GAGE)
C 21PCN, CUSHION PRESSURE (GAGE)
C 31PCR,TRUNK PRESSURE (GAGE)
C 41SINKT, VERTICAL SINK RATE, POSITIVE UPWARDS
C 91VC1, CC ELEVATION
C 61DPV1, PITCH RATE, VEHICLE FRAME
C 71QMT1, ROLL RATE, VEHICLE FRAME
C 81THE1, EULIN ROLL ANGLE
C 91PE1, EULIN PITCH ANGLE
C 101SY1, EULIN YAW ANGLE (APPROX. ZERO)
C 111V1, VELOCITY OF PRESSURE RELIEF VALVE
C 121YY1, VELOCITY OF PRESSURE RELIEF VALVE
C 131QFF1, FAN AIR INSTANT ST FLOW
C 141TP1, (XCN11-FF)*RCH*(QFF1+QFT1)
C 151QFF1, FAN AIR INSTANT ST FLOW
C 161QFF1, FAN AIR INSTANT ST FLOW
C 171QFF1, FAN AIR INSTANT ST FLOW
C 181QFF1, FAN AIR INSTANT ST FLOW
C 191QFF1, FAN AIR INSTANT ST FLOW
C 201QFF1, FAN AIR INSTANT ST FLOW
C 211QFF1, FAN AIR INSTANT ST FLOW
C 221QFF1, FAN AIR INSTANT ST FLOW
C 231QFF1, FAN AIR INSTANT ST FLOW
C 241QFF1, FAN AIR INSTANT ST FLOW
C 251QFF1, FAN AIR INSTANT ST FLOW
C 261QFF1, FAN AIR INSTANT ST FLOW
C 271QFF1, FAN AIR INSTANT ST FLOW
C 281QFF1, FAN AIR INSTANT ST FLOW
C 291QFF1, FAN AIR INSTANT ST FLOW
C 301QFF1, FAN AIR INSTANT ST FLOW
C 311QFF1, FAN AIR INSTANT ST FLOW
C 321QFF1, FAN AIR INSTANT ST FLOW
C 331QFF1, FAN AIR INSTANT ST FLOW
C 341QFF1, FAN AIR INSTANT ST FLOW
C 351QFF1, FAN AIR INSTANT ST FLOW
C 361QFF1, FAN AIR INSTANT ST FLOW
C 371QFF1, FAN AIR INSTANT ST FLOW
C 381QFF1, FAN AIR INSTANT ST FLOW
C 391QFF1, FAN AIR INSTANT ST FLOW
C 401QFF1, FAN AIR INSTANT ST FLOW
C 411QFF1, FAN AIR INSTANT ST FLOW
C 421QFF1, FAN AIR INSTANT ST FLOW
C 431QFF1, FAN AIR INSTANT ST FLOW
C 441QFF1, FAN AIR INSTANT ST FLOW
C 451QFF1, FAN AIR INSTANT ST FLOW
C 461QFF1, FAN AIR INSTANT ST FLOW
C 471QFF1, FAN AIR INSTANT ST FLOW
C 481QFF1, FAN AIR INSTANT ST FLOW
C 491QFF1, FAN AIR INSTANT ST FLOW
C 501QFF1, FAN AIR INSTANT ST FLOW
C 511QFF1, FAN AIR INSTANT ST FLOW
C 521QFF1, FAN AIR INSTANT ST FLOW
C 531QFF1, FAN AIR INSTANT ST FLOW
C 541QFF1, FAN AIR INSTANT ST FLOW
C 551QFF1, FAN AIR INSTANT ST FLOW
C 561QFF1, FAN AIR INSTANT ST FLOW
C 571QFF1, FAN AIR INSTANT ST FLOW
C 581QFF1, FAN AIR INSTANT ST FLOW
C 591QFF1, FAN AIR INSTANT ST FLOW
C 601QFF1, FAN AIR INSTANT ST FLOW
C 611QFF1, FAN AIR INSTANT ST FLOW
C 621QFF1, FAN AIR INSTANT ST FLOW
C 631QFF1, FAN AIR INSTANT ST FLOW
C 641QFF1, FAN AIR INSTANT ST FLOW
C 651QFF1, FAN AIR INSTANT ST FLOW
C 661QFF1, FAN AIR INSTANT ST FLOW
C 671QFF1, FAN AIR INSTANT ST FLOW
C 681QFF1, FAN AIR INSTANT ST FLOW
C 691QFF1, FAN AIR INSTANT ST FLOW
C 701QFF1, FAN AIR INSTANT ST FLOW
C 711QFF1, FAN AIR INSTANT ST FLOW
C 721QFF1, FAN AIR INSTANT ST FLOW
C 731QFF1, FAN AIR INSTANT ST FLOW
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C 771QFF1, FAN AIR INSTANT ST FLOW
C 781QFF1, FAN AIR INSTANT ST FLOW
C 791QFF1, FAN AIR INSTANT ST FLOW
C 801QFF1, FAN AIR INSTANT ST FLOW
C 811QFF1, FAN AIR INSTANT ST FLOW
C 821QFF1, FAN AIR INSTANT ST FLOW
C 831QFF1, FAN AIR INSTANT ST FLOW
C 841QFF1, FAN AIR INSTANT ST FLOW
C 851QFF1, FAN AIR INSTANT ST FLOW
C 861QFF1, FAN AIR INSTANT ST FLOW
C 871QFF1, FAN AIR INSTANT ST FLOW
C 881QFF1, FAN AIR INSTANT ST FLOW
C 891QFF1, FAN AIR INSTANT ST FLOW
C 901QFF1, FAN AIR INSTANT ST FLOW
C 911QFF1, FAN AIR INSTANT ST FLOW
C 921QFF1, FAN AIR INSTANT ST FLOW
C 931QFF1, FAN AIR INSTANT ST FLOW
C 941QFF1, FAN AIR INSTANT ST FLOW
C 951QFF1, FAN AIR INSTANT ST FLOW
C 961QFF1, FAN AIR INSTANT ST FLOW
C 971QFF1, FAN AIR INSTANT ST FLOW
C 981QFF1, FAN AIR INSTANT ST FLOW
C 991QFF1, FAN AIR INSTANT ST FLOW
C
C DETERMINE IF AILS IN TRANSITION ZONE
IF(YCG.GT.BUND1)GO TO 13
IF(YCG.GT.BUND) GO TO 14
GO TO 16

13

14 IFLAG=-1

C IN TRANSITION ZONE

15

C IN GROUND EFFECT ZONE
Edward Arthur Kenney was born 18 July 1948 in London, Ontario, Canada to James A. and Evelyn V. Kenney. After graduating from G. A. Wheable Secondary School in 1967, he entered the Royal Military College (RMC) in Kingston, Ontario. He graduated from RMC in 1971 with the degree of Bachelor of Science in Electrical Engineering, and a commission in the Canadian Armed Forces. After graduating from the Primary Flying School and attending the Flying Training School at CFB Moose-jaw, Saskatchewan, he reclassified to be an Aerospace Engineer. Subsequent tours of duty included CFB Shearwater, Nova Scotia, as the Deputy Maintenance Records Officer and Heavy Maintenance Repair Officer; H.M.C.S. Preserver as the Fleet Air Maintenance Officer; and National Defence Headquarters, Ottawa, in the Directorate of Avionics and Armament Subsystems Engineering. Capt Kenney was assigned to the United States Air Force Institute of Technology in June 1976 in the Graduate Guidance and Control Program.

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The inherent instability in pitch and roll associated with an Air Cushion Landing System (ACLS) aircraft at low airspeeds was investigated, and a means to aid control in pitch and roll was developed. The control system required the use of vertical wing tip thrusters which provided thrust up or down depending on the control signal (similar to space vehicle thrusters). These thrusters could be activated alternately to control roll angle and roll rate with the use of a bang-bang optimal controller.
As well, the thrusters would be set forward of the aircraft center of gravity and could be activated in tandem to aid in pitch control.

The Jindivik Remotely Piloted Vehicle, an Australian target drone, was fitted with an ACLS and taxi tests showed the instability and need for a stabilization system. Subsequent use of Jindivik wind tunnel and taxi test data served as the basis for the development of the roll/pitch control system presented in this paper. Due to computational problems with the air cushion model of the computer program, the controller designs could not be completely verified; but expected trends in pitch, roll, and yaw control were shown.