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ON THE RELATIVE IMPORTANCE OF THE LOW SPEED CONTROL REQUIREMENT FOR V/STOL AIRCRAFT

Stephen Goldberger

ARO, Inc.

December 1966

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ARNOLD ENGINEERING DEVELOPMENT CENTER
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NOTICES

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FOREWORD

The work reported herein was done under provisions of the operating contract as independent research under Program Element 65402234.

The results of research presented were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC), Arnold Air Force Station, Tennessee, under Contract AF40(600)-1200. The research was conducted from December 1965 to September 1966 under ARO Project No. BB3602, and the manuscript was submitted for publication on September 21, 1966.

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This technical report has been reviewed and is approved.

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ABSTRACT

The closed loop dynamic response of a V/STOL airplane, pilot, and autostabilization system was studied with the purpose of demonstrating which airplane parameters are most important in determining the airplane's low speed flight characteristics. The influence of the stability augmentation system was found to be so great that the other parameters are small by comparison. The most important stability and control parameter in low speed, V/STOL aircraft flight, therefore, is control power.

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NOMENCLATURE

G	Gyro
g	Gravitational constant
I	Moment of inertia
K_p	Computer constant, pitch rate gain
K_R	Computer constant, roll rate gain
K_u	Computer constant, horizontal velocity, deg/ft
K_w	Computer constant, vertical velocity, deg/ft

K_y	Computer constant, yaw rate gain
k	Transfer function constant
L	Moment around x axis
M	Moment around y axis
M_u	$\frac{1}{m}\frac{\partial M}{\partial u}$
m	Mass
$\dot{m}$	Mass flow rate
N	Moment around z axis
P_o	Atmospheric pressure
p	Angular velocity around x axis
q	Angular velocity around y axis
R	Control system time constant, $\frac{\text{deg}}{\text{deg/sec}}$
R_{Ro}	Initial setting of R
r	Angular velocity around z axis
S	Laplace operator, 1/sec
s	Laplace transform variable (differential operator)
u	Velocity along x axis
v	Velocity along y axis
w	Velocity along z axis
X	Force along x axis
Y	Force along y axis
Z	Force along z axis
β_{1LT}	Aft left-wing louver angle
β_{2LT}	Forward left-wing louver angle
β_{1RT}	Aft right-wing louver set angle, deg, measured from z-axis direction, positive aft
β_{2RT}	Forward right-wing louver set angle, deg, measured from z-axis direction, positive aft
β_{SLT}	Left-wing louver stagger angle β_{2RT} - β_{1RT}
β_{SRT}	Right-wing louver stagger angle β_{2RT} - β_{1RT}

$\beta_{V_{LT}}$	Left-wing louver vector angle $\frac{\beta_{1_{LT}} + \beta_{2_{LT}}}{2}$
$\beta_{V_{RT}}$	Right-wing louver vector angle $\frac{\beta_{1_{RT}} + \beta_{2_{RT}}}{2}$
δ_n	Nose fan deflector angle
ζ	Damping ratio
θ	Angular deflection around y axis
τ	Time constant
ϕ	Angular deflection around x axis
ψ	Angular deflection around z axis

SUBSCRIPTS

SN	Servo network
SNF	Fan servo network

SECTION I
INTRODUCTION

During recent years, increasing interest has been shown in Vertical and Short Takeoff and Landing (V/STOL) aircraft. The continuing trend toward larger, faster airplanes requiring increasingly lengthy runways, together with the congestion presently associated with their use, has pointed out the usefulness of a craft possessing both the high cruise speed of the fixed wing airplane and the vertical takeoff ability of the helicopter.

Several of these V/STOL airplanes have been built and test flown. One of the primary concerns in their development was the availability of reliable aerodynamic data, particularly in transition and hover modes of flight. This motivated the design of various highly specialized test facilities: whirling crane rigs, tracks, and unique wind tunnels. Because of the broad spectrum of flight requiring development testing, some effort should be made to categorize the aerodynamic flight parameters in levels of their relative importance to V/STOL aircraft flight.

SECTION II
DISCUSSION

Comparison of various sources of V/STOL aerodynamic characteristics has shown a significant lack of correlation. The resulting uncertainty in the measured coefficients has stimulated the design of specialized V/STOL test facilities. In such facilities it may be necessary to compromise the capability of determining certain characteristics to a high degree of accuracy in order to obtain high accuracy for some other parameter. Some effort should be made to determine what parameters need to be tested. It would be pointless, and indeed wasteful, to include the capability of testing some characteristic whose variation causes little or no change in the airplane's overall performance.

Previous studies have shown that the dynamic characteristics of a V/STOL show varying sensitivity to different airplane parameters (Ref. 1). For example, the XC-142, X-22, X-22A, and X-19 were all found to be highly sensitive to a variation in the derivative M_u, or the rate of change of the pitching moment with respect to change in the forward velocity, around hover conditions. These studies considered only the open loop characteristics response of the airplane. The V/STOL aircraft have such unstable characteristics (which are detrimental to safe control in hover and low speed flight) that a high degree of stability augmentation is needed to make the airplane flyable. This results from the lack of stabilizing forces, combined with relatively low aerodynamic damping, permitting instability caused by coupling. These problems are most severe in hover flight.

The question was asked whether or not a stability augmentation system powerful enough to cope with this instability would also be powerful enough to accommodate considerable variation in the airplane parameters without changing the airplane's dynamic characteristics. The stability augmentation, together with the pilot's adaptability, was thought to be sufficient to completely overshadow the low speed characteristics of the airplane.

To test this, an analog simulation of a V/STOL airplane, complete with control system, stability augmentation system (SAS), and pilot influence, was accomplished. The airplane chosen was the Ryan XV-5A (Fig. 1). The near-hover case was examined. However, it is this case which is most critical with regard to aircraft dynamics. Increasing forward speed results in increased stability, i. e., greater damping and restoring forces.

The equations presented for simulation were the 6-degrees-of-freedom, rigid body equations of motion in the body-fixed coordinate system. The equations assume symmetry in the x-y and y-z planes and that the angular deflections are small. Analysis shows that the gyroscopic effect of rotating engine components is negligible (Ref. 2).

$$\dot{u} = rv - qw + g\theta + \Sigma \frac{X}{m}$$

$$\dot{v} = pw - ru + g\phi + \Sigma \frac{Y}{m}$$

$$\dot{w} = qu - pv + g + \Sigma \frac{Z}{m}$$

$$\dot{p} = \frac{I_{xz}}{I_{xx}} (\dot{r} + pq) - \left(\frac{I_{xx} - I_{zz}}{I_{xx}}\right) qr + \Sigma \frac{L}{I_{xx}}$$

$$\dot{q} = \frac{I_{xz}}{I_{yy}} (r^2 - p^2) - \left(\frac{I_{xx} - I_{zz}}{I_{yy}}\right) rp + \Sigma \frac{M}{I_{yy}}$$

$$\dot{r} = \frac{I_{xz}}{I_{zz}} (\dot{p} - qr) - \left(\frac{I_{yy} - I_{xx}}{I_{zz}}\right) pq + \Sigma \frac{N}{I_{zz}}$$

After examining the moments of inertia and estimated maximum angular rates of the airplane (Appendix I), the following terms were discarded:

the pq term in the $\dot{p}$ equation, $\frac{I_{xz}}{I_{yy}}$ term in the $\dot{q}$ equation, the $\frac{I_{xz}}{I_{zz}}$ term in the $\dot{r}$ equation, and pq term in the $\dot{r}$ equation. The summation terms represent forces and moments attributable to the engines: thrust and ram drag. A detailed explanation of these may be found in Appendix I.

A block diagram of the simulated system is shown in Fig. 2. The transfer function for each of the system components is shown in the figure. The second-order roots of the servodrive network and the rate gyro were found to be of such a high frequency that their effect was negligible. They were disregarded for simulation purposes.

The display consisted of a line shown on a cathode ray tube. Tilting of the line represented rolling of the aircraft; horizontal translation, yaw; and vertical translation, pitch. A conventional aircraft stick-and-rudder pedal control was built and provided pilot inputs into the computer. Because of display limitations and the requirement to prevent amplifier saturation, it was necessary to provide automatic control over vertical and forward velocity. This system was left fairly loose in order to retain the coupling between rotational and longitudinal velocities.

Gains on the SAS were adjusted to give the simulated airplane pleasant handling characteristics. No further attempt was made to optimize SAS gains. While the pilot controlled the system, an oscillograph recorded aircraft orientation, control surface deflection, and stick deflection. Typical time histories are shown in Fig. 3.

In the oscillographs the following parameters are shown. Tracks 1 and 2 show airplane orientation, pitch angle, and roll angle, respectively. Tracks 3 through 6 record several control surface deflections. Tracks 7 and 8 represent the pilot inputs of pitch-stick deflections and roll-stick deflections, respectively.

During successive "flights" the values of the derivatives were varied to the limits of computer capability. Incidentally, the resulting range of values was many times greater than the uncertainty in even a theoretical analysis of the airplane. The time histories shown in Fig. 3 follow a change in the derivative M_u, which was found to be the most sensitive derivative in previous studies (Ref. 3). A given set of maneuvers was carried out at each value. This included roll at constant pitch angle and pitch at zero roll angle.

Note the very high frequency associated with the roll control system response. It is difficult, if not impossible, for a pilot to control this mode unaided. A similar condition exists for the pitch mode although it is not quite so severe.

The pilot's input is characterized by square pulses, whereas the control system output is characterized by spikes. This results from the much shorter reaction time associated with the stability augmentation system. It should be recognized that the airplane's natural frequency and damping characteristics are functions of the various SAS gains, and the designer has the capability to vary them almost at will.

The performance of the airplane may be derived from the roll and pitch traces. For example, such things as the ability to hold a given roll angle while performing a pitching maneuver and the amount of pilot work in carrying out a maneuver are among the phenomena to be considered. This performance did not change significantly when the value of the derivative was changed. In fact, it is impossible to distinguish roll and pitch angle traces for different values of any derivative, as can be seen in Fig. 3. Further discussion of the time histories shown in Fig. 3 is found in Appendix II.

While the XV-5A was the only airplane studied, analysis of the equations of motion shows the program to be relatively insensitive to aircraft configuration. The moments of inertia are representative to all airplanes of size and weight similar to the XV-5A. The various aircraft derivatives are small with respect to SAS terms, and they were varied to such an extent that any VTOL in the small fighter-observation class was essentially represented.

SECTION III
CONCLUSION

The dynamics of a representative V/STOL aircraft were studied in the near-hover flight mode. It appears that near-hover flight is accompanied by instability to such an extent that an attempt at direct manual control is impractical. A stability augmentation system sufficiently powerful to make the aircraft flyable appears to be powerful enough to accommodate a rather large change in the derivatives without significantly altering handling qualities. In addition, the stabilizing characteristics of the human pilot are such that the actual performance of the system is invariant with respect to the stability derivatives.

The most important parameter to be tested, therefore, is control system power. It must be known whether the control system can in fact deliver its design performance. Low speed V/STOL testing should be concerned primarily with this and secondly with the various stability derivatives.

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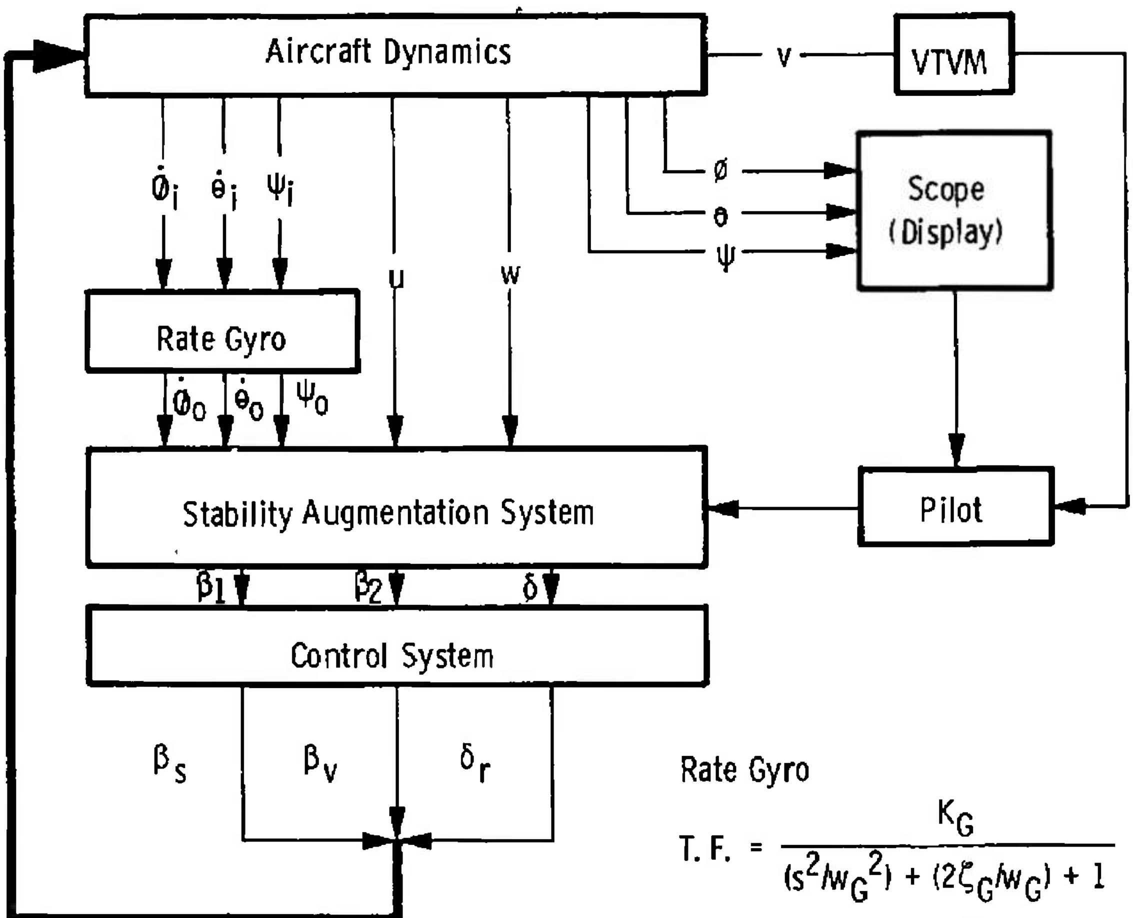
3. Kroeger, Richard A. and Rampy, J. M., Lt. "The Effect of Wind Tunnel Measurement Error on the Prediction of V/STOL Aircraft Flight Dynamics." Southeast Simulation Council, September 20, 1965.

Fig. 1 Ryan XV-5A



Rate Gyro

$$\text{T. F.} = \frac{K_G}{(s^2/w_G^2) + (2\zeta_G/w_G) + 1}$$

where K_G = 400 mv $\diagup$deg/sec

$\delta_G = 1/2$

$w_G = 160\ \text{sec}^{-1}$

a. Simulated Control System

Fig. 2 Simulated Control System and Nomenclature

$$\beta_{1_{LT}} = \left\{ -K_R\left(\frac{1 + R_{R0}\tau_R S}{1 + \tau_R S}\right)\left(\frac{1}{S} + R_{R1}\right)\phi_o - K_y\left(\frac{1}{S} + R_y\right)\dot{\psi}_o + K_w\left(\frac{1}{S} + R_w\right)w - K_u\left(\frac{1}{S} + R_u\right)u \right\}$$

$$\beta_{2_{LT}} = \left\{ K_R\left(\frac{1 + R_{R0}\tau_R S}{1 + \tau_R S}\right)\left(\frac{1}{S} + R_{R1}\right)\phi_o - K_y\left(\frac{1}{S} + R_y\right)\dot{\psi}_o - K_w\left(\frac{1}{S} + R_w\right)w - K_u\left(\frac{1}{S} + R_u\right)u \right\}$$

$$\beta_{1_{RT}} = \left\{ +K_R\left(\frac{1 + R_{R0}\tau_R S}{1 + \tau_R S}\right)\left(\frac{1}{S} + R_{R1}\right)\phi_o + K_y\left(\frac{1}{S} + R_y\right)\dot{\psi}_o + K_w\left(\frac{1}{S} + R_w\right)w - K_u\left(\frac{1}{S} + R_u\right)u \right\}$$

$$\beta_{2_{RT}} = \left\{ -K_R\left(\frac{1 + R_{R0}\tau_R S}{1 + \tau_R S}\right)\left(\frac{1}{S} + R_{R1}\right)\phi_o + K_y\left(\frac{1}{S} + R_y\right)\dot{\psi}_o - K_w\left(\frac{1}{S} + R_w\right)w - K_u\left(\frac{1}{S} + R_u\right)u \right\}$$

where

$K_R = 2.5 \quad R_{R0} = R_{R1} = 0.03$

$K_y = 12.4 \quad R_y = 0.19$

$K_u = 9.4 \quad R_u = 0.5$

$K_w = 10 \quad R_w = 0.2$

$$\delta = K_p\left(\frac{1 + R_{po}\tau_p S}{1 + \tau_p S}\right)\left(\frac{1}{S} + R_{p1}\right)\dot{\theta}_o$$

where

$K_p = 0.21$ deg/mv x 400 $R_{po} = R_{p1} = 0.05$

$$\beta_v = \left(\frac{\beta_2 + \beta_1}{2}\right)\left(\frac{1}{\tau_{SN}S + 1}\right)\left(\frac{1}{\frac{S^2}{\omega_{SN}^2} + \frac{2\zeta_{SN}S}{\omega_{SN}} + 1}\right)$$

$$\beta_s = \left(\beta_2 - \beta_1\right)\left(\frac{1}{\tau_{SN}S + 1}\right)\left(\frac{1}{\frac{S^2}{\omega_{SN}^2} + \frac{2\zeta_{SN}S}{\omega_{SN}} + 1}\right)$$

$$\delta_n = \delta\left(\frac{1}{\tau_{SNF}S + 1}\right)\left(\frac{1}{\frac{S^2}{\omega_{SNF}^2} + \frac{2\zeta_{SNF}S}{\omega_{SNF}} + 1}\right)$$

where

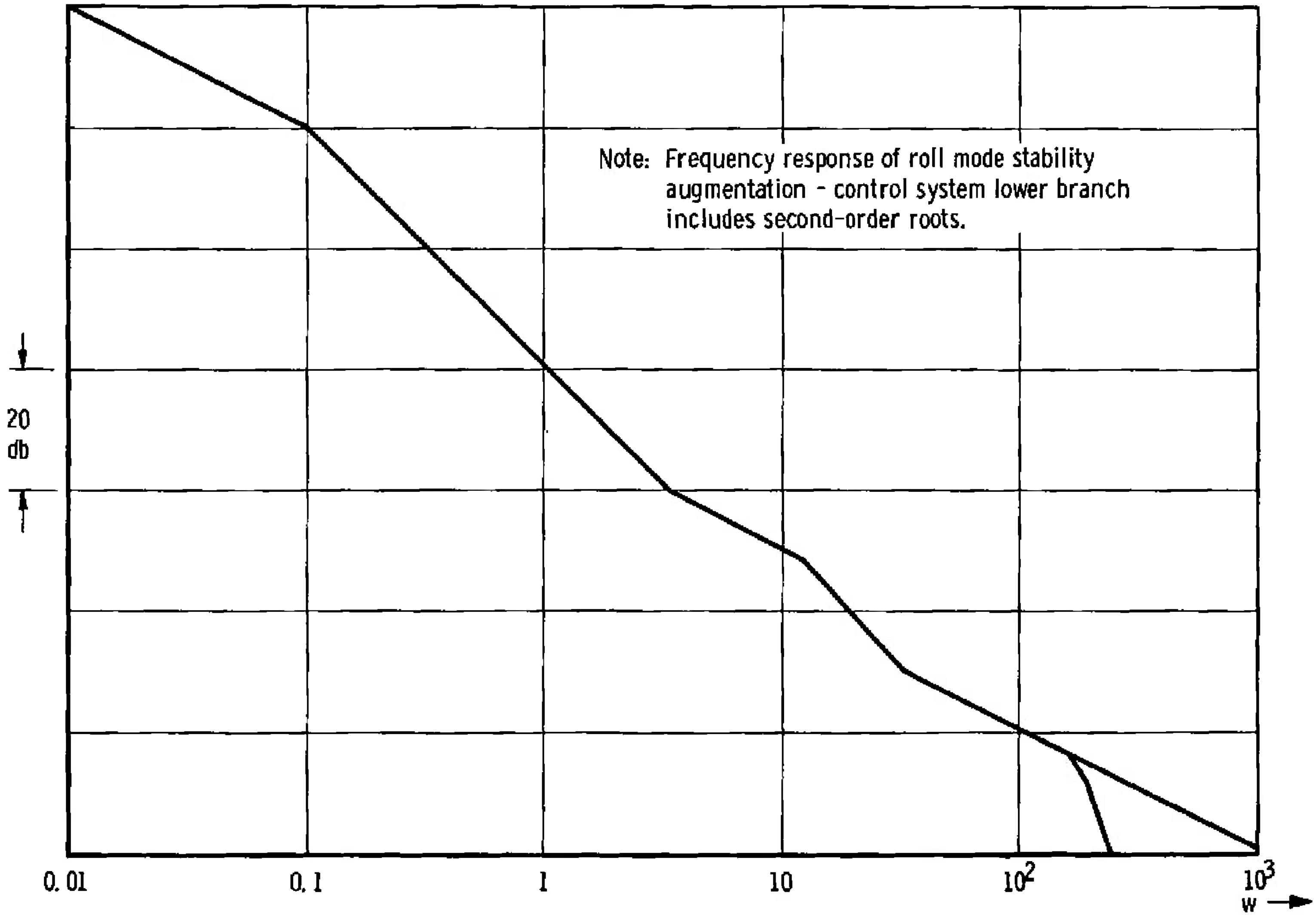
$\tau_{SN} = 0.08$ sec, $\tau_{SNF} = 0.045$ sec

$\omega_{SN} = 188\ \text{sec}^{-1} = \omega_{SNF}$

$\zeta_{SN} = 1/2 = \zeta_{SNF}$

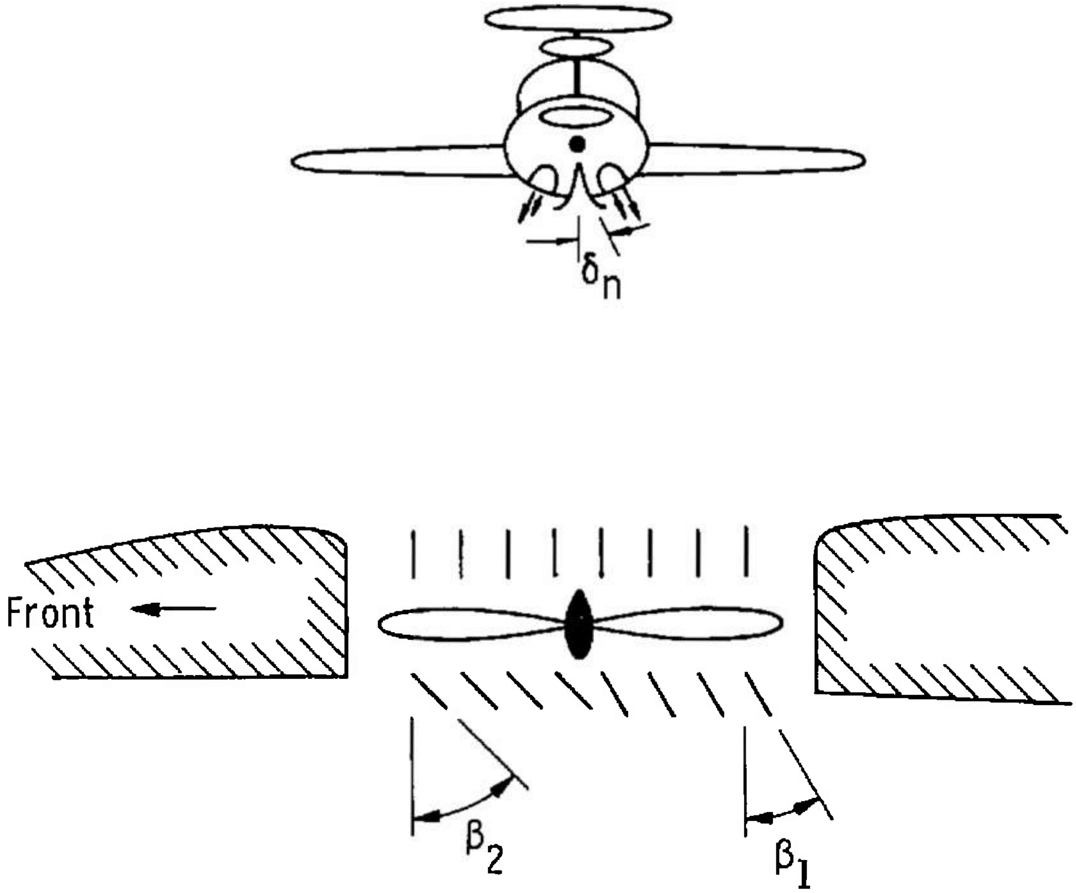
b. Simulated Control Equations

Fig. 2 Continued



c. Frequency Response of Roll Mode Stability Augmentation

Fig. 2 Continued

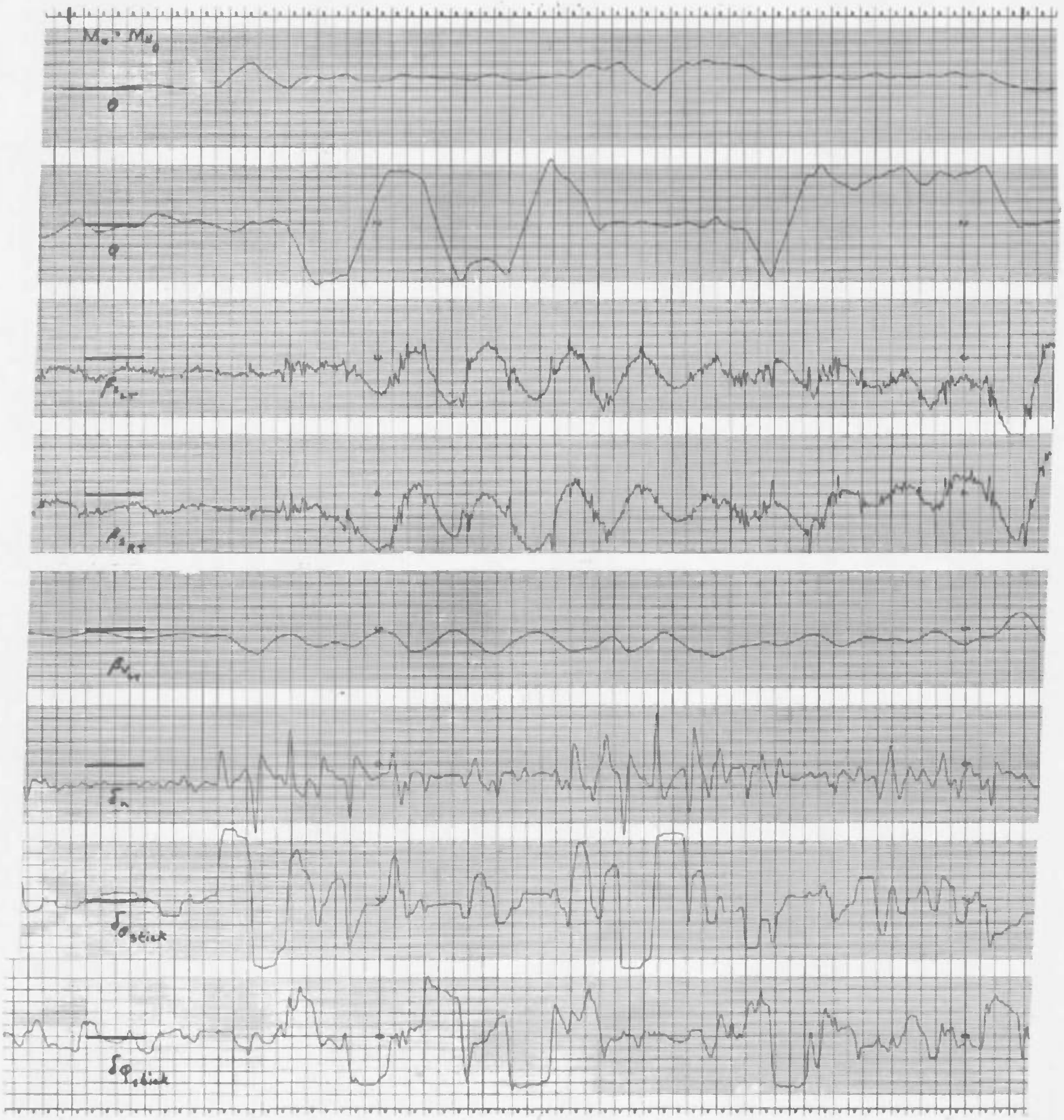


$$\beta_v = \frac{\beta_2 + \beta_1}{2}$$

$$\beta_s = \beta_2 - \beta_1$$

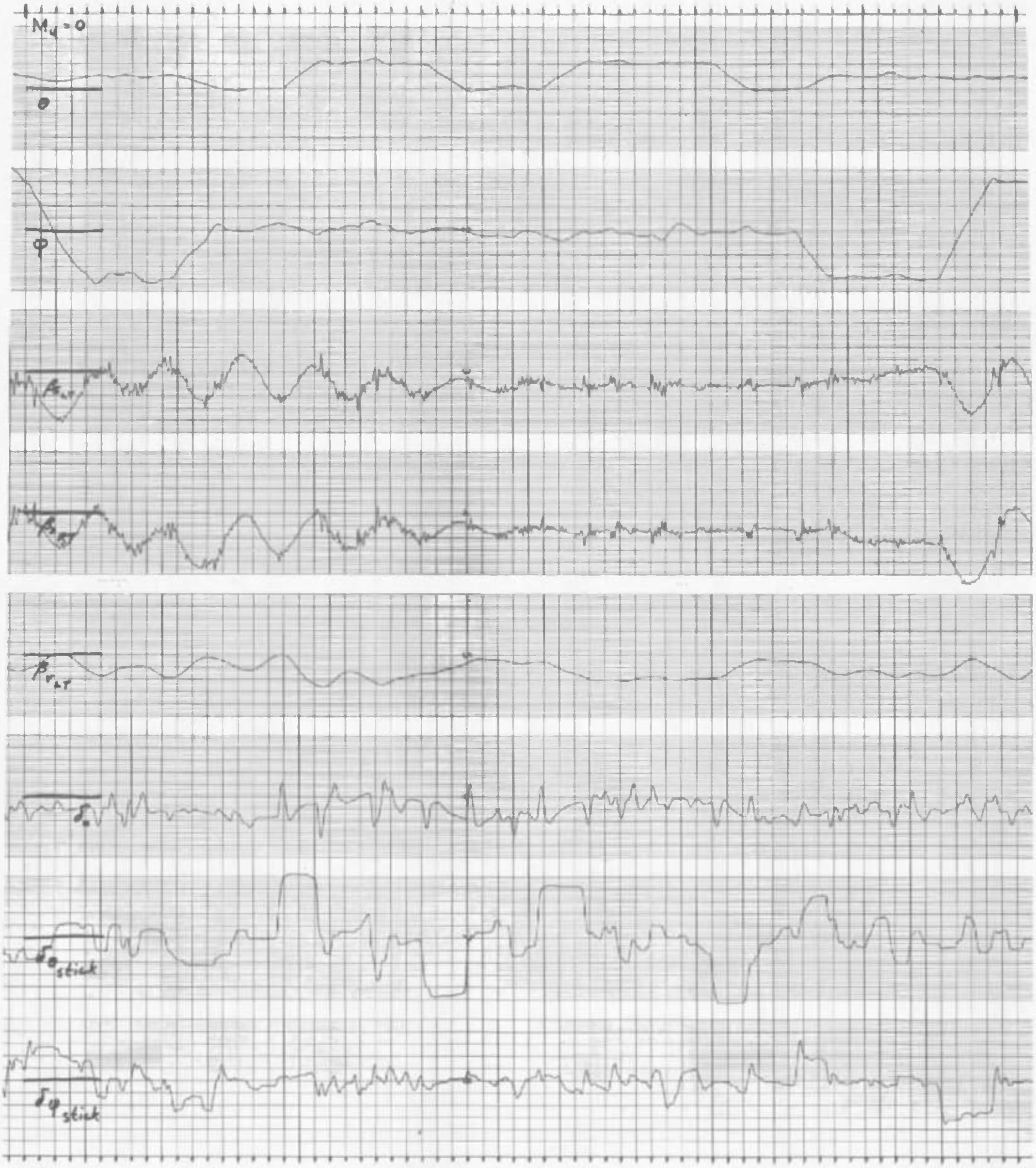
d. Control Nomenclature

Fig. 2 Concluded



a. $M_u = M_{u_0}$ (Basic Value of M_u)

Fig. 3 Typical Time Histories of Simulated Flights Showing Variation of M_u



b. $M_u = 0$

Fig. 3 Continued

c. $M_u = 2M_{u_o}$ (Twice Base Value)
Fig. 3 Continued

d. $M_u = 7.5\,M_{u_o}$ (Maximum Pot Setting)

Fig. 3 Concluded

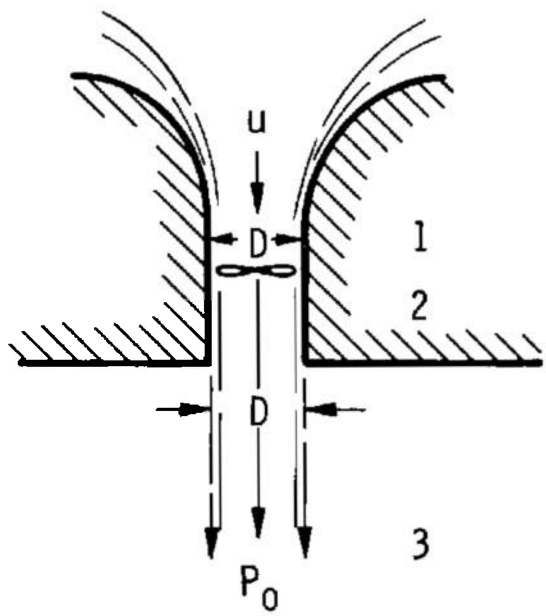


Fig. 4 Schematic Showing Aerodynamic Properties

APPENDIX I
AIRCRAFT EQUATIONS OF MOTION

The estimated maximum angular rates for the XV-5A (Ref. 1) are 0.4, 0.3, and 1 radians/sec in roll, pitch, and yaw, respectively. The estimated maximum acceleration is 1 radian/sec in each degree of freedom. The moments of inertia are: I_{xx} = 4200 slug ft^2, I_{yy} = 15,140 slug ft^2, I_{zz} = 17,420 slug ft^2, and I_{xz} = 920 slug ft^2. Substituting the above values into the equations of motion

$$\dot{p} = \frac{I_{xz}}{I_{xx}} (\dot{r} + pq) - \frac{I_{zz} - I_{yy}}{I_{xx}} qr + \frac{\Sigma L}{I_{xx}}$$

$$1 = \frac{920}{4200} (1 + 0.12) - \frac{2280}{4200} (0.3) + \frac{\Sigma L}{I_{xx}}$$

$$\dot{q} = \frac{I_{xz}}{I_{yy}} (r^2 - p^2) - \frac{I_{xx} - I_{zz}}{I_{yy}} (pr) + \frac{\Sigma M}{I_{yy}}$$

$$1 = \frac{920}{15,140} (1) - \frac{4200 - 17,420}{15,140} (0.4) + \frac{\Sigma M}{I_{yy}}$$

$$\dot{r} = \frac{I_{xz}}{I_{zz}} (\dot{p} - qr) - \frac{I_{yy} - I_{xx}}{I_{zz}} (pq) + \frac{\Sigma N}{I_{zz}}$$

$$1 = \frac{920}{17,420} (1 - 0.3) - \frac{10,060}{17,420} (0.12) + \frac{\Sigma N}{I_{zz}}$$

The following terms are at least an order of magnitude smaller than the left-hand side of the corresponding equation:

pq term in the $\dot{p}$ equation,

$\frac{I_{xz}}{I_{yy}}$ term in the $\dot{q}$ equation,

$\frac{I_{xz}}{I_{zz}}$ term in the $\dot{r}$ equation, and

pq term in the $\dot{r}$ equation.

These terms were thus not included in the analog simulation.

Finally, the engine contribution to the vehicle dynamics may be broken into two parts: thrust and momentum drag. The magnitudes of these elements may be calculated as follows.

THRUST

With reference to Fig. 4, the pressure relationship between "0" and "1" is

$$p_\infty = P_1 + \frac{1}{2}\rho u^2$$

and between "2" and "3",

$$p_2 = P_\infty$$

With these, the lift fan thrust may be calculated by

$$\text{Thrust} = T = (p_2 - p_1)\frac{\pi D^2}{4} = \frac{1}{2}\rho u^2 \frac{\pi D^2}{4}$$

MOMENTUM DRAG

The momentum drag may be given by

$$\text{Drag} = D = \frac{d}{dt}(mv)$$

$$D = \lim_{\Delta t \to 0} \frac{mv|_{t+\Delta t} - mv|_t}{\Delta t}$$

Next, the following are assumed to apply.

1. The velocity of airplane is a constant = w,
2. The propulsive jet is uniform and has a constant velocity, u, and
3. The mass of the airplane is a constant, M.

If constant values of jet area, A, and air density, ρ, are assumed, the drag is then given by

$$D = \lim_{\Delta t \to 0} \frac{Mw + w\int_{t=-\infty}^{t+\Delta t} \rho u A dt - \left[M_w + w \int_{t=-\infty}^{t} \rho u A dt\right]}{\Delta t}$$

$$D = \lim_{\Delta t \to 0} \rho w u A \int_{t}^{t+\Delta t} \frac{dt}{\Delta t}$$

$D = w\dot{m}$, where $\dot{m}$ is the mass flow rate of the jet.

The momentum drag force is parallel to the direction of w, regardless of the relation between w and u.

Thus the resulting dynamics equations which included the effects of thrust vectoring and momentum drag are

$$\Sigma X = \frac{\partial x}{\partial \beta_S}\left(\frac{\beta_{S_{LT}} + \beta_{S_{RT}}}{2}\right) + \frac{\partial x}{\partial \beta_V}\left(\frac{\beta_{V_{LT}} + \beta_{V_{RT}}}{2}\right) - \dot{m}u$$

$$\Sigma Y = -(\dot{m}v - \dot{m}_{NF}\,\bar{x}_{NF}\,r)$$

$$\Sigma Z = Z_0 + \frac{\partial z}{\partial \beta_S}\left(\frac{\beta_{S_{LT}} + \beta_{S_{RT}}}{2}\right) + \frac{\partial z}{\partial \beta_V{}^2}\left(\frac{\beta_{V_{LT}}{}^2 + \beta_{V_{RT}}{}^2}{2}\right)$$

$$+ \frac{\partial z}{\partial \delta_n}\delta_n - \dot{m}w - \dot{m}_{NF}\,\bar{x}_{NF}\,q$$

$$\Sigma L = \left\{\frac{1}{2}\left[\frac{\partial z}{\partial \beta_S}\left(\beta_{S_{LT}} - \beta_{S_{RT}}\right) + \frac{\partial z}{\partial \beta_V{}^2}\left(\beta^2_{V_{LT}} - \beta^2_{V_{RT}}\right)\right]\right.$$

$$\left. - \dot{m}_{MF}\,\bar{y}_{MF}\,p\right\}\bar{y}_{MF} + \frac{\partial L}{\partial v}v$$

$$\Sigma M = \left\{\frac{1}{2}\left[\frac{\partial z}{\partial \beta_S}\left(\beta_{S_{LT}} + \beta_{S_{RT}}\right) + \frac{\partial z}{\partial \beta_V{}^2}\left(\beta_{V_{LT}}^2 + \beta_{V_{RT}}^2\right)\right] + \dot{m}_{MF}\,(\bar{x}_{MF}\,q - w)\right\}\bar{x}_{MF}$$

$$- \left\{\frac{\partial z}{\partial \delta_n}\delta_n + \dot{m}_{NF}\,(\bar{x}_{NF}\,q - w)\right\}\bar{x}_{NF} + \frac{\partial M}{\partial u}u$$

$$\Sigma N = \left\{\frac{1}{2}\left[\frac{\partial x}{\partial \beta_S}\left(\beta_{S_{LT}} - \beta_{S_{RT}}\right) - \frac{\partial x}{\partial \beta_V}\left(\beta_{V_{LT}} - \beta_{V_{RT}}\right)\right] - \dot{m}_{MF}\,\bar{y}_{MF}\,r\right\}\bar{y}_{MF}$$

$$- \dot{m}_{NF}\,(\bar{x}_{NF}\,r + v)\,\bar{x}_{NF}$$

These are the complete dynamics equations to be presented for simulation.

APPENDIX II
DISCUSSION OF THE AIRCRAFT CONTROL SYSTEM

The purpose of this simulation was to investigate the importance of the control and stability augmentation system with respect to aircraft dynamics. It was not intended as a rigorous duplication of the General Electric-Ryan simulation effort nor was it a test bed for extensive flight studies. However, much was learned concerning the pilot-SAS-airframe interaction by observing the time histories.

Before continuing, the reader should thoroughly familiarize himself with the simulated control system schematic shown in Fig. 2 and the performance plots shown in Fig. 3. Track 1 of the performance plots is the pitch angle (θ), and Track 2 is the roll angle (ϕ) of the simulated airplane. Tracks 3 through 6 are control surface deflections; Tracks 3 and 4, the stagger angle on the left ($\beta_{S_{LT}}$) and right wing-fan louvers ($\beta_{S_{RT}}$), respectively (Fig. 2); Track 5, the vector angle on the left wing-fan louvers ($\beta_{V_{LT}}$); and Track 6, the nose deflector door angle (δ_n). The vector angle of the right wing-fan louvers is not shown because of lack of data recording capacity. Tracks 7 and 8 show pilot inputs; Track 7 is the pitch-stick deflection (δ_θ), and Track 8 is the roll-stick deflection (δ_ϕ).

The chart speed was such that each vertical division represents 1 sec of time. Every time history has a mark in the upper left-hand corner on top of one vertical division. This is defined to be time zero.

The stagger angle controls the normal (z-component) force exerted by its respective wing, thus providing roll and altitude control. When both stagger angles change in the same sense, the control system is correcting the vertical velocity or position of the airplane. When the stagger angles change in the opposite sense, a rolling moment is generated. For example, in Fig. 3a, at time 1 sec, Tracks 3 and 4 show an equal but opposite spike. This indicates a rolling moment applied to the airplane. The large amplitude low frequency oscillations starting approximately at time 20 sec are in phase; they represent variations in the airplane's gross lift. A trace of vertical position would show a corresponding oscillation in altitude.

The vector angle performs in a similar manner, except that the force is in the longitudinal direction. It is not clear from the single track of data presented whether the vector angle variations are controlling the horizontal (x) translation.

The deflector-door angle determines the normal force from the nose fan. This is used primarily for pitch control. The geometry is such that a downward pitching moment is available.

Comparison of pilot stick movement, airplane orientation, and the control surface deflection illustrates the pilot-airframe-stability augmentation system interaction. The most striking feature is the frequency of the control surface movement, especially in roll control. The pilot would be unable to accurately produce control stick deflections of this frequency.

The nature of the V/STOL is such that the control surface motion required to perform a maneuver is very unconventional. For example, consider the portion of Fig. 3c which is emphasized. The pilot wishes to develop a constant pitch rate over a short period of time. To do this, he makes a stick deflection and holds it constant for that length of time. The control surface deflection which causes this maneuver is a spike, followed by a ramp, followed by an inverse spike. The complexity of such a signal further illustrates the importance of the stability augmentation system, not only to make the airplane flyable, but to greatly simplify the task of stabilizing the airplane, thus leaving the pilot free to make higher order decisions.

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Available in DDC.	Arnold Engineering Development Center, Air Force Systems Command, Arnold Air Force Station, Tennessee

13. ABSTRACT

The closed loop dynamic response of a V/STOL airplane, pilot, and autostabilization system was studied with the purpose of demonstrating which airplane parameters are most important in determining the airplane's low speed flight characteristics. The influence of the stability augmentation system was found to be so great that the other parameters are small by comparison. The most important stability and control parameter in low speed, V/STOL aircraft flight, therefore, is control power.

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14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
1 V/STOL ~~aircraft~~ airplanes -- Controls low speed control closed loop response flight characteristics stability augmentation system 2. V/STOL airplanes -- Stability 1—	2—					

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