Chapter 3

SURVEY OF WIND TUNNEL TECHNIQUES FOR DETERMINATION OF TRANSLATIONAL ACCELERATION DERIVATIVES

3.1 BACKGROUND

As already discussed, an angular forced-oscillation wind tunnel test yields a composite derivative which is the sum of two effects; angular rate and translational acceleration (such as $C_{m\dot{\alpha}} + C_{m\ddot{\alpha}}$). In the past, the acceleration derivatives have been neglected in performing dynamic stability calculations because:

- i) at the low angles of attack and symmetric flow conditions associated with flight of conventional straight-wing-tailed aircraft, estimates indicated that the acceleration effect was negligible; all of the effect was attributed to the angular rate, and
- ii) little information was available to permit reasonably accurate estimates of the magnitude of the $\dot{\alpha}$ and $\ddot{\alpha}$ effects at high angles of attack. As a result, the entire composite derivative was used to represent the purely rotary derivative in the equations of motion.

As the NACA began to investigate the characteristics of the dynamic derivatives of swept and delta wings in the 1950's, it was discovered that large, nonlinear values of the lateral acceleration-in-sideslip derivatives were obtained at high angles of attack (Refs. 4 to 9). This is clearly illustrated in Figure 3.1 (Ref. 7), where the variation of damping-in-yaw with angle of attack are shown for a 45° swept-wing-fuselage-tail combination at very low speeds. The discrepancy between the purely rotary derivative C_{n_r} and angular forced-oscillation derivative $C_{n_r} - C_{n_{\dot{\alpha}}}$ is attributable to a large increase in the value of $C_{n_{\ddot{\alpha}}}$. As the angle of attack increases, various time-lag effects usually associated with flow separation and asymmetric vortex sheddings cause the "unsteady" (acceleration) effects to become more prominent.

There is a renewed interest in acceleration derivatives due to their predicted significance at high angles of attack, and there are several reasons why it is of vital importance to know their value:

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**A Wind Tunnel Apparatus for Determination
of
Translational Acceleration Stability Derivatives**

by

Janet C. Kuzina
BSc. (Honours Physics)

A thesis submitted to the Faculty of Graduate
Studies and Research in partial fulfillment
of the requirements for the Degree of

Master of Engineering

Department of
Mechanical and Aeronautical Engineering
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ABSTRACT

A description of a recently commissioned wind tunnel translational oscillation apparatus is presented. The apparatus was designed at the National Aeronautical Establishment with the particular goal of obtaining acceleration stability derivatives on aircraft configurations at high angles of attack due to translational oscillation along both the z and y body axes. The development of the apparatus and the techniques for data analysis in dynamic stability experiments are discussed. Results of laboratory static and dynamic calibrations and preliminary results of wind tunnel tests (Mach 0.25) of a simplified fighter aircraft configuration are presented. A complete set of force and moment acceleration derivatives was obtained due to translational oscillation along the z body axis at angles of attack to 42.5° and zero sideslip. The results show that acceleration effects are quite substantial at high angles of attack as indicated by the magnitudes of some of the dynamic derivatives. All derivatives displayed significant nonlinear trends.

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List of Symbols

a_1	— In-phase component of measured force or moment (N or Nm)
a_2	— Quadrature component of measured force or moment (N or Nm)
b	— Lateral reference length, wingspan (m)
$\bar{c}$	— Longitudinal reference length, wing mean aerodynamic chord (m)
C_{i_p}	— Dynamic derivatives with respect to roll rate parameter $\frac{\partial C_i}{\partial \left(\frac{p b}{2V_\infty} \right)}, i = l, m, n$
C_{i_q}	— Dynamic derivatives with respect to pitch rate parameter $\frac{\partial C_i}{\partial \left(\frac{q \bar{c}}{2V_\infty} \right)}, i = l, m, n$
C_{i_r}	— Dynamic derivatives with respect to yaw rate parameter $\frac{\partial C_i}{\partial \left(\frac{r b}{2V_\infty} \right)}, i = l, m, n$
C_{i_α}	— Static derivatives with respect to angle of attack $\frac{\partial C_i}{\partial \alpha}, i = l, m, n, z$
$C_{i_{\dot{\alpha}}}$	— Dynamic derivatives with respect to α -rate parameter $\frac{\partial C_i}{\partial \left(\frac{\dot{\alpha} \bar{c}}{2V_\infty} \right)}, i = l, m, n, z$
C_{i_β}	— Static derivatives with respect to sideslip angle $\frac{\partial C_i}{\partial \beta}, i = l, m, n, y$
$C_{i_{\dot{\beta}}}$	— Dynamic derivatives with respect to β -rate parameter $\frac{\partial C_i}{\partial \left(\frac{\dot{\beta} b}{2V_\infty} \right)}, i = l, m, n, y$
$C_{i_{\dot{z}}}$	— Derivatives with respect to $\dot{z}$ $\frac{\partial C_i}{\partial \left(\frac{\dot{z}}{V_\infty} \right)}, i = l, m, n, z$
$C_{i_{\ddot{z}}}$	— Derivatives with respect to $\ddot{z}$ $\frac{\partial C_i}{\partial \left(\frac{\ddot{z} \bar{c}}{2V_\infty} \right)}, i = l, m, n, z$
C_l	— Rolling moment coefficient, $L/(q_\infty S b)$
C_m	— Pitching moment coefficient, $M/(q_\infty S \bar{c})$
C_n	— Yawing moment coefficient, $N/(q_\infty S b)$

C_Y	— Sideforce coefficient, $Y/(q_\infty S)$
C_Z	— Normal force coefficient, $Z/(q_\infty S)$
d_1	— Sting diameter at aft end of cantilevers with shims in position (mm)
d_2	— Cantilever beam length (mm)
d_p	— Total distance travelled by laser light beam from model to sensor (mm)
d_s	— Shim thickness (mm)
F	— Force (N)
f	— Frequency of oscillation (Hz)
I	— Moment or product of inertia (kgm^2)
k	— Mechanical stiffness (N/m)
l	— Reference length (m)
L	— Rolling moment (Nm)
M	— Pitching moment (Nm)
M_∞	— Freestream Mach number
m	— Effective model and balance mass (kg)
N	— Yawing moment (Nm)
p, q, r	— Body axes angular velocities (rad/s)
q_∞	— Freestream dynamic pressure (N/m^2)
Re	— Reynolds number per meter (m^{-1})
S	— Reference area, wing plan area (m^2)
sn	— Sensitivity, Voltage/load ($V/(N \text{ or } Nm)$)
T_∞	— Freestream static temperature (K)
t	— Time (s)
t^*	— Time scale, l/u_∞ (s)
V	— Strain-gauge voltage (Volts)
V_∞	— Freestream velocity (m/s)
u, v, w	— Body axes linear velocities (m/s)
x, y, z	— Body-fixed axes system
x', y', z'	— Aerodynamic axes system
X, Y, Z	— Wind axes system
X_T, Y_T, Z_T	— Tunnel-fixed axes system
y_s	— Vertical movement of light spot on position sensing photodetector (mm)
Y	— Sideforce (N)

Z	— Normal force (N)
$\bar{Z}$	— Drive force (N)
α	— Angle of attack (deg or rad)
β	— Angle of sideslip (deg or rad)
γ	— Mechanical damping (Ns/m)
Δ	— Amplitude or increment
ΔG	— Aerodynamic vector (Nm)
$\Delta \zeta$	— Induced cantilever deflection under oscillatory conditions (deg)
$\Delta \zeta_d$	— Maximum cantilever deflection under oscillatory conditions (deg)
$\Delta \zeta_s$	— Cantilever deflection due to static loads (deg)
ζ_0	— Cantilever convergence angle at equilibrium position (deg)
η	— Phase angle of secondary motions with respect to primary motion (deg)
η_s	— Phase angle of sting pitching oscillation with respect to primary motion (deg)
θ_b	— Artificially induced model/balance pitch angle (deg)
θ_d	— Sting pitching oscillation (deg)
θ_s	— Static sting deflection (deg)
λ	— Phase angle of drive force with respect to primary motion (deg)
ξ	— Damping factor
ρ_∞	— Freestream density (kg/m^3)
σ	— Aerodynamic angle of attack (deg or rad)
ϕ'	— Aerodynamic angle of bank (deg or rad)
ψ, θ, ϕ	— Euler angles; yaw, pitch and roll (deg or rad)
ω	— Circular frequency of oscillation, $\omega = 2\pi f$ (rad/s)
$\bar{\omega}$	— Reduced frequency, $\bar{\omega}b/(2V_\infty)$ or $\bar{\omega}c/(2V_\infty)$, where appropriate
ω_n	— Natural frequency (rad/s)

Superscripts

$(\dot{}), (\ddot{})$	— First and second derivatives with respect to time
$(\cdot)$	— Nondimensional time parameter
$(\prime)$	— Aerodynamic axes system
$(*)$	— Stability derivative multiplication by $\cos \alpha_0$

Subscripts

A	— Aft yaw/pitch flexure
F	— Forward yaw/pitch flexure
x, y, z	— x, y, z body axis or direction
L, M, N	— Pitch, roll, yaw degrees of freedom
$\circ$	— Equilibrium (reference) condition
T	— Tare conditions
W	— Wind-on conditions
θ, ϕ, ψ	— Oscillation in pitch, roll or yaw

Abbreviations

AC	— Oscillatory component
A/D	— Analog-to-digital
AEDC	— Arnold Engineering Development Center
CG	— Center of gravity
DC	— Non-oscillatory component
DOF	— Degree of freedom
DFVLR	— Deutsche Forschungs- und Versuchsanstalt für Luft- und Raumfahrt
FFA	— Flygtekniska Försöksanstalten
FFT	— Fast Fourier Transform
LRC	— Langley Research Center
MAC	— Mean aerodynamic chord
MF	— Magnification factor
NACA	— National Advisory Committee for Aeronautics
NAE	— National Aeronautical Establishment
NASA	— National Aeronautics and Space Administration
RC	— Reference center
SDM	— Standard Dynamics Model
STA	— Station
UAL	— Unsteady Aerodynamics Laboratory
VPI & SU	— Virginia Polytechnic Institute and State University

Chapter 1

INTRODUCTION

Present day high performance military aircraft require the ability to maneuver effectively at high angle of attack flight conditions. They also fly at non-zero angles of sideslip and perform translational (rather than only rotary) maneuvers. These flight conditions are characterized by asymmetric, nonlinear and unsteady aerodynamic flow phenomena which affect the stability characteristics of the aircraft and introduce new problems to the prediction of dynamic stability parameters. Such parameters, or stability derivatives (which are strong functions of angle of attack) are necessary for the design of the control system and for evaluation of aircraft handling qualities at high angles of attack. Thus, in addition to quasi-steady effects, such as represented by derivatives of aerodynamic reactions due to angular velocity, the design of new aircraft will also require technology that addresses purely unsteady aerodynamic effects at high angles of attack, such as those represented by derivatives due to the time rates of change of angles of attack and sideslip (acceleration derivatives).

It is imperative that more information be gathered to improve the understanding of flow phenomena in the high angle of attack flight regime and to provide a complete data base for use in mathematical models to predict the dynamics of aircraft. In recognition of this need, several wind tunnel institutions around the world are currently engaged in research programs designed to furnish information regarding the aerodynamics and dynamic stability characteristics of modern fighter configurations and missiles at high angles of attack.

The dynamic stability study documented in this thesis was conducted at the National Aeronautical Establishment (NAE) as part of a collaborative research program with the National Aeronautics and Space Administration (NASA) Ames Research Center. One of the primary purposes of the program is to obtain a complete set of dynamic stability derivatives which includes those due to pitching, yawing, rolling and translational oscillation, for a general fighter aircraft configuration at high angles of attack. This thesis deals with that part of the program concerning measurement of stability derivatives due to translational oscillation.

To meet program objectives, a unique, sting-mounted forced-oscillation wind tunnel

apparatus was designed at the NAE, capable of translational oscillation along both z and y body axes. Angular oscillation apparatuses (pitch/yaw, roll) yield stability derivatives which are the sum of two effects; angular rate and acceleration. The new translational oscillation apparatus yields acceleration derivatives only, permitting a direct determination of a complete set of derivatives due to acceleration along either z or y body axes for a full aircraft configuration at high angles of attack, as yet unprovided by the wind tunnel testing community.

The work presented in this thesis focuses on the development and performance evaluation of the new apparatus. It was the author's responsibility to:

- (i) become familiar with the basis of design and principle of operation of the new apparatus and participate in its development to the final design stage;
- (ii) evaluate its static and dynamic response characteristics by performing a laboratory static and dynamic calibration;
- (iii) derive mathematical formulations for computing derivatives due to translational oscillation from experimentally obtained quantities and implement these into the data acquisition software and procedures utilized at the NAE;
- (iv) generate any additional software required for data reduction and processing of data from dynamic stability experiments with the new apparatus, and
- (v) conduct a wind tunnel verification test to verify the operating capability of the apparatus by obtaining a preliminary set of derivatives due to translational oscillation for a simplified fighter aircraft configuration.

Details of the abovementioned investigations are found in this thesis. Initially, a brief look at fundamental concepts concerning stability derivatives and a critical review of past techniques used in determining acceleration derivatives are presented.

Chapter 2 FUNDAMENTALS

2.1 EQUATIONS OF MOTION

In order to place the role of dynamic stability testing in perspective and demonstrate the basis for the experimental techniques used, it is useful to examine the equations of motion of an aircraft. Figure 2.1 shows a body-fixed system of axes that are rigidly attached to the aircraft and thus move with it. The equations of motion for the linear and angular degrees of freedom are simply expressions of Newton's laws for translation along and rotation around each axis, respectively. It is assumed that:

- i) the aircraft is a single rigid body with zz a plane of symmetry, and
- ii) the motion of the aircraft consists of small disturbances from a reference condition of steady, symmetric flight.

The equations of motion for an aircraft with controls locked, referred to the body-fixed axes are then as follows:

$$\sum F_x = m\dot{u} \quad (2.1a)$$

$$\sum F_y = m(\dot{v} + u_0 r) \quad (2.1b)$$

$$\sum F_z = m(\dot{w} - u_0 q) \quad (2.1c)$$

$$\sum L = \dot{p}I_x - \dot{r}I_{xz} \quad (2.2a)$$

$$\sum M = \dot{q}I_y \quad (2.2b)$$

$$\sum N = \dot{r}I_z - \dot{p}I_{xz} \quad (2.2c)$$

(A complete derivation can be found in any text on aircraft dynamics. See References 1, 2 for example). The summations F_x, F_y, F_z and L, M, N are the changes in the externally applied forces and moments (about the CG) on the aircraft resulting from a small disturbance from the reference flight condition in each degree of freedom. For an aircraft model oscillating in a wind tunnel, each of these external forces and moments consist of:

- i) a change in aerodynamic reaction resulting from oscillation;
- ii) a change in mechanical reaction caused by oscillation, and

$$\alpha = \tan^{-1} \frac{w}{u_0 + u} \approx \frac{w}{u_0}$$

$$\beta = \sin^{-1} \frac{v}{V_\infty} \approx \frac{v}{V_\infty}$$

$$u_0 \approx V_\infty$$

to the first order

Fig. 2.1 Aircraft body-fixed system of axes

iii) a forcing force or moment.

In the remainder of this chapter, a discussion of only the moment relations are considered for brevity, although the treatment applies to the force relations as well.

2.2 AERODYNAMIC MOMENTS

The method for specifying the aerodynamic moment relations is that developed by Bryan (see Ref. 1). It is based on the hypothesis that the aerodynamic reactions of equations (2.2) can be expanded with respect to the instantaneous values of the disturbance velocities u, v, w, p, q, r and their time derivatives. Expansions for the moments are obtained in the form of a Taylor series in these variables which is linearized by neglecting the higher order terms. It is convenient to assume, for the present, that we have a symmetric aircraft in a condition of symmetric flight. This will reduce the number of variables needed to describe the aerodynamic moment system. It should be emphasized that this assumption is not necessary for aircraft dynamics calculations and that any of the terms omitted can be restored into the equations when necessary. With this assumption, however, the roll, pitch and yaw aerodynamic expansions are:

$$L_{aero} = L_v v + L_{\dot{v}} \dot{v} + L_p p + L_r r \quad (2.3a)$$

$$M_{aero} = M_u u + M_w w + M_{\dot{w}} \dot{w} + M_q q \quad (2.3b)$$

$$N_{aero} = N_v v + N_{\dot{v}} \dot{v} + N_p p + N_r r \quad (2.3c)$$

The symbol L_v stands for $(\frac{\partial L}{\partial v})_0$, M_q for $(\frac{\partial M}{\partial q})_0$, and so on, such quantities being called the *stability derivatives* of the aircraft. The subscript zero indicates that the derivative is to be evaluated at the reference flight condition.

2.3 STABILITY DERIVATIVE COEFFICIENTS

In practice, stability derivatives are almost invariably put into nondimensional form, and the equations of motion are treated nondimensionally. The NACA system (Table 2.1) will be used here with time constant defined as $t^* = l/u_0$ and characteristic length $l = \frac{1}{2} \bar{c}$ in the longitudinal equations and $l = \frac{1}{2} b$ in the lateral equations. All coefficients are obtained assuming a symmetric, steady state reference condition, $w_0 = v_0 = p_0 = q_0 = r_0 = 0$, typical for a model oscillating about a reference angle of attack α_0 .

TABLE 2.1
THE NONDIMENSIONAL SYSTEM (NACA)

Dimensional Quantity	Divisor	Nondimensional Quantity
$L \ M \ N$	$\rho_{\infty} V_{\infty}^2 S l$	$C_l \ C_m \ C_n$
$u \ v \ w$	u_0	$\hat{u} \ \beta \ \alpha$
$p \ q \ r$	$1/t^*$	$\hat{p} \ \hat{q} \ \hat{r}$
$\dot{\beta} \ \dot{\alpha}$	$1/t^*$	$D\beta \ D\alpha$
t	$t^* = l/u_0$	$\hat{t}$

With the assistance of Figure 2.1, the application of the definitions of Table 2.1 to equations (2.3) leads to the desired nondimensional equations:

$$L_{aero} = q_{\infty} S b \left(C_{l_{\beta}} \beta + C_{l_{\dot{\beta}}} \frac{\dot{\beta} b}{2V_{\infty}} + C_{l_p} \frac{pb}{2V_{\infty}} + C_{l_r} \frac{rb}{2V_{\infty}} \right) \quad (2.4a)$$

$$M_{aero} = q_{\infty} S \bar{c} \left(C_{m_{\alpha}} \alpha + C_{m_{\dot{\alpha}}} \frac{\dot{\alpha} \bar{c}}{2V_{\infty}} + C_{m_q} \frac{q \bar{c}^2}{2V_{\infty}} \right) \quad (2.4b)$$

$$N_{aero} = q_{\infty} S b \left(C_{n_{\beta}} \beta + C_{n_{\dot{\beta}}} \frac{\dot{\beta} b}{2V_{\infty}} + C_{n_p} \frac{pb}{2V_{\infty}} + C_{n_r} \frac{rb}{2V_{\infty}} \right) \quad (2.4c)$$

2.4 DYNAMIC DERIVATIVES

Equations (2.4) contain *static* derivatives (the variation of a given moment with displacement coordinate α or β) and *dynamic* derivatives. The various dynamic derivatives are categorized in Table 2.2. Those due to pure rotation (p, q, r) include *damping* derivatives ($C_{m_q}, C_{n_r}, C_{l_p}$) and *cross* derivatives (C_{n_p}, C_{l_r}). In addition, *cross-coupling* derivatives ($C_{n_q}, C_{l_q}, C_{m_r}$, and C_{m_p}) may be included in the equations of motion (2.4) when the aircraft is expected to have aerodynamic cross-coupling between the longitudinal and lateral degrees of freedom. These terms would be of significant magnitude where asymmetric flow conditions exist, for instance, for an aircraft that is flying at high angles of attack at zero sideslip, and also at nonzero angles of sideslip. In this case it is necessary to consider the equations of motion of an aircraft together and not in two separate longitudinal and lateral groups.

TABLE 2.2

DYNAMIC MOMENT DERIVATIVES

Pure Rotation	Translational Acceleration	Fixed Axis Oscillation
		<u>Damping Derivatives</u>
C_{l_p} C_{m_p} C_{n_p} C_{l_q} C_{m_q} C_{n_q} C_{l_r} C_{m_r} C_{n_r}	$C_{l_{\dot{\alpha}}}$ $C_{m_{\dot{\alpha}}}$ $C_{n_{\dot{\alpha}}}$ $C_{l_{\dot{\beta}}}$ $C_{m_{\dot{\beta}}}$ $C_{n_{\dot{\beta}}}$	<u>Cross Derivatives</u>
		<u>Cross-Coupling Derivatives</u>
		$C_{l_p} + C_{l_{\dot{\beta}}} \sin \alpha_0$ $C_{m_q} + C_{m_{\dot{\alpha}}}$ $C_{n_r} - C_{n_{\dot{\beta}}} \cos \alpha_0$
		$C_{l_r} - C_{l_{\dot{\beta}}} \cos \alpha_0$ $C_{n_p} + C_{n_{\dot{\beta}}} \cos \alpha_0$
		$C_{l_q} + C_{l_{\dot{\alpha}}}$ $C_{m_r} - C_{m_{\dot{\beta}}} \cos \alpha_0$ $C_{m_p} + C_{m_{\dot{\beta}}} \sin \alpha_0$ $C_{n_q} + C_{n_{\dot{\alpha}}}$

The second category of Table 2.2 are those derivatives due to the time rate of change of the two aerodynamic angles ($\dot{\alpha}$, $\dot{\beta}$). These are aerodynamically equivalent (in the linear domain) to derivatives due to translational acceleration in the same plane of motion and thus are referred to as *translational acceleration* derivatives. For example, the aerodynamic reactions due to vertical acceleration ($\ddot{z}$) are equivalent to those due to the time rate of change in the angle of attack ($\dot{\alpha}$), and similarly, the lateral acceleration ($\ddot{y}$) is related to the time rate of change in the angle of sideslip ($\dot{\beta}$). It is these acceleration derivatives that are the items of concern in this thesis.

For a model oscillating around a single fixed axis in a wind tunnel experiment (angular forced-oscillation), the result is a sum of derivatives:

- i) due to pure rotation about that axis, and
- ii) due to the time rate of change of angular deflections.

In the oscillation-in-pitch tests, for example, the combination of effects due to q and $\dot{\alpha}$ result in the *composite* form of the damping-in-pitch derivative, $C_{m_q} + C_{m_{\dot{\alpha}}}$ (see Fig. 2.2). As explained by Orlik-Rückemann in Reference 3: "For a free-flying aircraft the variations in the angle of pitch and in the angle of attack can occur independently of each other, and each gives rise to a different longitudinal distribution of the normal velocity (Fig. 2.2). The distribution due to the angle-of-pitch variation (ie. due to the pitching velocity q) varies along the chord and intersects zero at the axis of rotation, while the distribution due to the angle-of-attack variation is constant along the chord. In the case of an oscillation around a fixed axis, both variations occur at the same time and even if the two variables q and $\dot{\alpha}$ are themselves equal, their effects are different and have to be superimposed."

$$\Delta\alpha \neq 0 \quad q \neq 0$$

Measures $C_{mq} + C_{m\dot{\alpha}}$

(a) Angular forced-oscillation

$(\alpha_0 = 0)$

$$\Delta\alpha = \frac{\dot{z} \cos \alpha_0}{V_\infty} \quad q = 0$$

$$\Delta\dot{\alpha} = \frac{\ddot{z} \cos \alpha_0}{V_\infty} \text{ Measures } C_{m\dot{\alpha}}$$

Normal velocity distribution

(b) Translational oscillation (note: For all α_0 , component of z body axis translational oscillation $\perp V_\infty$; i.e., $\dot{z} \cos \alpha_0$, is defined "plunging").

$$\Delta\alpha = 0$$

$$q \neq 0$$

Measures C_{mq}

Normal Velocity distribution

(c) Pure rotation (pitching)

Fig. 2.2 Longitudinal motions of the model (after Ref.18)

In the lateral plane, a yawing or rolling motion around a fixed axis produces a combination of effects due to r and $\dot{\beta}$, and p and $\dot{\beta}$ respectively, resulting in such composite forms as $C_{n_r} - C_{n_\beta} \cos \alpha_0$ and $C_{n_p} + C_{n_\beta} \sin \alpha_0$. For a body-fixed system of axes, where the mean angle of attack (α_0) is not zero, the expression involves a trigonometric function of α_0 .

Chapter 3

SURVEY OF WIND TUNNEL TECHNIQUES FOR DETERMINATION OF TRANSLATIONAL ACCELERATION DERIVATIVES

3.1 BACKGROUND

As already discussed, an angular forced-oscillation wind tunnel test yields a composite derivative which is the sum of two effects; angular rate and translational acceleration (such as $C_{m_q} + C_{m_{\dot{\alpha}}}$). In the past, the acceleration derivatives have been neglected in performing dynamic stability calculations because:

- i) at the low angles of attack and symmetric flow conditions associated with flight of conventional straight-wing-tailed aircraft, estimates indicated that the acceleration effect was negligible; all of the effect was attributed to the angular rate, and
- ii) little information was available to permit reasonably accurate estimates of the magnitude of the α and β effects at high angles of attack. As a result, the entire composite derivative was used to represent the purely rotary derivative in the equations of motion.

As the NACA began to investigate the characteristics of the dynamic derivatives of swept and delta wings in the 1950's, it was discovered that large, nonlinear values of the lateral acceleration-in-sideslip derivatives were obtained at high angles of attack (Refs. 4 to 9). This is clearly illustrated in Figure 3.1 (Ref. 7), where the variation of damping-in-yaw with angle of attack are shown for a 45° swept-wing-fuselage-tail combination at very low speeds. The discrepancy between the purely rotary derivative C_{n_r} and angular forced-oscillation derivative $C_{n_r} - C_{n_{\dot{\beta}}}$ is attributable to a large increase in the value of $C_{n_{\dot{\beta}}}$. As the angle of attack increases, various time-lag effects usually associated with flow separation and asymmetric vortex sheddings cause the "unsteady" (acceleration) effects to become more prominent.

There is a renewed interest in acceleration derivatives due to their predicted significance at high angles of attack, and there are several reasons why it is of vital importance to know their value:

(Results in stability system of axes)

Fig. 3.1 Comparison of pure rotary with composite derivative for a 45° swept-wing-fuselage-tail combination (after Ref.7)

- i) The inability to separate the $\dot{\alpha}$ and $\dot{\beta}$ derivatives obtained from the angular forced-oscillation tests hinders our ability to accurately model the aircraft dynamics. This was first shown years ago (Ref. 4) when calculations were made of the dynamic lateral stability of a 60° delta-wing model aircraft to investigate the effect of varying the value of the rate and acceleration derivatives one at a time. Gross differences in stability occurred at high angles of attack (20°–30°) when $C_{n\dot{\beta}}$ and $C_{l\dot{\beta}}$ were included and neglected, and these effects on stability were different from the effects when the individual $\dot{\beta}$ derivatives were assumed zero and the composite sums used to represent the purely rotary derivatives. More recent *sensitivity studies*, reviewed by Orlik-Rückemann in References 10 and 11, show that the derivatives $C_{l\dot{\beta}}$ and possibly $C_{l\dot{\alpha}}$ are significant in predicting modern military aircraft motion at high angles of attack and that $C_{m\dot{\alpha}}$ is significant for both aircraft and missiles. It was also concluded that in both the longitudinal and lateral cases it is crucial that acceleration derivatives be included in the equations of motion separate from their rate counterparts.
- ii) Special cases exist where the separate treatment of acceleration derivatives is necessary. Recall that the effects due to $\dot{\alpha}$ are aerodynamically equivalent (in the linear domain) to those due to vertical acceleration, since $\dot{\alpha} = \ddot{z} \cos \alpha_0 / V_\infty$, and effects due to $\dot{\beta}$ are equivalent to those due to lateral acceleration, thus, if an aircraft is designed to use direct-lift or direct side-force controls, the derivatives due to $\dot{\alpha}$ and $\dot{\beta}$ become important.

The present review attempts to outline and discuss the variety of wind tunnel techniques that have been used in determining translational acceleration derivatives for aircraft and includes a brief summary of pertinent results.

3.2 WIND TUNNEL TEST TECHNIQUES

It is important to realize that the calculated aircraft motions are meaningless unless the stability derivatives are representative of the aircraft under consideration. Extracting acceleration derivatives from controlled flight tests of the full scale aircraft is difficult and has met with limited success. For example, Reference 12 describes the results of appropriately designed rolling maneuvers performed in a T-37B jet aircraft which allow separate estimation of $C_{m\dot{\alpha}}$ and $C_{m\dot{\beta}}$, yielding consistent, reasonable estimates of $C_{m\dot{\alpha}}$. Although this information would be very useful for final verification of predicted aerodynamic characteristics, it is too late to affect the design of a new aircraft. As a result of the extremely complex aerodynamic phenomena involved (outlined in Section 3.3), valid

theoretical methods for the prediction of α and β derivatives at high angles of attack are yet to be developed. The methods remaining are the various ground test facilities. It is mandatory that experiments be conducted at a Reynolds number representative of actual flight because flow separation and vortex shedding are functions of viscous effects and it is felt that there is probably a strong dependence of stability derivatives on Reynolds number. This criterion selects the wind tunnel as the most suitable facility because it enables higher Reynolds number testing than other ground facilities, although most tunnels cannot give Reynolds numbers typical of flight conditions. Thus the most reliable source of information on acceleration derivatives has been captive model wind tunnel tests of scaled aircraft models.

This Section describes the unique wind tunnel test methods utilized over the years for prediction of acceleration derivatives, including the object, advantages and limitations of each technique. References are cited for additional background and details.

Acceleration derivatives have been measured in two ways in a wind tunnel:

- i) *Directly*, by simulating a pure translational motion with the angular rate equal to zero, or
- ii) *Indirectly*, by measuring the purely rotary derivative and then subtracting it from the composite derivative obtained in angular forced-oscillation tests (for the same model under the same test conditions).

3.2.1 INDIRECT MEASUREMENT TECHNIQUES

CURVED-FLOW (VPI)

One technique which has been developed to obtain pure rotary derivatives due to pitching (C_{m_q}) and yawing (C_{n_r}, C_{l_r}) in a steady-state environment is to use a wind tunnel with curved-flow through the test section. The $1.83m \times 1.83m$ ($6ft \times 6ft$) curved-flow test section of the Langley "Stability Wind Tunnel", a unique facility designed and operated at NASA Langley Research Center (LRC) in the early 1940's, is presently in use at the Virginia Polytechnic Institute and State University (VPI & SU), which acquired the tunnel in 1958.

Simulation of a steady curved-flight condition with a rigidly mounted model in the test section requires that the relative motion existing between the aircraft and air stream in curved-flight be reproduced. This is accomplished if two conditions are met:

- i) the airflow must follow a circular path in the region near the model, and
- ii) the velocity variation normal to the circular streamlines must be directly proportional to the local curvature of the flow ($V_{\infty} \propto R$).

The air stream is curved by means of the tunnel's flexible side walls which can be deformed into a curve. The second condition is achieved by using vertical-wire drag screens in the flow upstream of the test section. Six screens having different wire diameter and spacing are incrementally used to obtain the desired velocity distribution required for each increment in flow curvature. A sketch of a typical arrangement for yawing tests is shown in Figure 3.2. The model may also be mounted from the side wall for pitching tests. Details of curved-flow theory, calibration and tunnel operation are given in Reference 13; data acquisition and reduction in Reference 14.

Having the model fixed in the test section has the important advantage that standard wind tunnel procedures can be used for all measurements and blockage corrections. In addition, the steady-state environment permits ease of implementing flow visualization techniques. The range of angle of attack that can be tested is limited only by the method used for support.

It should be noted, however, that curved-flow tests can only be used for low speed work and even there it has certain disadvantages. A significant side static pressure gradient exists contributing to a buoyancy force on the model, which does not exist in curved-flight, introducing the need for correction procedures (Ref. 14). Also there is a high level of turbulence behind the drag screens which is graded according to the spacing of the wires, but the influence of this on the aerodynamic characteristics of the model appears to be small.

When the tunnel was at NASA LRC, the curved-flow technique was used to determine the yawing derivatives (C_{n_r}, C_{l_r}) of simple wing-body shapes, which resulted in the separation of the acceleration derivatives ($C_{n_{\dot{\alpha}}}, C_{l_{\dot{\alpha}}}$) from the classical combinations $C_{n_r} - C_{n_{\dot{\alpha}}} \cos \alpha_0$ and $C_{l_r} - C_{l_{\dot{\alpha}}} \cos \alpha_0$ obtained from angular forced-oscillation testing.

ROLLING-FLOW (VPI)

The curved-flow test section of the low-speed Stability Wind Tunnel at VPI is interchangeable with a round (1.83m(6ft) in diameter) or rolling-flow test section, where the pure rotary derivatives due to rolling (C_{l_r}, C_{n_r}) can be measured in a steady-state environment.

Fig. 3.2 Curved-Flow Technique. VPI. (Ref.28)
 (Dimensions in centimeters and parenthetically in inches)

Fig. 3.3 Rolling-Flow Technique. VPI. (Ref.14)

In order to simulate a rolling-flight maneuver with a rigidly mounted model, two conditions must be satisfied:

- (i) the airflow must rotate about the model, and
- (ii) the velocity component normal to the tunnel axis must increase in proportion to the radial distance from the centerline.

A large variable speed rotor located upstream of the test section is used to impart a rotary motion to the flow. The rotor vanes are specially shaped to provide a velocity distribution which simulates a solid vortex, a flow field *similar* to that experienced by a rolling aircraft in level flight (Fig. 3.3). Reference 15 presents a more detailed description and results of current calibration of the rolling-flow test section.

Due to the centrifugal force, there is a radial pressure gradient in the air during rolling-flow tests which does not exist when rolling an aircraft in flight. For a symmetrical model which is symmetrically mounted, this pressure gradient is symmetric about the tunnel centerline; its effects tend to cancel out and no correction is applied. In other cases corrections are needed. The comments for curved-flow tests with respect to the advantages of fixed model testing apply to rolling-flow tests as well.

Using this method, the derivatives C_{l_p} , C_{n_p} may be obtained over an extensive range of angle of attack for zero and nonzero values of sideslip, resulting in the separation of the acceleration derivatives ($C_{l_{\dot{\beta}}}$, $C_{n_{\dot{\beta}}}$) from the composite sums $C_{l_p} + C_{l_{\dot{\beta}}} \sin \alpha_0$ and $C_{n_p} + C_{n_{\dot{\beta}}} \sin \alpha_0$.

FORCED-SNAKING (NASA LRC)

A *combined-motion* oscillatory test method developed and previously used at the NASA LRC many years ago was known as forced- snaking (Refs. 16,17). In this method, the model performed a combination of lateral and yawing oscillations in such a way as to simulate pure yawing motion in a conventional straight-wall wind tunnel. The oscillation was in the x - y plane, during which the lateral velocity component at the aircraft center of gravity was maintained at zero, a condition fulfilled when $V_\infty \sin \psi = \dot{y}$; i.e., the aircraft is always heading into the relative wind (see sketch in Fig 3.4). From the resulting "snaking" motion, pure rotary derivatives C_{l_r} and C_{n_r} have been successfully determined.

The apparatus is shown schematically in Figure 3.5. The support system for the model and strain-gauge balance consisted of a large streamline tube. The tube was supported

Fig. 3.4 "Snaking" motion (after Ref.16)

Fig. 3.5 Forced-Snaking Apparatus for simulation of pure yawing.

NASA LRC. (VPI). (Ref.16)

at each end by counter-rotating flywheels, driven by means of several shafts, gears and a variable frequency motor-generator. Reference 16 contains a detailed description of the design and function of the instrumentation employed.

The forced-snaking test rig was also transferred to VPI and has not been in operation since. The NASA LRC has recently sponsored research at VPI to renovate and apply the dynamic test capabilities of the Stability Tunnel (Ref. 15). To date, the curved and rolling-flow test methods have been put back into operation (as previously mentioned), and work is under way to update the oscillatory test capabilities as well.

3.2.2 DIRECT MEASUREMENT TECHNIQUES

LATERAL-OSCILLATION APPARATUS (NASA LRC)

A novel concept originating at the NASA Langley Research Center in the mid 1950's led to the development of an apparatus which provided a linear sidewise oscillatory motion, permitting the first *direct* measurements of the β derivatives. Data were obtained over a wide range of angle of attack by forcing a model to perform a lateral oscillation across the jet of the $1.83 \times 1.83m$ test section of the old NASA Langley-Stability Tunnel (Fig. 3.6).

The oscillating part of the apparatus (the support strut) consisted of a $2.7m$ ($9ft$) length of streamlined steel tubing of approximately $10.2cm$ ($4in$) chord which spanned the tunnel jet and extended through the walls of the test section. A push rod mounted off-center on a rotating high inertia flywheel supplied the drive to the support strut to oscillate the model in either the free-oscillation (Ref. 6) or forced-oscillation (Ref. 8) modes. Due to the complexity of the problem, the effects of flow interference from the support strut (which may be substantial at high angles of attack) were not known, and thus not corrected for. A *pure* lateral oscillatory motion could not be produced by this apparatus, but, the streamwise motion of the model was minimized to an extent that it was considered negligible by restricting the lateral motion to a maximum amplitude.

The rolling and yawing moments were measured by a two component strain-gauge balance located at the model center of gravity, and provided reliable values of the acceleration-in-sideslip derivatives $C_{n\dot{\beta}}$ and $C_{l\dot{\beta}}$ of simple wing-fuselage shapes.

Since its transfer to VPI, the Lateral-Oscillation Apparatus has been inoperative but as indicated earlier, plans are in progress to update the test rig.

Fig. 3.6 Lateral Oscillation Apparatus. NASA LRC.(VPI). (Ref.8)

Fig. 3.7 Forced-Oscillation 2 Degree of Freedom Apparatus.
Calspan. (Ref.18)

FORCED-OSCILLATION 2 DEGREE OF FREEDOM APPARATUS (CALSPAN)

In the early 1960's the U.S. Air Force Aeronautical Systems Division instituted an investigation at the Cornell Aeronautical Laboratory (now known as Calspan Corporation) to devise a method for measuring dynamic derivatives for aircraft through wind tunnel tests. The result of this was the development of a unique oscillatory mounting system capable of separating and measuring directly the individual components of the angular damping-in-pitch derivative $C_{m\dot{q}} + C_{m\dot{\alpha}}$ (Ref. 18). This was accomplished by inexorably forcing ¹ a model sinusoidally in a pure pitching (q effects) or pure plunging ($\dot{\alpha}$ effects) motion. Lateral dynamic tests at small angles of attack could also be performed by rotating the model by 90° on its support, but these motions were never investigated.

Figure 3.7 illustrates the model support and mechanical drive system. The sting support for the model formed one arm of the forcing linkage which was driven by connecting rods mounted eccentrically on two high-inertia flywheels. A plunging motion was obtained by controlling the eccentric amplitude and the angular relation between the flywheels. Motion sensors consisted of linear accelerometers in the model for measurement of vertical motion, and a conventional strain-beam balance system between the model and support sting for measurement of moments. The mechanical oscillator operated in a frequency range from 3 to 12 Hz, and at that frequency could provide a 0.3m (1ft) peak-to-peak vertical displacement and translational acceleration up to 20 g's with a normal load limit of 5300N (1200lbs) at the model center of gravity. Further details on the design and operation, signal processing and data analysis are provided in Reference 18.

Tests on an F-80 model and F-104 fighter model performed in Calspan's 2.4m (8ft) Transonic Wind Tunnel permitted, for the first time, a direct measurement of the pitching moment due to rate of change of angle of attack ($C_{m\dot{\alpha}}$). Unfortunately, errors in the measurement of the magnitude and phase of the extraneous rotational motions contributed heavily to error in the measurement of $C_{m\dot{\alpha}}$ during plunging tests. The static angle of attack could be adjusted by using a bent sting so that the model could oscillate about various angles of attack, however, all tests were conducted with the model at its trim mean angle of attack.

The Calspan apparatus has not been operative since and there are no definite plans for further tests regarding acceleration derivatives.

¹ using a constant oscillatory displacement

HALF MODEL PLUNGING APPARATUS (NAE)

Plunging oscillation experiments using a half-model technique were successfully performed at the Unsteady Aerodynamics Laboratory of the NAE in an attempt to determine information about the pitching moment due to translational acceleration ($C_{m,\dot{z}}$) for a delta-wing aircraft (Refs. 19, 20). The principle of operation is to force the half model to oscillate sinusoidally in a translational motion in one *primary* degree of freedom.

The NAE half model apparatus is shown schematically in Figure 3.8. The model is mounted on a balance consisting of a single pitch-sensitive cruciform flexure, which is attached to a double-cantilever spring allowing the model to translate in the vertical direction. An electromagnetic drive is composed of a pair of coils inside magnetic fields generated by a pair of electromagnets. These coils carry an alternating current and are connected to the balance, thus applying the necessary driving force to produce a plunging oscillation. The entire apparatus is mounted on a circular wind tunnel plate which can be rotated to vary the model angle of attack.

The use of the half-model technique for oscillation experiments offers certain advantages over full models (Ref. 21), namely:

- i) the technique is free of any sting or strut-interaction and sting oscillation effects;
- ii) a larger size model can be employed, resulting in a higher Reynolds number, and
- iii) with the half model mounted on the tunnel wall, an oscillatory test rig can be located outside the tunnel resulting in a simple, more efficient design in comparison with internal balances and test rigs.

On the other hand, the technique introduces the possibility of flow interference effects due to the presence of the tunnel wall boundary layer, and to the existence of a gap between the model and tunnel wall introduced by the method of mounting (Ref. 21). To ensure the reliability of the results, these sources of error must be minimized. Boundary layer effects in the NAE apparatus are significantly reduced by mounting the model on a reflection plate displaced from the tunnel wall (Fig. 3.8). The reflection plate is fixed with respect to the tunnel wall, the model being rotated with respect to the plate. The effect of the gap is diminished by narrowing the gap between model and reflection plate as much as possible. It should also be noted that the use of the half-model technique is limited to symmetric flow conditions thus permitting a maximum angle of attack of approximately 20° (Ref. 20).

Preliminary measurements of $C_{m_{\alpha}}$ for a delta-wing aircraft are available from tests performed in the NAE $0.75m \times .40m$ intermittent suction wind tunnel. The half model apparatus is fully operational, although there are no immediate plans for future testing.

TRANSLATIONAL OSCILLATION APPARATUS (NAE)

A full model, sting-mounted translational oscillation apparatus is available at the NAE to determine all moment derivatives due to either lateral or plunging translational acceleration (Ref. 22).

The apparatus and model shown in Figure 3.9 are oriented for a lateral oscillation (to determine derivatives due to β). By rotating the apparatus by 90° against both the model and the sting support, derivatives due to α can be obtained.

As in the half model case, the model is mounted on a balance that is in turn attached to an elastic support system in the primary degree of freedom. In this case, the support is a multileaved cantilever flexure (from Ref. 22)... "characterized by a relatively high compliance in the direction perpendicular to the plane of the leaves, allowing the balance (and the model) to perform an oscillatory translational motion in that direction. Such a flexure is also very stiff in other degrees of freedom, which is necessary to render any motion in the other degrees of freedom negligibly small and to withstand the aerodynamic loading in these other degrees of freedom." The balance is made of a single piece featuring four components for measuring the angular deflections perpendicular to and in the plane of oscillation. These motions are sensed by semi-conductor strain-gauges.

The oscillatory motion is imparted to the model by an electromagnetic drive mechanism. A constant magnetic field is established by twelve rare earth magnets situated on both sides of a longitudinal slot in the sting, and a "single turn" coil carries an alternating current through the magnetic field generating an alternating electromagnetic force which oscillates the model in the primary degree of freedom.

Details on data acquisition and reduction can be found in Reference 22, but basically, the system is fully digital, based on a minicomputer, with extensive use of a cross-correlation technique in order to extract signals that can be buried in aerodynamically and mechanically induced noise.

Fig. 3.8 Half Model Plunging Apparatus. NAE. (Ref.20)

**Fig. 3.9 Front part of the Translational Oscillation Apparatus. NAE.
(Lateral setup). (Ref.22)**

Fig. 3.10 Orbital Fixed-Plane Motion. NAE. (Ref.23)

To date, preliminary data on the lateral acceleration derivatives ($C_{l\beta}$, $C_{n\beta}$, $C_{m\beta}$) are available for a delta-wing aircraft configuration. The use of sting-mounting permits testing of more realistic aircraft models, however, the derivatives obtained may easily be affected by *dynamic sting interference*². An additional drawback is that the apparatus has a relatively small normal load capacity that prevents its use for experiments at the high angles of attack required in the current test program at NAE.

ORBITAL FIXED-PLANE MOTION (NAE)

An apparatus based on the concept of orbital fixed-plane motion is now in the design stages at the NAE (Refs. 23,24). The term "orbital fixed-plane" motion denotes motions where the model is rotated in such a way that its transverse axis is constrained to lie in a plane which maintains a fixed orientation in an inertial reference system. One special case of that motion - translational motion - is depicted in Figure 3.10 and described as follows (Ref. 3): "The model performs a combination of a vertical oscillation and a lateral oscillation. The two motions are 90° out of phase. No rotation is involved. All the position angles are constant and all body angular velocities are zero. Only the aerodynamic angles are variable. With proper, rather complex, experimental procedures the dynamic derivatives due to α and β can be determined".

A complete discussion on the dynamic stability apparatus, data acquisition and reduction, and applications of the principle can be found in References 23, 24.

3.3 SUMMARY OF RESULTS

3.3.1 EARLY RESULTS

In view of the lack of understanding and the apparent magnitude and importance of the acceleration derivatives, the NACA and NASA launched an extensive research program in the early 1950's in an attempt to isolate the individual β derivatives on aircraft or components of aircraft through a range of angle of attack. The principal findings of these investigations can be summarized briefly as follows:

² Dynamic sting interference is the effect of support sting oscillation on measured stability derivatives, and will be discussed in depth in section 4.2.3.

Sources of the β derivatives

Wind tunnel studies at high angles of attack revealed that *lag of flow separation and reattachment* on highly swept wings and *lag of sidewash* at the vertical tail are the two physical flow phenomena responsible for the β derivative.

Qualitative examination of a series of tuft-grid flow visualization studies (Ref. 25) indicated that at high angles of attack swept and delta wings (especially delta) form strong vortices emerging from the apex of the wing. Under oscillatory conditions (in this case yawing), the vortical flows lag behind the dynamic motion of the wing and thus introduce effects not accounted for in a quasi-steady representation. The result is a flow field which produces a rolling and yawing moment due to rate of change of sideslip, i.e., the existence of purely *unsteady* effects.

The studies on the delta wing of Reference 25 also showed that the vortex flow from the leading wing panel moved inboard and created a sidewash over the rear portion of the model. Under oscillatory conditions, this sidewash *also* lags behind the dynamic motion of the model so that at any instant the vertical tail (tests of Ref. 5) is at a higher angle of attack than it would have been with no sidewash present. The derivative $C_{n\dot{\beta}}$ provides a first order model of the effects of this sidewash lag on the yawing moment.³

Reference 26 further examines the lag phenomenon during angular forced-oscillation-in-yaw tests and includes a comprehensive discussion of the probable cause of the large magnitude of the acceleration derivatives at the high angles of attack. In short, the data indicated that an increase in magnitude with increasing angle of attack is not caused by an increase in the time lag, but is owing to the increased strength of the vortex flows.

Effect of Wing Planform

The values of $C_{n\dot{\beta}}$ and $C_{l\dot{\beta}}$ measured with the Lateral Oscillation Apparatus for a 60° delta, a 45° swept and an unswept wing (Ref. 6) are presented in Figure 3.11 as functions of angle of attack. The effect of wing planform on the β derivative is small in the low angle of attack range. In the higher range, the values for the 60° delta wing develop at a higher angle of attack and increase more gradually than those for the 45° swept wing, but reach higher values at the maximum test angle of attack. The β derivatives for the unswept wing

³ This is analogous to the classical "lag of the downwash" and its effect on the damping-in-pitch of a horizontal tail (see Ref. 1).

planform are of small magnitude and relatively constant in comparison. The leading-edge sweep affects the strength of wing vortices and also the location at which they separate, and it appears that the $\dot{\beta}$ derivatives evolve in a pattern proportional to the degree that the vortices are detached from the wing surface. These results are substantiated in References 17 and 27 for the same wing models.

Effect of Vertical Tail

Past measurements of the damping-in-yaw derivative ($C_{n_r} = C_{n_{\dot{\beta}}} \cos \alpha_0$) for a 45° swept wing (Ref. 5) display a definite increase in damping at high angles of attack with the addition of a vertical tail. As illustrated in Figure 3.12, results of direct measurements with the Lateral Oscillation Apparatus on a 60° delta wing (Ref. 8) indicate that at angles of attack above 30° addition of a vertical tail to the fuselage tended to increase the value of $C_{n_{\dot{\beta}}}$ as compared with the slight decrease occurring for the wing alone. It is evident that the contribution of the vertical tail to damping-in-yaw can be attributed to an increase in the magnitude of $C_{n_{\dot{\beta}}}$ which, as noted earlier, is dependent upon the influence of the lag of the sidewash on the vertical tail.

Effect of Frequency and Amplitude of Oscillation

A series of lateral oscillation experiments on delta and swept wings (Refs. 6, 8, 27) indicate, in general, that the magnitude of $C_{n_{\dot{\beta}}}$ and $C_{l_{\dot{\beta}}}$ are very dependent on the frequency of the oscillatory motion. The largest changes in the variation of the derivatives with angle of attack occur for the lower frequencies; as the frequency increases, its effects become smaller and the derivatives more linear. The results of one such investigation (Ref. 27) for a delta wing comparing frequencies of 1 and 3.31 Hz ($\omega = 0.066$ and 0.218 respectively) are shown in Figure 3.13(a).

At 3.31 Hz, varying the amplitude of oscillation from $\pm 2^\circ$ to $\pm 6^\circ$ has only a minor effect on $C_{n_{\dot{\beta}}}$, $C_{l_{\dot{\beta}}}$ (Fig. 3.13(b)). Unfortunately, lower frequencies were not investigated, however, considerable effects of amplitude on the composite damping in yaw derivative (Ref. 9) exist at much lower frequencies of $\omega = .033$ (not shown). It would appear that the alternating increase and decrease in the separating flow on the wing produces moments out-of-phase with the model oscillation and causes the derivatives to become functions of frequency and possibly amplitude of oscillation.

Fig. 3.11 Effect of wing planform. (Ref.6)

Fig. 3.12 Effect of vertical tail. (after Ref.8)

Fig. 3.13(a) Effect of frequency. (Ref.27)

Fig. 3.13(b) Effect of amplitude. (Ref.27)

Fig. 3.13 Effect of frequency and amplitude of oscillation for a 60° delta wing

One final observation is that all data confirm that acceleration derivatives at low angles of attack are very small and thus negligible. A good account of the experience obtained by NACA and NASA on low speed lateral-directional derivatives including the effects of β is given in Reference 28, along with a chronological listing of publications pertaining to related subject matter.

3.3.2 RECENT RESULTS

A diminished interest in dynamic stability was responsible for the lengthy gap in research after the 1950's, however, interest was revived with the introduction of new aircraft configurations which maneuvered at very high angles of attack. There was a need for information, including the effects of acceleration derivatives on stability, over a much larger range of angle of attack than previously considered. The NASA is currently engaged in a wide research program, part of which sponsors testing at VPI. Simultaneous research is being conducted at the NAE.

VPI

The previous NACA investigations at the NASA LRC using curved- and rolling-flow tests were primarily of simple wing-fuselage combinations, but recent tests at the VPI facility have been conducted for three current full model fighter configurations, with particular attention given to comparing the purely rotary derivatives with the composite derivatives obtained through angular forced-oscillation tests. To date, the F-5 and F-18 configurations have been tested in both curved and rolling-flow (Refs. 29 to 32); the F-4 in curved-flow (Ref. 33).

As an example, a comparison of yawing stability derivatives for the F-4 and F-5 models is shown in Figure 3.14(a) (Refs. 33, 34). The difference between each set of curves denotes the unsteady effects due to β . For the F-4 model, the curved-flow and forced-oscillation results exhibit opposite trends above the stall whereas the F-5 model shows similar trends for both curved-flow and forced-oscillation tests. The unsteady effects are certainly more prominent for the F-4's greater leading-edge sweep, which agrees with earlier NASA findings.

The most recent rolling-flow data compared with the composite sums for the F-5 configuration are shown in Figure 3.14(b), illustrating the approximate magnitudes of the $C_{n\dot{\beta}}$ and $C_{l\dot{\beta}}$ derivatives (Ref. 34).

F-4 (Ref.33)

F-5 (Ref.34)

Fig. 3.14(a) Comparison of curved-flow and angular forced-oscillation tests for F-4 and F-5 models

Fig. 3.14(b) Comparison of rolling-flow and angular forced-oscillation tests for F-5 model.(Ref.34)

Fig. 3.14 Effect of β derivatives

Fig. 3.15 Pitching moment derivative due to vertical acceleration. (Ref.19)

Fig. 3.16 Preliminary results for a cropped delta-wing-and-flap-planform model. (Ref.22)

NAE

Up to this point, all efforts have been focused on the isolation of β derivatives, with the exception of the Calspan apparatus whose estimate of C_{m_α} , although noteworthy, was restricted to the model's trim mean angle of attack.

The preliminary measurements of C_{m_α} using the NAE half model apparatus are shown in Figure 3.15 for a delta-wing configuration at Mach number 0.7 (Ref. 19). As would be expected, the results for angles of attack greater than 20° may be unreliable due to the limitations of the half-model technique discussed earlier. The reason for the unusual asymmetry of C_{m_α} about the $\alpha = 0$ axis is not certain and future testing is required to further examine the problem. This data, although preliminary, provides a basis for comparison with results of future full model plunging experiments.

The magnitude and nonlinear nature of the lateral acceleration derivatives, as determined with the NAE translational oscillation apparatus, is illustrated by the preliminary results given in figure 3.16 (Ref. 22). Lateral oscillation experiments were conducted through a range of angle of attack from 0° to 35° on a cropped delta-wing-and-fin-planform model under a joint research program with NASA Ames. It is interesting to note that the measurement of the cross-coupling acceleration derivative $C_{m_{\dot{\alpha}}}$ is the first of its kind. These particular derivatives were thought to be insignificant, but recent full body flow visualization motion picture studies made in a water tunnel at the NAE (Ref. 10) indicated the presence of aerodynamic cross-coupling caused by the flow asymmetries at high angles of attack.

3.4 CONCLUDING REMARKS

Although the results of the foregoing discussion serve to illustrate the importance of acceleration derivatives at high angles of attack, it is clear that they are fragmentary, and that many questions regarding the measurement of translational acceleration derivatives remain unanswered.

Consistent results are needed to provide confidence that a particular technique used is capable of reliable predictions, thus it is important to test a standard configuration under similar conditions at different facilities and through comparison of data, verify the validity of the techniques used. There also is a need for a dynamic calibration technique which would independently confirm the validity of the experimental procedure used.

It is felt that more attention must be given to the *oscillatory* methods of obtaining ac-

celeration derivatives in view of the differences between curved or rolling-flow over a fixed model and an aircraft in curved or rolling-flight. For example, there have been discrepancies encountered between oscillatory forced-snaking and steady-state curved-flow tests (Ref. 16), probably due to substantial differences occurring in the basic aerodynamic flow phenomena acting on the model under oscillatory conditions. In particular the curved and rolling flows give a steady flow situation; the oscillatory methods do not, and amplitude and frequency of oscillation may even affect the results for derivatives C_{m_q} , C_{n_r} , C_{l_p} .

Finally, efforts should be maintained to correct for and minimize support interference and dynamic sting interference, although it can be appreciated that this is a difficult task.

Chapter 4

NAE FORCED-OSCILLATION APPARATUS FOR TRANSLATIONAL OSCILLATION EXPERIMENTS

4.1 DESIGN

As reviewed in Chapter 3, the original translational oscillation apparatus (Ref. 22) developed at the Unsteady Aerodynamics Laboratory (UAL) of the NAE was incapable of operation at the higher angles of attack required in the current test program, due to its small normal force capacity. In addition, translational motion was, to some extent, contaminated with angular oscillation and hence acceleration derivatives had to be determined after accounting for angular oscillation effects. Subsequently, the concept for a new and improved full model, sting-mounted apparatus was proposed in 1980 and detail design and fabrication carried out during the following year. The author was not a participant during the initial design stages, however, a concise description of operation, mechanical design and construction of the new apparatus is furnished.

4.1.1 PRINCIPLE OF OPERATION

The principle of operation is based on the *direct forced-oscillation technique*. In this technique, a preselected sinusoidal motion—designated “primary motion”—is imparted to the aircraft model forcing it to oscillate at a constant amplitude in a single DOF—the “primary DOF”. Recall that under asymmetric flow conditions, aerodynamic coupling may occur between the primary DOF and the remaining “secondary DOFs”. The present apparatus provides a primary motion in translational oscillation (along either z or y body axes) resulting in induced aerodynamic reactions in the secondary angular DOFs, pitch, yaw and roll. These “secondary motions” are measured, leading to the required cross and cross-coupling derivatives. More specifically, for primary oscillation along the z axis, analysis of the primary motion yields the direct force derivative C_{Z_α} and analysis of the secondary motions in pitch, yaw and roll yields the cross and cross-coupling moment derivatives, for this experiment, C_{m_α} , C_{n_α} , C_{l_α} , which are the desired acceleration derivatives. Similarly, for primary oscillation in lateral translation, analysis of the primary motion yields the

direct force derivative $C_{Y\beta}$ and analysis of the secondary motions yields the cross and cross-coupling acceleration derivatives $C_{n\beta}$, $C_{m\beta}$, $C_{l\beta}$.

4.1.2 DESIGN CRITERIA

The design of the apparatus was influenced primarily by the following conditions:

- (i) it had to be capable of testing the simplified F-16 model configuration ("Standard Dynamics Model" (SDM)—described in Chapter 6) used in the current test program at NAE;
- (ii) it had to be capable of performing tests at Mach 0.7, at angles of attack from 0° to 40° and at two angles of sideslip;
- (iii) it had to be designed for use in the NAE $0.75m \times 0.40m$ Dynamics Wind Tunnel.

The abovementioned conditions resulted in specific design criteria:

- (i) the apparatus should be capable of sustaining the following loads:

Normal force — 800 N

Pitching moment — 7 Nm

Sideforce — 130 N

Yawing moment — 7 Nm

Rolling moment — 4 Nm

- (ii) it should provide a primary oscillation amplitude of $\pm 2.5\text{ mm}$ minimum, and
- (iii) it should provide a primary oscillation frequency range of 20 to 50 Hz , depending on model mass.

4.1.3 DETAILS OF MECHANICAL ARRANGEMENT

Figure 4.1 shows an exploded view of the apparatus in which the model is set up for translational oscillation along the z body axis.

A mounting spigot clamps the aircraft model to the forward end of a dynamic balance which supports the model. The balance is attached to an elastic suspension comprised of a pair of parallel cantilever beams. The aft ends of the cantilever beams are secured to a support sting while the forward ends are pin-joined to the balance by frictionless flexural

pivots, so that the beams deform identically under force allowing the model to perform translational motion. An electromechanical drive mechanism forces the model to oscillate at small amplitudes at the resonance frequency of the model/balance system.

By rotating the balance 90° about its longitudinal axis with respect to the model and sting, the primary motion becomes lateral translation. (Note: The sting is of cylindrical cross-section and not as shown in the Figure).

Suspension System

The shape and stiffness of the cantilever springs were selected so that the suspension features a relatively high flexibility in the primary DOF permitting translational model motion, while being stiff enough in the other DOFs to resist aerodynamic loading. The beams are of parabolic cross-section to evenly distribute the high stresses encountered. Also, the cantilever stiffness was chosen so as to ensure that the resulting natural frequency in the primary DOF would be far from the natural frequencies of the secondary DOFs, and in addition, to satisfy drive current requirements.

The major part of the stiffness in the primary DOF ($79,080 \text{ N/m}$) is provided by the cantilever suspension with other contributions from the sting, electrical conductors and various other components. As a consequence of the finite stiffness and model space constraints (45.7 mm model cavity), the desired 800 N normal force was not achieved. Allowing for oscillatory deflections of $\pm 2.5 \text{ mm}$ and allowing for clearance between the sting and model to accommodate model deflection under aerodynamic loads, the apparatus provides approximately 550 N normal force capacity. In order to achieve maximum normal force capacity during z axis translational oscillation, an electrical alarm has been installed to detect touching between the model and sting. The other load criteria specified are not critical from a stress point of view.

A strain-gauge bridge with long-life foil gauges is mounted on the top cantilever to measure the deflection in the primary DOF.

Dynamic Balance

A schematic illustration of the balance appears in Figure 4.2. The function of the balance is to detect the secondary motions induced by the primary translational oscillation. It is comprised of four components (flexures), machined in a single piece to avoid the slippage and alignment problems associated with mechanical joints, and is designed to fit within a hollowed out cavity in the model.

- 1 MODEL
- 2 BALANCE
- 3 FLEXURAL PIVOT
- 4 CONNECTOR
- 5 ELECTRICAL CONDUCTOR
- 6 BALANCE TONGUE
- 7 CANTILEVER SPRING (PARABOLIC PROFILE)
- 8 COIL
- 9 RARE-EARTH MAGNETS
- 10 STING

0 4 cm
approximate scale

Fig. 4.1 NAE electromechanically driven translational oscillation apparatus

■ STRAIN-GAUGE

Fig. 4.2 Schematic of the four component balance: (a) side view, (b) top view of the balance performing z axis translational oscillation

Each flexure is instrumented with a strain-gauge bridge and shaped to measure a particular secondary moment. In the case of z axis translational oscillation, in-plane bending or pitching moment is sensed by a single beam shaped flexure located on the longitudinal axis. The effective location of the strain-gauges of this flexure (electrical center) is designated as the apparatus moment reference center (RC). Yawing moment about the RC is resolved by sensing the out-of-plane bending at two beam flexures situated at 90° angles to, and located forward of the pitching beam. (Note that if the apparatus is set up for lateral translation, the pitching flexure becomes a yawing flexure, and the two yawing flexures would measure the pitching moment). Rolling moment corresponds to torsion in the balance. An axially oriented flexure with a cruciform section provides high resistance to axial forces and bending moments but low resistance to torsional moments. Such a flexure is located between the forward and aft yaw flexures with the strain-gauge bridge mounted on the main web of the cruciform.

In essence, the balance provides an electrical output which is proportional to the applied load at a particular flexure; i.e., the stress caused by an applied load produces a strain in the flexure and the strain-gauge converts this strain into an electrical output. Semi-conductor gauges were chosen for the balance because of their inherently high sensitivity resulting in an improved signal-to-noise ratio of the small secondary signals. The signal wires from all bridges run along the axis of the balance and support sting well shielded from the magnetic field of the drive by a steel braided sheath. Since the balance moves with the model, it senses all loads with respect to the model body axes.

A prime balance design consideration is to limit deflections from the secondary DOFs due to oscillatory loading. This criterion was satisfied by:

- (i) ensuring balance stiffness in the 3 secondary DOFs is high, thereby minimizing corresponding deflections, and
- (ii) operating well below the resonance frequency of the secondary DOFs (achieved by cantilever design).

Drive Mechanism

The primary motion is imparted to the model by an electromechanical drive mechanism. A constant magnetic field is established by rare-earth magnets positioned on either side of a narrow longitudinal opening in the sting. Alternating current at the frequency of oscillation is passed through an axially oriented beryllium copper conducting (drive) coil located within the crosswise flux provided by the magnets. The result is an alternat-

ing electromagnetically induced driving force. The conducting coil transfers the driving force to the balance by means of a "balance tongue" situated between the two suspension cantilevers, causing the balance and model to oscillate in the primary DOF.

Amplitude of oscillation is regulated by an electronic feedback loop. The primary amplitude output (from cantilever bridge) is fed back through the system and upon exceeding a pre-selected datum, supply current is reduced, and vice versa.

Carpenter 455 stainless steel was selected for the balance and cantilever beams. Both were properly heat treated after fabrication to avoid fatigue failure and hysteresis. The support sting was machined from low carbon cold rolled steel, which has a low resistance to magnetic flux, necessary for the drive mechanism.

4.2 DEVELOPMENT

Upon completion of the initial assembly (October 1984), a lengthy series of bench tests were undertaken, designed to diagnose potential problems, fine tune and evaluate the performance of the apparatus under controlled conditions prior to wind tunnel testing.

The remainder of this chapter deals with the relevant events during the development stages leading to finalization of the design of the test apparatus. A considerable amount of time was spent at each stage in troubleshooting and in the tedious dismantling and reassembly of the intricate apparatus.

4.2.1 ELECTRICAL CONNECTIONS

One of the most troublesome aspects encountered throughout the entire development stage was the electrical wiring connections, complicated by both the small size of the dynamic balance and the fragility of the strain-gauges.

In the early phases of development, it was found that the drive current required to oscillate the model was unacceptably high (≈ 50 amps). To accommodate the high current requirement, the cross-sectional area of the rear conductor springs (joining drive coil to the supply wire on the sting) was increased (thus also stiffening the springs). This resulted in even a higher current requirement. It was found that the conductor springs were actually restricting the movement of the drive coil. The problem was remedied by reducing both the cross-sectional area and the stiffness of the conductor springs through multi-leaved construction, resulting in a comfortable current level of only 15 amps.

4.2.2 MODEL MOUNTING AND BALANCING

As mentioned in the previous section, the balance outputs are resolved to give moments with respect to the apparatus RC. Accordingly, the model mounting requirements are such that the model moment RC (located at 35% of the wing mean aerodynamic chord of the SDM) coincides with the apparatus RC.

In an ideal situation, the aircraft model would perform absolutely "pure" translational oscillation with secondary reactions in pitch, yaw and roll produced by aerodynamic loads only. As previously mentioned, efforts have been made through balance design to limit deflections from the secondary DOFs. In reality, however, some cross-coupling exists and the balance experiences small amounts of structural distortion due to inertial loads imposed by the forced-oscillation, as well as externally applied aerodynamic loads. The inertial loads represent a "tare" which must be subtracted from balance outputs to arrive at a measure of the aerodynamic loads. Tare readings are determined with satisfactory accuracy in "still-air" under oscillatory conditions.

During primary oscillation tare outputs may be substantial, and in order to allow detection of the minute secondary aerodynamic signals, the inertial reactions must be minimized by balancing the model around the reference center. In other words, the model/balance system must be precisely balanced so that its CG is located at the apparatus RC. (Balancing the model also reduces oscillatory loads acting on the flexural pivots joining the cantilevers to the balance).

This was achieved through the following static measurement technique. The entire model/balance/sting assembly was mounted in such a manner that it could be easily inverted. The output from the pitching flexure strain-gauge bridge was monitored with the model and sting in the upright, and then the inverted position. The model was then ballasted by the addition of temporary weights, and repeated inversions carried out until identical readings were obtained from the pitching flexure for the model in both upright and inverted configurations. Findings of the investigation necessitated the following modifications to the system:

- (i) addition of a heavy metal insert to the back of the SDM fuselage;
- (ii) material removed from the front of the system;
 - (a) mounting spigot hollowed and balance trimmed, and

- (b) detachable nose cone section of the aluminium SDM replaced with a plastic nose cone.

Once statically balanced, any pitch output under oscillatory conditions indicates other phenomena at work; namely, still-air aerodynamic loads or sting bending, thoroughly investigated later in this report.

4.2.3 OPTIMIZATION OF SUSPENSION SYSTEM

The parallel cantilever suspension system provides a pure translational motion provided there is no angular deflection of the sting. Thus, the need to modify the existing suspension system is clarified by a brief consideration of the effects of sting deflections on measured stability derivatives.

The long, cylindrical sting required to minimize support interference on the test apparatus is subject to significant angular deflection and pitching oscillation caused by aerodynamic and inertial loads. Both the model and sting are subjected to a sudden application of aerodynamic load during the wind tunnel run and suffer large static deflections due to the magnitude of the load applied, particularly at high angles of attack. However, it is not the static deflections alone which cause concern during dynamic stability testing. Sting pitching oscillations occur under the inertial loads imposed by the forced-oscillation, and cause the model/balance system to pitch as well as translate along the z axis.¹ Thus, the measured stability derivatives are affected in two ways:

- (i) The model angular position (mean angle of attack, α_0) is not correctly measured due to static sting deflection. Corrections can be applied to determine the actual α_0 and are treated appropriately during data reduction in Chapter 5.
- (ii) The aerodynamic reactions from which the moment derivatives are obtained contain pitching effects; i.e., sting oscillation presents an additional, unwanted DOF of the model motion.

Figure 4.3 illustrates the geometry of the model/sting system deflected under inertial and aerodynamic loads. It is apparent from the figure that model angle of attack can be written as;

¹ The discussion is limited to translational oscillation along the z axis, although all comments apply to lateral translational oscillation experiments as well where sting yawing oscillations would cause the model/balance system to yaw as well as to translate laterally.

$$\alpha = \alpha_0 + \theta_d + \tan^{-1}[\dot{z} \cos \alpha_0 / V_\infty] \quad (4.1)$$

where

θ_d — represents sting pitching oscillation (a dynamic deflection),

$\tan^{-1}[\dot{z} \cos \alpha_0 / V_\infty]$ — is the induced angle of attack ($\Delta\alpha$) due to translational oscillation in plunge ($\dot{z} \cos \alpha_0$),

and α_0 is shown corrected for θ_s , the static sting deflection.

As the prime goal is to obtain a pure translational motion with respect to the aircraft body-fixed axes, the problem is to account for θ_d . Two possible courses of action exist:

- (i) Introduce a multi-DOF analysis where the equations of motion include sting oscillations superimposed on the model/balance oscillatory motion. The sting deflections can be measured by instrumenting the sting with strain-gauges, or can be calculated from the sting equations of motion using known static stability data acquired from a second independent experiment. Either way, the complexity and difficulty of obtaining accurate results is increased.
- (ii) Correct the actual model motion; i.e., modify the suspension system in such a way that θ_d can be compensated for.

It is preferable, of course, to correct the motion enabling utilization of a one DOF analysis rather than complicate data acquisition and reduction to accommodate additional DOFs.

An attempt was made to correct the model motion by slightly altering the cantilever installation. Steel shims were placed underneath the aft end of each cantilever, forcing the beams apart at the aft end. The two beams slightly converge toward the balance and move in such a way as to "artificially induce" a model/balance pitching angle which compensates for (is equal in magnitude and opposite in direction to) the sting pitching oscillation, θ_d . This angle, designated "artificially induced model/balance pitch angle", θ_b , corrects the model primary oscillation by removing unwanted pitching to regain pure translational oscillation along the z axis. The motion is depicted in Figure 4.4.

Fig. 4.3 Schematic illustrating model and sting motion

Fig. 4.4 Schematic illustrating suspension system motion: (a) without shim compensation, and (b) with shim compensation

Before implementation of the compensation scheme, a geometrical analysis of suspension system/sting movement was undertaken to investigate the behavior of θ_b (see Appendix A for details). The results of the mathematical model predicted the following:

- (i) the waveform of θ_b is essentially a pure sinusoid;
- (ii) a shim thickness of 0.38 mm is required to counteract θ_d ;
- (iii) the efficiency of the compensation scheme is reduced if a phase lag exists between the primary motion and sting oscillation.

As the geometrical model cannot exactly represent the suspension system/sting deflection, it served only as a guide to provide a "starting point" shim size for the compensation scheme. Necessarily, the above predictions had to be verified and any phase relationships determined. In consequence, an optical imaging system was designed to provide an illustration of the absolute motion of the aircraft model relative to an inertial datum by reflecting a light beam off the model surface during oscillation. Utilizing this system it was possible to:

- (i) ascertain the magnitude of model pitching caused by sting pitching oscillation, θ_d , and
- (ii) dynamically "fine tune" the test apparatus; i.e., optimize the suspension system by optimizing shim thickness.

Description of Optical Technique

A schematic of the optical arrangement is shown in Figure 4.5. The SDM was mounted in the wind tunnel at 0° angle of attack and 0° sideslip. A Helium-Neon laser² providing a coherent light source was reflected off a mirror positioned at the moment reference center of the model. The reflected beam travelled to a position sensing photodetector. During primary oscillation, model pitching motion was manifested as a vertical movement of the laser light spot on the sensor.

² A laser mounting system was specifically designed and fabricated for this investigation with four DOFs to allow for accurate laser beam alignment on the mirror surface. The author wishes to acknowledge this special contribution to this phase of the development by Mr. B.E. Moulton.

The photodetector is a continuous single axis position cell which senses linear movement of a light spot to a resolution of 0.01 mm. As the spot moves across the active area, output currents are generated which are proportional to the distance of the spot from the end contacts of the detector. Appropriate electronics were required to fully utilize the capability of the sensor and obtain position information representative of model pitching. From Figure 4.6;

$$2\theta_d = \tan^{-1} \left(\frac{y_s}{2d_b} \right) \quad (4.2)$$

where

y_s -- represents the vertical movement of light spot on sensor (sensor output), and

d_b -- represents the total distance travelled by the light beam from model to sensor (3.73 m).

By minimizing y_s through varying shim size, it was possible to minimize θ_d . The optical technique was implemented in the following manner:

- (i) Working with a statically balanced model/balance system a datum was measured without shims in position yielding $\theta_d = \pm 0.080^\circ \pm 1.5\%$ (agreeing with 0.078° measured under static loading) in the presence of a ± 2.5 mm primary oscillation. Also $\eta_s = -1.0^\circ$, (where η_s denotes phase shift between sting pitching and primary motion) calculated from the in-phase and quadrature components (real and imaginary components respectively) of the sensor output with respect to the primary motion. Since the shim correction addresses only the in-phase component of θ_d , having η_s close to zero ensured that the shim correction could be applied with almost 100% efficiency.
- (ii) The test apparatus was dynamically fine tuned through repeated measurements with slightly varied shim size (initial size as predicted by the geometrical analysis) until the sensor output was minimized (y_s in equation (4.2)). Measurements were performed under tare conditions.
- (iii) Findings were confirmed in a vacuum test to ensure that a correction was not being applied for any aerodynamically induced pitching moment in still-air measurements, however, no appreciable changes were observed. These particular tests posed some difficulty in measurement owing to the tunnel wall deflecting under vacuum.

Fig. 4.5 Optical arrangement for model pitching measurements

Fig. 4.6 Movement of laser beam with model pitching

Results

The results of the investigation are documented in the following:

- (i) Optimum shim thickness = 0.26 mm (for a 38 mm diameter sting, clamped 38.7 cm from the apparatus RC).
- (ii) Model pitching was minimized to θ_d less than $\pm 0.01^\circ$ at $\eta_s \simeq 10 \rightarrow 20^\circ$. The criterion for an acceptable value of θ_d would be θ_d less than 10% of the induced angle of attack, $\Delta\alpha$ (equation 4.1). This corresponds to θ_d less than $\pm 0.05^\circ$ based on a $\Delta\alpha$ of 0.5° (for $\alpha_0 = 0$) which is representative of the wind tunnel test conditions (to be shown in Chapter 6). The result obtained indicates that model pitching was less than 2% of the primary oscillation and hence the model motion was essentially pure translational oscillation under tare conditions.
- (iii) A somewhat surprising discovery was a small amount of horizontal movement of the light spot on the sensor, indicating rolling motion. With the sensor rotated 90° it was determined that frequency in roll corresponded to exactly 3 times the primary frequency. Because of this resonance condition, a finite oscillation in roll was induced which was measured to be less than $\pm 0.018^\circ$. The roll inertia of the balance or model could be changed to eliminate the motion by adding mass in the rolling plane, however, this proved to be unnecessary since the small magnitude of roll presented no immediate concern.

In conclusion, the optimization of the suspension system was successful and correction for sting pitching oscillation sufficient to proceed with a full calibration of the apparatus. Also, the optical imaging system proved to be very effective in determining the motion and no doubt will be used in future dynamic stability experiments.

4.2.4 STATIC CALIBRATION

Static calibration comprised the mounting of the apparatus in a calibration fixture outside the wind tunnel, loading the balance with known forces and moments which covered the range expected during wind tunnel testing, and recording strain-gauge readings at each force and moment combination. The calibration procedure served several purposes:

- (i) to determine calibration slopes (sensitivities) for the cantilever beam and each flexure;
- (ii) to check the linearity, hysteresis and degree of repeatability of load data;
- (iii) to determine deflection under load (for stiffness calculations), and

(iv) to determine balance interactions.

The results of the investigation (assuming translational oscillation along z axis) yielded the properties listed below:

Sensitivities

Cantilever beam	$4.11 \times 10^{-5} V/N \pm 0.4\%$
Pitch flexure	$4.51 \times 10^{-2} V/Nm \pm 0.5\%$
Forward yaw flexure	$3.25 \times 10^{-2} V/Nm \pm 0.7\%$
Aft yaw flexure	$3.49 \times 10^{-2} V/Nm \pm 0.6\%$
Roll flexure	$2.17 \times 10^{-2} V/Nm \pm 1.0\%$
Drive force	$1.02 \times 10^{-2} V/N$

Electrical centers (distance from apparatus reference center)

Pitch flexure	0.0 m
Forward yaw flexure	0.04755 m
Aft yaw flexure	0.01572 m

Apparatus stiffness

Primary DOF	79,080	N/m
Pitch DOF	1,580	Nm/rad
Yaw DOF	880	Nm/rad
Roll DOF	565	Nm/rad

Balance interactions (see Appendix B for definition)

(1) Interaction with normal force:

Forward yaw	1.10 %
Aft yaw	0.12 %
Roll	0.32 %

(2) Interaction with sideforce:

Pitch	0.18 %
Roll	0.90 %

(3) Interaction with rolling moment:

Forward yaw 1.0 %

Aft yaw 2.0 %

Strain-gauges were carefully positioned to minimize static interactions. For dynamic stability experiments, interactions of 1.0% are normally tolerated when determining stability derivatives and it can be seen that the percentages listed are within the acceptable limit (except for one). However, when determining static force and moment coefficients, interactions greater than approximately 0.3 % cannot be ignored and corrections must be applied to the appropriate data during the data reduction procedure.

All load data exhibited a high degree of linearity and repeatability. Details of the calibration procedure and results in graphical form are provided in Appendix B.

4.2.5 DYNAMIC CALIBRATION

In view of the complexity of the translational oscillation apparatus, the instrumentation and analytical procedures, and the unavailability of data on acceleration derivatives, an assesment of the reliability of wind tunnel results is very difficult. A 3 DOF dynamic calibrator was developed at the NAE several years ago (Ref. 35) which has proven to be invaluable as a means to independently verify the validity of the experimental techniques employed and identify any hardware or software faults which could otherwise easily go unnoticed.

Description and Operation of Calibrator

In a dynamic calibration the outputs from the dynamic stability apparatus are compared to a set of accurately known oscillatory loads which simulate the secondary aerodynamic pitch, yaw and roll moments. The purpose of the dynamic calibrator is to apply these loads to the apparatus while it performs a translational oscillatory motion. The applied moments must be:

- i) sinusoidal and of the same frequency as the primary motion;
- ii) independent of each other, enabling selection of amplitude and phase with respect to the primary motion; and
- iii) of the same order of magnitude as the amplitudes of oscillatory aerodynamic loads encountered in the wind tunnel experiment. These values were unknown at the time of calibration but 0.05 Nm and 0.1 Nm were considered adequate for the oscillatory mo-

ment. (From the wind tunnel experiment it is now known that the actual magnitudes encountered were up to 0.1 Nm).

Figure 4.7 shows the translational apparatus installed in the dynamic calibrator. The calibrator consists of two concentric octagonally shaped stationary structures that form a magnetic circuit necessary to establish a radially oriented magnetic field across four air gaps located on the horizontal and vertical axes of symmetry of the octagons (only two air gaps were needed in this case).

Since it is impractical to apply loads directly to the aircraft model, the calibration was performed with a floating calibration frame (shown aligned with the z axis) which is attached to a calibrating body mounted on the dynamic balance. The outer ends of the two arms of the calibrating frame support single-turn driving coils (oriented parallel to the y axis) located within the air gaps. By passing alternating currents through the driving coils, which thus interact with the surrounding magnetic field, the necessary moments were applied through the calibrating frame to the dynamic stability apparatus.

As the effectiveness of the calibrator depends on the accuracy of the applied moments, it is necessary to calibrate them in terms of the driving currents. This was accomplished by means of a static calibration performed prior to the start of this research (Ref. 35). The driving-current-to-moment transfer functions were determined by applying accurately known DC currents to the driving coils and measuring the static deflections, a valid procedure, since the relationship between the currents and generated loads is not a function of frequency.

The first attempt to oscillate the apparatus in the calibrator was unsuccessful and as a result some minor modifications were made to ensure future safe operation during calibration:

- (1) The initial configuration was such that the CG of the calibrating body was approximately 5 cm forward of the pitching flexure, placing additional loads on the flexural pivots (causing them to break) and producing a large pitching tare output. As discussed in Section 4.2.2 on model balancing, again the system must be precisely balanced so that the CG coincides with the pitching flexure location, achieved through the addition of a 780 gm mass to the calibrating body.
- (2) A stiffening rod was attached to the loading conductors, which experienced significant fluctuations during translational oscillation (not rotational oscillation for which they were initially designed).

Fig. 4.7 Translational oscillation apparatus installed in the dynamic calibrator

- (3) A semi-stiff nylon stop was installed with 3.0 mm clearance to prevent excessive amplitudes of oscillation during the initial setup.

Calibration Procedure

The verification procedure was analogous to an actual wind tunnel test except that the SDM was substituted by the calibrating frame and the oscillatory aerodynamic moments were simulated by electromagnetically induced ones. Both tare measurements (in the absence of applied moments) and wind-on measurements (in the presence of applied moments) were taken, and the data acquisition system and data reduction techniques (described in detail in Chapter 5) were fully utilized.

It is necessary to minimize secondary responses to the primary forcing input; i.e., responses in pitch, yaw or roll to forced plunging (vertical) translational oscillation. The response of a single DOF system of natural frequency ω_n to forcing at frequency ω can be expressed in terms of a magnification factor (Ref. 36);

$$\begin{aligned} \text{MF} &= \text{amplitude of motion / maximum static deflection} \\ &= \frac{1}{\left[\left(1 - \left(\frac{\omega}{\omega_n} \right)^2 \right)^2 + 4 \left(\xi \cdot \frac{\omega}{\omega_n} \right)^2 \right]^{\frac{1}{2}}} \end{aligned} \quad (4.3)$$

where

ξ — represents the damping factor.

From this it is clear that the natural frequencies of the secondary DOFs should be as far as possible from the frequency of the primary (forcing) oscillation. Since the calibrating frame did not exactly reproduce the inertial characteristics of the SDM, the frequency ratios, ω/ω_n , during the calibration procedure differed from those prevailing during the wind tunnel test. During calibration the ratios were 0.64, 0.71, and 0.84 for plunge/pitch, plunge/yaw and plunge/roll, respectively. These values are certainly not ideal and do not permit a "direct" dynamic calibration (in the sense of input-output transfer functions as in the case of the static calibration) but rather a verification of the approach used under a specific set of conditions present during the calibration. Note that the corresponding values for the wind tunnel test (using the SDM) were 0.54, 0.73 and 0.32, respectively, thus an even better agreement between input and measured moments can be expected for the wind tunnel test conditions.

Input moments in the pitch, yaw and roll DOFs were applied over a range of phase angles and at two amplitudes, and compared with outputs from the data reduction. The balance sensitivities used were the statically determined values multiplied by the appropriate

magnification factor. (Due to time restrictions and unavailability of required equipment, the direct force responses in the translational DOF were not dynamically calibrated).

Results

The results are displayed in the polar plots shown in Figures 4.8 to 4.10. The small differences between the applied and measured moments can be explained by the following reasons (Ref. 35):

- i) The accuracy with which the applied and measured moments are generated is approximately within 2 %.
- ii) The values of the resonance frequencies needed for the MF computations are associated with a complex mechanical system and cannot be assumed absolutely constant. Their accuracies were determined to be around 0.05 %.
- iii) Errors of electronic origin; i.e., gain and phase instabilities, DC drifts, etc., are established to be less than 3 %.

The measured moments and phase angles were within $\pm 2\%$ and $\pm 2^\circ$ respectively of their input counterparts. No interactions were detected on the forced DOFs outputs, a confirmation that the use of a tare measurement successfully eliminates such effects. It can be concluded that a complete verification of the system including instrumentation and data reduction was accomplished.

Fig. 4.8 Comparison of applied and measured dynamic pitching moments

Fig. 4.9 Comparison of applied and measured dynamic yawing moments

Fig. 4.10 Comparison of applied and measured dynamic rolling moments

Chapter 5

DATA ANALYSIS FOR DYNAMIC STABILITY TESTS

5.1 BASIC EQUATIONS

5.1.1 DIRECT DERIVATIVES

The information required to determine the direct static and dynamic derivatives is that associated with the primary motion — translational oscillation. As with previous chapters, the discussion is confined to oscillation along the z axis but is applicable to lateral oscillation as well. After successfully compensating for sting pitching oscillation, the model/balance motion has only one independent DOF. The motion of this one DOF system can be mathematically modeled and the equations analytically solved for the externally applied loads. In accordance with the assumptions stated in Section 2.1, the equation of motion (from equation (2.1c)) is;

$$\sum F_z = m\ddot{z} \quad (5.1)$$

with z designated as the vertical displacement of the model from its equilibrium position (along the body axis and positive downward). Also recall that for an aircraft model oscillating in a wind tunnel, the summation ($\sum F_z$) are the changes in the externally applied forces on the aircraft which can be categorized as follows:

- i) The mechanical reactions experienced by the model performing a translational oscillatory motion are those due to the mechanical damping γ and the mechanical stiffness k of the system, thus the *change in mechanical reaction* caused by model oscillation is given by;

$$Z_{mech} = -(\gamma\dot{z} + kz) \quad (5.2)$$

Both the damping and stiffness terms oppose the motion and displacement, respectively, as indicated by the negative sign.

- ii) The *change in aerodynamic reaction* resulting from model oscillation is represented in accordance with the method of Bryan introduced in Section 2.2; i.e.,

$$Z_{aero} = Z_z z + Z_{\dot{z}} \dot{z} + Z_{\ddot{z}} \ddot{z} \quad (5.3)$$

where Z_s , Z_d , Z_m are the aerodynamic stiffness, damping and "mass", respectively, by association with their mechanical counterparts. As there is ample clearance between oscillating model and tunnel wall, ground effects are not present and Z_s may be assumed to be zero.

iii) The system is also subjected to an externally applied sinusoidal driving force;

$$\bar{Z}(t) = |\bar{Z}| \sin(\omega t + \lambda) \quad (5.4)$$

where λ is referred to the primary oscillation.

Summing all of the above external forces and inserting them into equation (5.1), the equation of motion for the moving system becomes;

$$(m - Z_s)\ddot{z} + (\gamma - Z_d)\dot{z} + kz = |\bar{Z}| \sin(\omega t + \lambda) \quad (5.5)$$

For a sinusoidal excitation under steady state conditions, the primary motion is sinusoidal; i.e.,

$$z = |z| \sin \omega t \quad (5.6)$$

Therefore equation (5.5) becomes, in complex or vectorial form;

$$-\omega^2(m - Z_s)|z|e^{i\omega t} + i\omega(\gamma - Z_d)|z|e^{i\omega t} + k|z|e^{i\omega t} = |\bar{Z}|e^{i(\omega t + \lambda)} \quad (5.7)$$

Dividing through by $e^{i\omega t}$ equation (5.7) becomes for tare and wind-on conditions, respectively (referred to by subscripts T and W);

$$-\omega_T^2 m |z_T| + i\omega_T \gamma |z_T| + k |z_T| = |\bar{Z}_T| e^{i\lambda_T} \quad (5.8)$$

$$-\omega_W^2 (m - \bar{Z}_s) |z_W| + i\omega_W (\gamma - \bar{Z}_d) |z_W| + k |z_W| = |\bar{Z}_W| e^{i\lambda_W} \quad (5.9)$$

Examining the real parts of equations (5.8) and (5.9) gives;

$$-\omega_T^2 m |z_T| + k |z_T| = |\bar{Z}_T| \cos \lambda_T \quad (5.10)$$

$$-\omega_W^2 (m - \bar{Z}_s) |z_W| + k |z_W| = |\bar{Z}_W| \cos \lambda_W \quad (5.11)$$

Solving for k in equation (5.10) and substituting into (5.11) results in;

$$Z_s = \left[\frac{|\bar{Z}_W| \cos \lambda_W}{|z_W|} - \frac{|\bar{Z}_T| \cos \lambda_T}{|z_T|} + m(\omega_W^2 - \omega_T^2) \right] \frac{1}{\omega_W^2} \quad (5.12)$$

Similarly, by taking the imaginary parts of equations (5.8) and (5.9); namely,

$$\omega_T \gamma |z_T| = |\bar{Z}_T| \sin \lambda_T \quad (5.13)$$

$$\omega_W (\gamma - Z_i) |z_W| = |\bar{Z}_W| \sin \lambda_W \quad (5.14)$$

Solving for γ in equation (5.13) and substituting into (5.14) yields;

$$Z_i = - \left[\frac{|\bar{Z}_W| \sin \lambda_W}{\omega_W |z_W|} - \frac{|\bar{Z}_T| \sin \lambda_T}{\omega_T |z_T|} \right] \quad (5.15)$$

Expressions (5.12) and (5.15) define the desired aerodynamic "mass" and damping in terms of:

- i) the primary oscillation amplitude and frequency;
- ii) the in-phase and quadrature components of the driving force with respect to the primary motion.

Thus, these are the quantities required in the data acquisition for both tare and wind-on conditions and are readily measured using the instrumentation system described in Section 2 of this Chapter.

The derivatives Z_i and Z_i' are reduced to nondimensional form using the definitions included in the List of Symbols (note: * denotes multiplication by $\cos \alpha_0$; i.e., $C_{Z_i}^* = C_{Z_i} \cos \alpha_0$):

$$C_{Z_i} = C_{Z_i}^* = \frac{2V_\infty^2}{q_\infty S \bar{c}} Z_i = \frac{2V_\infty^2}{q_\infty S \bar{c}} \left[\frac{|\bar{Z}_W| \cos \lambda_W}{|z_W|} - \frac{|\bar{Z}_T| \cos \lambda_T}{|z_T|} + m(\omega_W^2 - \omega_T^2) \right] \frac{1}{\omega_W^2} \quad (5.16)$$

$$C_{Z_i} = C_{Z_i}^* = \frac{-V_\infty}{q_\infty S} Z_i = \frac{-V_\infty}{q_\infty S} \left[\frac{|\bar{Z}_W| \sin \lambda_W}{\omega_W |z_W|} - \frac{|\bar{Z}_T| \sin \lambda_T}{\omega_T |z_T|} \right] \quad (5.17)$$

The above analysis is applicable to primary oscillation in the lateral DOF as well, resulting in the direct derivatives C_{Y_β} , C_{Y_β}' of similar formulation.

5.1.2 CROSS AND CROSS-COUPLING DERIVATIVES

To determine the cross and cross-coupling moment derivatives it is necessary to consider the change in aerodynamic reactions in the secondary DOFs (roll, pitch, yaw) caused by oscillation in the primary translational DOF. First, recall that after correcting for sting pitching oscillation, the primary oscillation is pure translational, of the form;

$$z = |z| e^{i\omega t} \quad (5.18)$$

It is sufficient for illustrative purposes to consider, as an example, the aerodynamic pitching moment induced by the pure translational oscillation, represented below using linear superposition (as per Section 2.2):

$$M_{aero} = M_z \dot{z} + M_{\ddot{z}} \ddot{z} \quad (5.19)$$

For the small amplitude of the sinusoidal primary oscillation of the model ($\Delta\alpha$ less than 1°), it can be assumed that, for a given model attitude, the system behaves linearly about its equilibrium position. In this case, the induced secondary aerodynamic reactions are also sinusoidal and of the same frequency as that of the primary motion; i.e., they are *synchronous* with the primary motion. Therefore, the oscillatory pitching moment caused by a primary translational oscillation is of the form;

$$M_{aero} = |M| e^{i(\omega t + \eta_\theta)} \quad (5.20)$$

where η denotes phase shift with respect to the displacement in the primary DOF. Substituting equations (5.18) and (5.20) into (5.19);

$$M_z |z| i\omega e^{i\omega t} - M_{\ddot{z}} |z| \omega^2 e^{i\omega t} = |M| e^{i(\omega t + \eta_\theta)} \quad (5.21)$$

Taking the real and imaginary parts of equation (5.21) and dividing through by $e^{i\omega t}$ yields;

$$-M_z |z| \omega^2 = |M| \cos \eta_\theta \quad (5.22)$$

$$M_z |z| \omega = |M| \sin \eta_\theta \quad (5.23)$$

Thus equations (5.22) and (5.23) become;

$$M_{\ddot{z}} = -\frac{|M| \cos \eta_\theta}{\omega^2 |z|} \quad (5.24)$$

$$M_z = \frac{|M| \sin \eta_\theta}{\omega |z|} \quad (5.25)$$

The derivatives $M_{\ddot{z}}$ and M_z are reduced to nondimensional form using the definitions included in the List of Symbols:

$$C_{m_{\ddot{z}}} = C_{m_{\ddot{a}}} = -\frac{2V_\infty^2}{q_\infty S \bar{c}^2} \cdot \frac{|M| \cos \eta_\theta}{\omega^2 |z|} \quad (5.26)$$

$$C_{m_z} = C_{m_a} = \frac{V_\infty}{q_\infty S \bar{c}} \cdot \frac{|M| \sin \eta_\theta}{\omega |z|} \quad (5.27)$$

Expressions (5.26) and (5.27) define the desired dynamic and static "cross" derivatives respectively. Similar expressions can be derived for the L and N moment derivatives due to

translational oscillation ("cross-coupling" derivatives) by including cross-coupling terms between longitudinal and lateral DOFs in the aerodynamic moment expansions. The resulting expressions are;

$$C_{l_{\dot{\alpha}}} = -\frac{2V_{\infty}^2}{q_{\infty}Sb\bar{c}} \cdot \frac{|L|\cos\eta_{\phi}}{\omega^2|z|}, \quad C_{l_{\alpha}} = \frac{V_{\infty}}{q_{\infty}Sb} \cdot \frac{|L|\sin\eta_{\phi}}{\omega|z|} \quad (5.28)$$

$$C_{n_{\dot{\alpha}}} = -\frac{2V_{\infty}^2}{q_{\infty}Sb\bar{c}} \cdot \frac{|N|\cos\eta_{\psi}}{\omega^2|z|}, \quad C_{n_{\alpha}} = \frac{V_{\infty}}{q_{\infty}Sb} \cdot \frac{|N|\sin\eta_{\psi}}{\omega|z|} \quad (5.29)$$

Equations (5.26) to (5.29) indicate that the quantities required in the data acquisition for each derivative are:

- i) the amplitude and frequency of the primary oscillation;
- ii) the amplitude and phase of the *induced* secondary moment (aerodynamic load) with respect to the primary motion.

The procedure for obtaining these aerodynamic loads in terms of the measured strain-gauge outputs originating from the balance flexures (roll, pitch, yaw) is not as straightforward as that outlined in the previous section for the determination of direct derivatives. This technique is implemented in the data reduction summarized in Section 5.4 to follow.

The reader is reminded that all stability derivatives are local derivatives, that is, derivatives applicable to the particular equilibrium position (α_0 , β_0) of the aircraft model.

Again, the foregoing analysis is applicable to primary oscillation in the lateral DOF resulting in the cross and cross-coupling dynamic derivatives $C_{n_{\beta}}$, $C_{l_{\beta}}$, $C_{m_{\beta}}$ and static derivatives $C_{n_{\beta}}$, $C_{l_{\beta}}$, $C_{m_{\beta}}$.

5.2 DATA ACQUISITION

With the current data acquisition system routinely used in forced-oscillation experiments at the UAL, the desired quantities in expressions (5.16), (5.17), (5.26) to (5.29) can be acquired, enabling calculation of the particular stability derivatives. A brief description of the hardware is presented herein.

Figure 5.1 shows a block diagram of the instrumentation system. As described in Section 4.1.3, strain-gauge bridges are mounted on the top cantilever and on the balance flexures, where the voltage difference measured across each bridge is proportional to the cantilever deflection or the applied load at a particular flexure. The system accepts, simultaneously through six channels, the corresponding voltages originating from

the strain-gauge bridges; the primary signal from the cantilever bridge and secondary signals from the pitch, yaw and roll flexures. An additional channel is a driving coil current monitor representing the driving forcing function. The resulting unfiltered analog signals from the six channels travel from the wind tunnel to a computer room where they are monitored on a multi-channel oscilloscope and the levels are increased in amplitude to attain the desired signal magnitude.

Typical balance outputs from the oscillating model are composed of an oscillatory component (AC) superimposed on a non-oscillatory component or offset (DC) due to the loads experienced under wind-on conditions. The output signals pass through a set of filters (one for each channel) which separate the AC and DC components. The DC components are recorded by a strip chart recorder and are used to calculate the static force and moment coefficients. The AC components pass through a separate set of amplifiers to bring the signal up into the volts range and are transmitted to an analog-to-digital (A/D) converter whose conversion timing is controlled by a pacing control unit. Finally, the numerical output is fed to a computer for signal processing and data reduction.

To accurately measure the frequency of the primary oscillation, the primary signal is also sent to a digital frequency counter where the number of cycles in the signal is counted during an accurately known time interval. This is also fed to the computer.

5.3 SIGNAL PROCESSING

The numerical data are now processed in an HP1000F computer by the signal processing software developed at the UAL and illustrated in flow chart form in Figure 5.2. The process is controlled by a "main program" which calls on various "sub-programs". Initially, sub-program *REDAT* acquires data from the A/D converter and stores it in memory, recording signals for subsequent analysis. During each wind tunnel run of approximately 10 seconds duration, each channel is sampled at a fixed rate of 250 samples/second, satisfying Nyquist's criterion. Up to 50,000 samples/run may be taken, but to avoid memory overflow problems the samples/run are stored in 4 "records".

In Section 5.1.2 of this Chapter, it was established that the interest is in extracting only the secondary signal components synchronous with the primary motion. However, these balance outputs are heavily masked by aerodynamic noise created by the unsteady flow environment during wind tunnel testing. It is important to minimize aerodynamic noise effects in order to characterize the signals with the greatest possible precision, and as

the noise can be of the order of approximately one thousand times larger than the desired aerodynamic signal, conventional band pass filtering methods are not sufficient.

Instead, the knowledge that the secondary signals are synchronous with the primary motion allows the use of a sophisticated filtering technique— "cross-correlation technique"— based on computing the cross-correlation function between the primary and secondary signals (Ref. 37). After the completion of the wind tunnel run, the records are retrieved one at a time from memory and the cross-correlation functions are computed for the record under consideration (sub-program *CORDT*). The process is repeated for subsequent records (until the 4 records/run are correlated) and a running average of the respective functions is continually maintained. Since it is known that the correlations are sinusoidal and of the same frequency as the primary motion, sub-program *SINFIT* implements a standard sinusoidal curve fitting procedure to the ensemble average of the cross-correlation functions. So basically, the end result of *CORDT* is that the secondary waveform of interest for each channel is extracted by:

- i) separating the synchronous components of the signal from the noise;
- ii) establishing its amplitude and phase with respect to the known primary motion reference signal.

At this point it must be realized that the various instrumentation components in the system introduce errors in the amplitude and phase of the secondary signals for each channel, and since these must be accurately determined for data reduction, an electrical calibration of the system is necessary. During calibration, which is performed typically twice a day before wind tunnel runs, a computer controlled signal generator (see Fig. 5.1) feeds sinusoids of known frequency and amplitude into each of the six channels. Basically, the computer reads these calibration signals (inputs), utilizes the cross-correlation software to determine the amplitude and phase of each channel with respect to the primary (outputs), and thus obtains an electrical transfer function for each channel in vectorial form. With a known transfer function, the information from *CORDT* is now corrected for distortion and time delays introduced by the system, thus yielding the "true" form of the strain-gauge signals for data reduction (output (*CORDT*)/transfer function - input).

Fig. 5.1 Block diagram of instrumentation system

Fig. 5.2 Flow chart of signal processing software

5.4 DATA REDUCTION

5.4.1 CALCULATION SEQUENCE FOR AERODYNAMIC DERIVATIVES

Voltages from *CORDT* are now fed to subroutine *AERO* in which the data is reduced to obtain the desired aerodynamic derivatives. The aforementioned software is standard signal processing for any forced-oscillation experiment at the UAL, however, *AERO* must be rewritten to accommodate the data reduction equations for the particular experiment. A brief outline of the reduction procedure for this translational experiment is provided below, accompanied by the flow chart of Figure 5.3. The calculation sequence for the data reduction of a single test run is as follows:

- i) conversion of the aerodynamic angles from aerodynamic axes to body axes:

Figure 5.4 shows the various axes systems used in the analysis. The relationship between the aerodynamic angles in the aerodynamic axes (x' , y' , z') and body axes (x , y , z) systems is given by (Ref. 38);

$$\begin{aligned}\sin \beta &= \sin \sigma \sin \phi' \\ \tan \alpha &= \tan \sigma \cos \phi'\end{aligned}\tag{5.30}$$

- ii) calculation of flow parameters;
- iii) transformation of voltages into absolute loads:

The voltages from channel one, in conjunction with the standard sensitivity (sn) in terms of voltage/load and stiffness (k) in terms of load/deflection are transformed into the amplitude of primary oscillation for both wind-on and tare conditions (assuming z axis translation):

$$\frac{V}{k \cdot sn} \rightarrow \text{amplitude}(z)\tag{5.31}$$

Also, the additional voltages from channels two through six are converted into the driving force ($\bar{Z}$) and the respective secondary moments ($\bar{M}$, $\bar{N}$, $\bar{L}$) for both wind-on and tare conditions:

$$\frac{V}{sn} \rightarrow \text{force or moment}\tag{5.32}$$

It should be noted that the data are presented in the "in-phase" and "quadrature" format; i.e., if the amplitude and phase of the measured force or moment vector ($a_1 + ia_2$) is defined as;

$$\text{Amplitude} = \sqrt{a_1^2 + a_2^2}$$

$$\text{Phase} = \tan^{-1} \frac{a_2}{a_1}$$

then a_1 = in-phase component and a_2 = quadrature component.

iv) computation of aerodynamic loads:

The in-phase and quadrature components of the driving force and primary oscillation amplitude for both tare and wind-on conditions are now easily substituted into expressions (5.16, 5.17) to obtain direct derivatives. However, for cross and cross-coupling derivatives, solely the *aerodynamic moment due to the forcing motion* is required (equations (5.26 to 5.29)) and the reduction procedure is slightly different (As with the direct derivatives the data are represented by both tare and wind-on vectors). The following assumption must be realized:

The very high stiffness in the roll, pitch and yaw planes of motion results in amplitudes of the secondary motions that are small relative to the forcing amplitude (from Section 4.1.3), thus there are negligible aerodynamic and mechanical interactions among the secondary DOFs. It is safely assumed that the systems acts independently in the roll, pitch and yaw planes with an independent equation describing the motion for each plane.

With this in mind it can be shown (Ref. 37) that for a one DOF system;

$$\Delta G_{aero}^{in} = \text{MOMENT}(\text{wind}) - \left[\text{MOMENT}(\text{tare}) \times \frac{\text{AMPLITUDE}(\text{wind})}{\text{AMPLITUDE}(\text{tare})} \right] \quad (5.33)$$

meaning that subtraction of the tare vector from the wind-on vector yields the unknown aerodynamic moment in vectorial form or "aerodynamic vector", (ΔG_{aero}^{in}) in the angular DOF under consideration.

v) resolution of yaw/pitch moments about RC:

It should be noted that the moments measured at the "forward" and "aft" yaw flexures (pitch flexures in the case of a lateral oscillation experiment) must be converted to a single moment or "aerodynamic" vector about the apparatus reference center. The equations which convert the data are well commented in the computer program.

vi) calculation of nondimensional aerodynamic derivatives:

Knowledge of the magnitude and phase of the "aerodynamic vectors" completes the calculation sequence and these quantities are substituted into expressions (5.26) to (5.29) yielding the required cross and cross-coupling moment derivatives.

Fig. 5.3 Data reduction flow chart

Fig. 5.4 Reference systems. (Ref. 38)

vii) output of tabulated results:

The entire data are stored on disc in a format acceptable for further calculation and the results printed on a line printer in the wind tunnel control room.

For the appropriate software and details of the abovementioned calculations, the reader is referred to Appendix C.

5.4.2 POST TESTING DATA PROCESSING

An additional computer program was developed for data handling at the end of the test series or whenever inspection was required.

The raw derivative data for each run (from *AERO*) are read from disc file. The test data are sorted into groups of runs at common test conditions (i.e. α , β , M_∞ , $\Delta\alpha$) which have been labeled with an "identification number". Running averages of the derivative data are computed for each of these groups. (Data are averaged after the test series as opposed to on-line to first allow any errant runs to be removed, thus optimizing the usable data).

The corresponding static aerodynamic coefficients for each identification number are also calculated from chart recorder traces of the DC components of the balance outputs. These yield data at the dynamic test conditions for which the corrections for model/balance/sting static deflections could be determined.

All static and dynamic derivative data are stored in plot files enabling access to a separate plotting program to display the data whenever required. Refer to Appendix C for software and calculation details.

Chapter 6

DYNAMIC STABILITY TESTS

A wind tunnel verification test of the translational oscillation apparatus was conducted with three main objectives:

- (i) to verify the operating capability of the apparatus at high angles of attack;
- (ii) to measure cross and cross-coupling derivatives for the SDM during translational acceleration along the z and y body axes, and
- (iii) to become familiar with the details associated with this new technique.

6.1 TEST FACILITIES

6.1.1 TEST MODEL

The Standard Dynamics Model is representative of a current fighter type aircraft and was originally designed at the NAE in October of 1978. The model was later built to different scales at several wind tunnel institutions in an effort to establish the validity of new dynamic stability test techniques using a common configuration.

The significant dimensions of the NAE SDM are presented in Figure 6.1 and a detailed description given in Reference 39. As mentioned in Chapter 4, results are referred to the 35% mean aerodynamic chord reference center. All external components such as wings, stabilators, canopy etc. may be removed for bench tests if required. SDM's manufactured from aluminium and brass are available but only the aluminium model was used for this experiment.

To date, the NAE has obtained direct, cross and cross-coupling derivatives from oscillation-in-roll (Refs. 39, 40), in-pitch and in-yaw (Ref. 38) experiments with the SDM at Mach 0.6, for angles of attack up to 40° and angles of sideslip 0° and 5° . The AEDC in Tennessee, FFA in Sweden and the DFVLR in the Federal Republic of Germany have performed similar experiments with the SDM at several Mach numbers, but for maximum angles of attack between 20° and 29° and zero sideslip (Refs. 41, 42, 43, respectively).

Fig. 6.1 Geometrical details of the Standard Dynamics Model. (Ref. 38)

Thus, a reasonably complete set of data exists for the SDM with the exception of translational acceleration derivatives.

6.1.2 WIND TUNNEL AND SUPPORT SYSTEM

The tests were performed in the NAE $0.75\text{ m} \times 0.40\text{ m}$ Dynamics Wind Tunnel. This is an intermittent suction facility capable of being operated at Mach numbers from 0.2 to 2.0. The subsonic Mach number range from 0.2 to 0.5 (required in the current test program) can be achieved with a double wedge placed in the variable diffuser throat. Stagnation pressures are normally slightly less than atmospheric pressure due to inlet losses, and stagnation temperatures are close to ambient air temperature. With the subsonic wall liner on one side of the test section, its dimensions are $0.75\text{ m} \times 0.38\text{ m}$.

The dynamic stability test apparatus is supported in the tunnel by a sidewall-mounted support system shown in Figure 6.2. The sting is clamped within a split, A-frame base, keyed and bolted to a plate flush with the tunnel sidewall. The side support operation is such that model angle of attack and sideslip (α , β in body axes system) are obtained through adjustments of the model pitch and roll angles (σ , ϕ' in aerodynamic axes system) (equation (5.30)). There are manually adjustable mechanisms for setting the model pitch and roll angles σ , ϕ' , with an accuracy of $\pm 0.1^\circ$ while maintaining the model location at the center of the test section. In addition, provision is made for small yaw adjustments $\Delta\psi$ required for flow alignment purposes (Section 6.2.1). The reader is directed to Reference 38 for details of the adjustment mechanisms.

6.2 TEST PROGRAM

6.2.1 FLOW ALIGNMENT

Small deviations in the tunnel flow directions were expected in the relatively small test section of the Dynamics Wind Tunnel due to the presence of the sidewall-mounted support. In Figure 6.3, the flow or wind axes system (X , Y , Z) is shown inclined through angles $\Delta\psi_0$ and $\Delta\theta_0$ with respect to the tunnel fixed system (X_T , Y_T , Z_T). The purpose of the flow angularity investigation was to align the model body axes (x , y , z) with the wind X , Y , Z system.

The approach used was the same as that published for past experiments (Refs. 38, 39). Initially, the model orientations σ , ϕ' (see Fig. 6.3) were set to zero with respect to X_T , Y_T , Z_T . The SDM itself was used as a flow angle probe while static data were recorded in a series of runs with the model in both upright and inverted (180°) positions.

Fig. 6.2 The SDM in position on the sidewall support system. (Ref. 38)
(The apparatus shown is not the translational oscillation apparatus.)

Fig. 6.3 Aircraft angles and setup geometry. (Ref. 38)

From the normal force outputs and C_{Z_α} (obtained from SDM static aerodynamic data in Ref. 39), the effective flow angularity $\Delta\alpha$ was calculated and eliminated through a number of fixed sting adjustments of $\Delta\theta$ made relative to the Z_T axis (Z_T not shown in Fig. 6.3). Similarly, from sideforce outputs and C_{Y_β} (Ref. 39), the effective flow angularity $\Delta\beta$ was calculated and eliminated through fixed sting adjustments of $\Delta\psi$ made relative to the horizontal X_T axis.

The criterion for acceptable flow alignment is 0.1° or less flow angularity based on previous SDM experiments at NAE which evaluated the effects of flow curvature on dynamic stability data (Ref. 38). The final adjustments in the present investigation resulted in $\Delta\alpha$ and $\Delta\beta$ of approximately $+0.1^\circ$ and -0.05° respectively, yielding normal force and sideforce less than 1 N and 0.1 N respectively, so that no corrections for flow alignment were required in the data reduction.

6.2.2 TEST CONDITIONS

A summary of the test conditions for z axis translational oscillation experiments¹ is presented in Table 6.1. Data were taken at a nominal Mach number of 0.25 and Reynolds number per meter of 5.3×10^6 , with tunnel stagnation pressures ranging from 94–99 kPa. The apparatus performed translational oscillation at a nominal amplitude of $\pm 2.5\text{ mm}$ and nominal wind-on frequency of 45.5 Hz (for the aluminium SDM). The angle of attack ranged from 0° to 42.5° at increments of 2.5° under symmetrical conditions ($\beta = 0^\circ$). Proposed non-zero sideslip tests ($\beta = 5^\circ$) are to be conducted at 5° angle of attack increments.

The above-mentioned test conditions were chosen after consideration of the project objectives and careful examination of preliminary data from a number of preparatory tests yielding the following information:

- (i) For a translational oscillation of the form;

$$z = |z|e^{i\omega t} \quad (6.1)$$

the maximum translational velocity is;

$$|\dot{z}| = \omega |z| \quad (6.2)$$

¹ Due to time restrictions, a full set of lateral oscillation experiments were not conducted at the time of writing this thesis.

and the maximum induced angle of attack is;

$$|\Delta\alpha| = \frac{|\dot{z}|}{V} = \omega \frac{|z|}{V} \quad (6.3)$$

The aerodynamic angle, $\Delta\alpha$, induced by the translational oscillation must be sufficient to cause aerodynamic effects large enough to be measurable. From oscillation in pitch experiments (Ref. 38), where $\Delta\alpha$ was decreased from 1.2° to approximately 0.2° , it was found that the threshold below which no measurable effects occurred was $\Delta\alpha \approx 0.5^\circ$. Although these tests were conducted in a different DOF, it was reasonable to assume that a minimum $\Delta\alpha \approx 0.5^\circ$ would be required for translational oscillation as well. From equation (6.3), for a given frequency of oscillation, an acceptable value of $\Delta\alpha$ could be attained by oscillating at the maximum apparatus design amplitude ($|z| = 2.5 \text{ mm}$) and reducing Mach number from 0.7 to 0.25.² This was verified in the preparatory tests.

- (ii) The preparatory tests included a preliminary inspection of the effect of flow interference between the model and side support on the data. The full investigation is to be reported by M.E. Beyers in a future NAE publication.
- (iii) Considerable effort was made to verify pure translational motion under wind-on conditions (for maximum sting length of 23.8 cm). This was checked at $\alpha = 0^\circ, 10^\circ$ and 30° utilizing the previously described optical measurement technique.
- (iv) To achieve an acceptable level of accuracy, sampling rates of 4 and 5 wind tunnel runs per test data point were chosen for $\alpha < 32.5^\circ$ and $\alpha > 32.5^\circ$ respectively, based on observation of data scatter. Fewer runs would have been sufficient at the lower angles of attack as the scatter was minimal.

A typical wind tunnel run consisted of three parts; an electrical calibration, a tare run with model oscillating but wind-off and finally, wind-on. Data were sampled for approximately 7 seconds.

². In any event, on the basis of the data obtained, it now appears that experiments at a higher Mach number (maybe even as high as 0.7), would still produce measurable data.

2 OF 2

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS
STANDARD REFERENCE MATERIAL 1010a
(ANSI and ISO TEST CHART No. 2)

Table 6.1

Test Conditions Summary

$$M_{\infty} = 0.25, Re = 5.3 \times 10^6, f = 45.5 \text{ Hz}, \Delta z = \pm 2.5 \text{ mm}$$

$\alpha \text{ (deg)}$	$\beta \text{ (deg)}$	$\Delta\alpha \text{ (deg)}^+$	$\bar{\omega}$
0, 2.5, (2.5)... 42.5	0	± 0.480	0.144
0, 5.0, (5.0)... 40	5	± 0.480	0.144
0, 5, 15, 30	-5	± 0.480	0.144

⁺ Refers to maximum $\Delta\alpha$

6.3 RESULTS

The results of the wind tunnel tests are presented in Figures 6.4 to 6.10, which constitute a complete set of stability derivatives due to translational oscillation along the z body axis at zero sideslip. The angles of attack (corrected for static deflections) are plotted on the abscissa and the measured aerodynamic coefficients and stability derivatives are plotted on the ordinate. A cubic spline curve fit was applied to the data points to obtain a smooth curve. The data are based on a total of 86 wind tunnel runs.

6.3.1 STATIC AERODYNAMIC COEFFICIENTS

The static moment and force coefficients, measured during the acquisition of dynamic data, are displayed in Figure 6.4. There were no important flow asymmetries with respect to the vertical plane at zero sideslip, reflected by the very small magnitude of the lateral coefficients C_n , C_l , C_y over the entire range of angle of attack tested. The behavior of the longitudinal coefficients C_m and C_z is as expected; an increase in C_z with α , and the variation of C_m with α due to the increase in magnitude of the normal force and shift in its line of action with respect to the RC.

Previous C_m values for the SDM obtained from pitch oscillation experiments ($\beta = 0^\circ$, $M_\infty = 0.6$, $Re = 10.4 \times 10^6$) with the NAE electro-mechanical pitch/yaw apparatus on the sidewall support (Ref. 38) are also presented in Figure 6.4 for comparative purposes. There is a definite correspondence, with minor differences at moderate angles of attack (20° – 30°) most likely due to a combination of the lower Mach number and Reynolds number for translational oscillation tests. The lateral coefficients also compare well to the previous static data (not shown). These findings reinforce the confidence in the static coefficients obtained as a by-product of the dynamic measurements.

To examine the static longitudinal characteristics of the SDM during translational oscillation, $\partial C_m / \partial \alpha$ was calculated from the pitching moment coefficient. These points are plotted with respect to α in Figure 6.5. The values of the static stability parameter, C_{m_α} , indicate that the configuration tested under the given conditions was statically stable for $\alpha < 4^\circ$ and $13^\circ < \alpha < 36^\circ$. The values of C_m (and thus C_{m_α}) depend, of course, on the RC location chosen. The static stability data obtained for the SDM (RC = 35% $\bar{c}$) could be applied to an aircraft with a negative static margin such as some "Controlled Configuration Vehicle" aircraft of today.

6.3.2 AERODYNAMIC DERIVATIVES

The static and dynamic force and moment derivatives due to translational oscillation along the z axis at $0^\circ \beta$ and $\bar{\omega} = 0.144$ are shown in Figures 6.6 to 6.10.

Even though emphasis in this experiment is on the moment derivatives, the direct (force) derivatives $C_{Z\dot{\alpha}}^*$, $C_{Z\alpha}^*$ are shown for completeness (Fig. 6.6). The reader is reminded that drive force measurements were not dynamically calibrated and therefore these results provide information only on the trend of the direct derivatives whereas the actual magnitudes may be off by a small factor. Nonlinear trends with angle of attack are evident for both the static and dynamic direct derivatives. Note that $C_{Z\dot{\alpha}}^*$ may change from approximately +10 to -10 with a change in α of only 2° . Its static counterpart, equivalent to damping, $-C_{Z\alpha}^*$, is positive over the range of angle of attack tested.

Figures 6.7 to 6.9 present the three moment derivatives due to translational oscillation, each dynamic derivative plotted with its static counterpart. The change in pitching moment due to translational oscillation; i.e., the cross derivatives $C_{m\dot{\alpha}}^*$, $C_{m\alpha}^*$ are shown in Figure 6.7. The same principal trends of the direct derivatives may be observed here. Nonlinear variations with α are displayed as α increases above $8-10^\circ$. $C_{m\dot{\alpha}}^*$ initially drops to a value of -4.0 at $\alpha = 10^\circ$ and then experiences local fluctuations between an α of 15° and 20° . Beyond this point a steady drop occurs to a value of -9.0 at $\alpha = 32^\circ$. $C_{m\alpha}^*$ behaves in a similar manner with its positive and negative peaks at the same angles of attack as in the dynamic case, but of opposite sign.

The aerodynamic cross-coupling derivatives; i.e., the aerodynamic rolling and yawing moments attributable to translational oscillation are shown in Figures 6.8 and 6.9. These out-of-plane derivatives maintain much smaller values over all angles of attack as compared to the in-plane derivatives, as to be expected at zero sideslip. The curves illustrate the suddenness of the variations with α , both $C_{l\dot{\alpha}}^*$ and $C_{n\dot{\alpha}}^*$ becoming highly nonlinear as α increases above 5° and 15° respectively. $C_{l\dot{\alpha}}^*$ develops a maximum negative value at $\alpha = 17^\circ$ and a maximum positive value at 32° followed by a rapid drop. This pattern is also reflected in $C_{n\dot{\alpha}}^*$, however, the trends are smoother and the peaks at $\alpha = 17^\circ$ and 30° are much less pronounced. The mirror like symmetry of dynamic and static curves is again apparent, but not as well defined as with the in-plane derivatives.

The three dynamic moment derivatives $C_{m\dot{\alpha}}^*$, $C_{n\dot{\alpha}}^*$, $C_{l\dot{\alpha}}^*$ are presented together in Figure 6.10. The very small magnitudes of the cross-coupling derivatives at zero sideslip in comparison with $C_{m\dot{\alpha}}^*$ are clearly illustrated. The use of forebody strakes (or leading

edge extensions) on configurations such as the SDM is a common method of ensuring a symmetrical shedding of forebody vortices at zero sideslip, thus reducing the magnitude of cross-coupling derivatives. However, it is expected that future results from the proposed test program (Table 6.1) with $\beta = 5^\circ$ (asymmetry introduced) may show much larger variations in the cross-coupling derivatives at high angles of attack, as observed in previous pitching, yawing and rolling experiments with the SDM (Refs. 38, 39 respectively).

The overall uncertainty in the results is a combination of those from the dynamic balance and data measurement system, the tunnel measured quantities, and wind tunnel and support interference. Absolute errors in the balance and measurement system were determined to be minimal from the static and dynamic calibration (Chapter 4). Thus the precision of the derivative measurements is influenced more heavily by errors in the measurement of flow parameters (pressure and temperature) and the wind-on dynamic moments. The scatter in the data between different wind tunnels runs at the same test conditions is a good indication of this uncertainty; found to be minimal ($\pm 2\%$) for direct and cross derivatives at all angles of attack and considerably larger for cross-coupling derivatives at $\alpha > 20^\circ$. However, the major variations in the cross-coupling curves described above are still larger in magnitude than the scatter.

It is premature at this point to attempt an indepth interpretation of the aerodynamic phenomena responsible for the derivative behavior based on the preliminary results obtained. It is also beyond the scope of this thesis. However, some general comments can be made:

- (i) At moderate angles of attack, strong vortices are formed on the SDM leading edge extensions. These vortices become unstable and finally break down near the trailing edge. It may be possible that this highly localized phenomenon is responsible for the pronounced variations in the dynamic derivatives (seen on a much smaller scale in the cross-coupling derivatives) between an α of 15° and 20° . This would be consistent with observations made by Beyers (Ref. 38) from previous SDM pitch oscillation experiments. It may also be possible that the notable behavior of the moment derivatives in the vicinity of $\alpha = 32^\circ$ is due to the establishment of main flow separation. This characteristic of the SDM configuration is also in agreement with previously published data (Refs. 38, 39).
- (ii) In Chapter 3, it was stated that under oscillatory conditions, the motions of the vortical flows formed over the components of the aircraft do not take place simultaneously with the motion of the aircraft, but with a time delay due to a convective time lag.

As a result, aerodynamic reactions that are both in-phase and out-of-phase with the model motion can be expected. Evidence of this time lag are the opposing trends between static and dynamic derivative results (particularly in the in-plane derivatives).

- (iii) It is apparent there is some discrepancy between $\partial C_m / \partial \alpha$ obtained from static coefficients (Fig. 6.5) and $C_{m_a}^*$, the static counterpart of $C_{m_d}^*$ (Fig. 6.7). Although the general trends are somewhat similar, $C_{m_a}^*$ is surprisingly steady in comparison and always positive. It is possible there was an undetected systematic error when determining the static coefficients, but this is highly unlikely in view of the consistency of C_m , C_l , C_n with previous SDM experiments. In an attempt to reconcile the two results, it is plausible to turn to the mathematical model. Recall from Bryan (Chapter 2), the aerodynamic moments resulting from a small disturbance from the reference flight condition are expressed as a Taylor series in the displacement variables and their time derivatives; i.e.,

$$M_{aero} = M_z \dot{z} + M_{\dot{z}} \ddot{z} + M_{\ddot{z}} \ddot{\dot{z}} + M_{\ddot{\dot{z}}} \ddot{\ddot{z}} + \dots \quad (6.4)$$

The classical assumption of linear aerodynamic theory is to accept a first order Taylor series approximation as a representation of the aerodynamic moment. Thus the mathematical model allows for higher order terms which are assumed to be negligible.

The static aerodynamic pitching moment measurements depend only on a change in α , itself, and not on any time derivatives. Recall that under oscillatory conditions, the measured M_{aero} (oscillatory component of the aerodynamic pitching moment (equation (5.19)) out-of-phase with the primary motion (for a sinusoidal primary oscillation) yields $C_{m_a}^*$ (equation (5.27)), assumed to be due to a change in $\dot{z}$ alone. Ascribing this phenomenon to one cause yields a discrepancy between $\partial C_m / \partial \alpha$ from static coefficients and the stability derivative $C_{m_a}^*$. It is suspected that the contribution to the variation of M_{aero} under oscillatory conditions is due to other effects than $\dot{z}$ alone; i.e., the oscillatory components are contaminated by the higher order terms in the Taylor expansion (acceleration effects) which may be significant for translational oscillation. Thus it is possible that additional terms in equation (6.4) out-of-phase with the primary motion ($\ddot{z}$, etc.) must be retained for sufficient accuracy. The measurement technique used cannot differentiate between the additional effects during one experiment, however, as the additional terms of the equation are a function of frequency, the acceleration effects can be isolated by repeating the experiment at several different frequencies (under the same test conditions). The abovementioned phenomena could account for part of the discrepancy between static and oscillatory

results.

- (iv) The trend of the $C_{m\dot{\alpha}}$ curve (Fig. 6.7) is similar to that of the composite derivative ($C_{m\dot{q}} + C_{m\dot{\alpha}}$) obtained from previous SDM oscillation-in-pitch tests (Ref. 38) at zero sideslip ($M_\infty = 0.6$, $Re = 10.4 \times 10^6$). One must be cautious when attempting a rigorous comparison as variations are expected due to Mach number differences and Reynolds number dependency (particularly at $\alpha > 20^\circ$). Also, the oscillatory components measured may be contaminated with higher order terms and there is no reason for the higher order terms obtained during a pitching experiment to be the same as those obtained during a translational oscillation experiment. Nevertheless, the current investigation demonstrates that acceleration effects are substantial at high angles of attack and thus it is possible that a major part of $C_{m\dot{q}} + C_{m\dot{\alpha}}$ above $\alpha = 20^\circ$ is due to $\dot{\alpha}$ effects.
- (v) Recall that acceleration derivatives owe their existence to the fact that the pressure distribution on the wing or tail does not adjust itself instantaneously to its equilibrium value when the angle of attack is suddenly changed. Etkin (Ref. 1) presents an indepth discussion of the two effects. The contribution of the tail (based on the lag of the downwash theory) is always present. It may be possible to separate the effect due to the wing flow alone (at high angles of attack) by repeating the experiment without the horizontal tail on the SDM.

Fig. 8.4 Static aerodynamic coefficients

Fig. 6.5 Static stability parameter determined from static data

Fig. 6.6 Direct derivatives due to translational oscillation

Fig. 6.7 Cross derivatives due to translational oscillation

Fig. 6.8 Cross-coupling derivatives due to translational oscillation

Fig. 6.9 Cross-coupling derivatives due to translational oscillation

Fig. 6.10 Summary of dynamic moment derivatives due to translational oscillation

Chapter 7

CONCLUSIONS AND FUTURE RESEARCH

7.1 CONCLUSIONS

- (1) The new NAE electromechanically driven translational oscillation apparatus has been described which is suitable for translational oscillation dynamic stability tests along either the z or y body axis. It has been possible to fine tune the apparatus to provide pure translational motion to the aircraft model despite sting bending.
- (2) A dynamic calibration, conducted to verify the moment measurement and data acquisition system, has produced excellent results.
- (3) Translational oscillation (along z body axis) wind tunnel tests were conducted on a simplified fighter aircraft configuration at Mach 0.25 from angles of attack 0° to 42.5° and zero sideslip, yielding a set of direct, cross and cross-coupling derivatives. The determination of these, never-before-measured, translational acceleration derivatives has served to demonstrate the capability of the apparatus and measurement technique. The apparatus has proven operational for the high- α range up to 42.5° .
- (4) The results have shown that translational acceleration derivatives can attain significant values as indicated by the magnitudes of $G_{z\dot{\alpha}}$ and $C_{m\dot{\alpha}}$ at higher angles of attack. Nonlinear trends are confirmed in all the derivatives. Cross-coupling derivatives were measured for the model at zero sideslip, although the levels were small.

7.2 FUTURE RESEARCH

As indicated previously in this report, nonlinear effects may have introduced a discrepancy between the static derivatives obtained from oscillatory measurements and those obtained from static coefficients. This disagreement must be resolved, if possible, by oscillating at several discrete frequencies, either by adding weights to the aluminium SDM or by testing with the brass SDM.

In addition, to properly interpret the aerodynamic flow phenomena responsible for the derivative behavior, a definitive program is required to:

- (i) complete the proposed test program (Table 6.1) at $\beta = 5^\circ$ to enable an examination of cross-coupling effects. Work is under way to provide this information;
- (ii) perform a systematic investigation of the effects of varying test conditions, such as Mach number, Reynolds number, amplitude of oscillation and configuration parameters. Some of these variations may not be feasible due to the requirement for an induced angle of attack sufficient to cause measurable aerodynamic effects;
- (iii) gather data at two different sting lengths to examine the effect of support interference on the data. It can be appreciated that this is a lengthy task, requiring readjustment of the suspension system to maintain pure translational motion and therefore recalibration of the apparatus in the longitudinal DOF;
- (iv) carry out the wind tunnel investigation at high angles of attack without the horizontal tail to isolate the contribution of the wing to the $\dot{\alpha}$ derivatives, and
- (v) complete a flow visualization study to correlate with the derivative data.

The experience of the investigation in this thesis would suggest that some acceleration derivatives may attain sizeable magnitudes and therefore should be carefully evaluated and properly taken into account in the aircraft equations of motion.

Future application of the translational oscillation apparatus includes a similar test program for lateral oscillation. The apparatus is currently being prepared for this.

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Appendix A

GEOMETRICAL ANALYSIS OF SUSPENSION SYSTEM MOTION

The purpose of the following analysis is to obtain a mathematical expression for and investigate the behavior of the "artificially induced" model/balance pitching motion (introduced in Chapter 4), necessary to counteract sting pitching oscillation.

A simplified sketch of the cantilever suspension illustrating its movement relative to a fixed sting is shown in Figure A-1(a). The following terminology applies:

d_1 — sting diameter at aft end of cantilevers with shims in position (mm), (varies with shim thickness)

d_2 — cantilever beam length = 76.2 mm

d_s — shim thickness (mm), (variable)

ζ_0 — cantilever convergence angle at equilibrium position (deg), (varies with shim thickness)

$\Delta\zeta$ — induced cantilever deflection under oscillatory conditions (deg)

$\Delta\zeta_d$ — maximum cantilever deflection under oscillatory conditions (deg)

$\Delta\zeta_s$ — cantilever deflection due to static loads (deg)

η_s — phase angle of sting pitching oscillation with respect to primary motion (deg)

θ_b — artificially induced model/balance pitch angle (deg)

θ_d — sting pitching oscillation (deg)

ω — circular frequency of primary oscillation (rad/sec)

For a sinusoidal excitation, $\Delta\zeta$ is sinusoidal and of the form;

$$\Delta\zeta(t) = \Delta\zeta_s + \Delta\zeta_d \sin(\omega t) \quad (A.1)$$

where $\Delta\zeta_d = 1.91^\circ$ for a maximum amplitude of oscillation of ± 2.54 mm , and the magnitude of $\Delta\zeta_s$ increases with increasing model angle of attack.

Fig. A-1(a) Geometry of planar motion of suspension system

Fig. A-1(b) Geometry of phase relationship between sting oscillation and model/balance motion

From Figure A-1(a):

$$d_1 = 22.2 \text{ mm} + 2d_s \quad (\text{A.2})$$

and

$$\zeta_o = \sin^{-1}(d_s/d_2) \quad (\text{A.3})$$

and

$$d_3 = d_1 - 2d_2 \sin \zeta_o \quad (\text{A.4})$$

In triangle ABC:

$$\overline{AB} = d_1$$

$$\overline{BC} = d_2$$

$$\angle ABC = 90^\circ - (\zeta_o + \Delta\zeta(t)) \quad (\text{A.5})$$

i) Applying the Law of Cosines along with equation (A.5);

$$\overline{AC}^2 = d_1^2 + d_2^2 - 2d_1d_2 \sin(\zeta_o + \Delta\zeta(t)) \quad (\text{A.6})$$

ii) By inspection;

$$\angle BAC = \sin^{-1} \left[\frac{d_2 \cos(\zeta_o + \Delta\zeta(t))}{\overline{AC}} \right] \quad (\text{A.7})$$

In triangle ACD:

$$\overline{AC} - \text{obtained from equation (A.6)}$$

$$\overline{AD} = d_2$$

$$\overline{CD} = d_3$$

i) Applying the Law of Cosines;

$$\angle DAC = \cos^{-1} \left[\frac{(\overline{AC})^2 + (\overline{AD})^2 - (\overline{CD})^2}{2(\overline{AC})(\overline{AD})} \right] \quad (\text{A.8})$$

Therefore, by inspection;

$$\angle DAE = \angle BAC + \angle DAC - 90^\circ \quad (\text{A.9})$$

Also from the figure;

$$\overline{AF} = d_2 \cos(\angle DAE) \quad (\text{A.10})$$

and

$$\overline{BG} = d_2 \cos(\zeta_o + \Delta\zeta(t)) \quad (\text{A.11})$$

Finally, θ_b is given by;

$$\theta_b(t) = \sin^{-1} \left[\frac{\overline{AF} - \overline{BG}}{d_3} \right] \quad (A.12)$$

Upon substitution of equations (A.1) to (A.11) into equation (A.12), a geometrical formulation of the artificially induced model/balance pitch angle, θ_b , is obtained. Three questions arise:

- i) Is the motion of θ_b a pure sinusoid ?
- ii) What is the required shim thickness (d_s) for θ_b to precisely counteract the sting pitching oscillation, θ_d ?
- iii) Will the phase difference (if any) between sting pitching oscillation and model/balance oscillation affect the efficiency of the compensation scheme ?

The first problem is of immediate concern. If θ_b is not a pure sinusoid, it will impart unwanted harmonics to the primary oscillation, an undesirable situation. A Fortran computer program, written specifically for this investigation, constructs the waveform θ_b (equation (A.12)), for a specified shim size, in the time domain for 256 points in a cycle. A Fast Fourier Transform (FFT) algorithm converts θ_b to the frequency domain and generates the magnitude and phase for the fundamental component and 9 harmonics. The outcome of the exercise disclosed harmonics of insignificant magnitude, indicating that θ_b is essentially a pure sinusoid.

Simulating the inertial loads of approximately 220 N encountered during model oscillation, the apparatus was statically loaded resulting in a sting pitching angle of 0.078°. Provision was made in the computer program to vary shim thickness until the artificially induced model/balance pitching, θ_b , was of equal magnitude and opposite direction. The required shim thickness was calculated to be 0.38 mm.

Ideally, the compensation scheme would yield the desired pitching angle $\theta_d - \theta_b \approx 0^\circ$, where $\theta_d = |\theta_d| \sin(\omega t + \eta_s)$. However, for $\eta_s \neq 0$, it can be seen from the vector diagram of Figure A-1(b) that for kinematic reasons the correction addresses only the component of θ_d in-phase with θ_b , resulting in;

$$-\theta_d - \theta_b = \theta_d \times 2 \sin \left(\frac{\eta_s}{2} \right) \quad (A.13)$$

The value of η_s depends on the damping factor and frequency ratio which are estimated to be approximately 0.015 and 1/3 respectively, for this apparatus. From response curves in damping (Ref. 36), $\eta_s \sim 10 \rightarrow 20^\circ$, and equation (A.13) becomes;

$$\theta_d - \theta_b \approx 0.35\theta_d \quad (A.14)$$

meaning that in the worst case, the shims compensate for approximately $2/3$ of sting pitching oscillation. It is concluded that the efficiency of the compensation scheme is reduced if a phase shift exists between the primary motion and sting oscillation. The following contains the previously discussed program listing:

FTN7X

PROGRAM PLUNG

```
C*****
C
C   DESCRIPTION: This program calculates the shim thickness required for "artificially induced" model/balance pitching motion("THETAB") to counteract oscillatory sting bending and determines the sinusoidal nature of THETAB. The program reads inputs consisting of estimated shim thickness and cantilever deflection angles under static and dynamic conditions for primary oscillation  $\pm 2.64\text{mm}$ . The waveform THETAB is constructed in the time domain for 256 points in a cycle and then an FFT algorithm converts it to frequency domain, generating the magnitude and phase of fundamental component and 9 harmonics. Exact shim thickness is obtained through iteration.
C
C   AUTHOR: J. Kuzina
C   DATE: Feb., 1985
C*****
C   IMPLICIT NONE
C
C   REAL AC, AF, ZETAO, BAC, THETAB(256), B, DAC, DAE, DZETA, DZETAD, DZETAS
C   REAL D1, D3, D2, DS, OMEGAT, PI, BG, AMP, PHASE
C
C   INTEGER INC, LU, I
C
C   COMPLEX THETAC(129)
C
C   EQUIVALENCE (THETAB(1), THETAC(1))
C
C   READ IN INPUTS (SI UNITS)
C   WRITE(1, '("Enter LU number for output")')
C   READ(1, *) LU
C   WRITE(1, '("Enter DS, DZETAS, DZETAD")')
C   READ(1, *) DS, DZETAS, DZETAD
C   WRITE(LU, '("DS (mm.) = "F9.5")') DS
C   WRITE(LU, '("DZETAS (rad.) = "F9.5" DZETAD (rad.) = "F9.5, /")')
C   1 DZETAS, DZETAD
C
C   CALCULATE THETAB VS. TIME FROM GIVEN GEOMETRY
C   PI = 3.141592654
C   D1 = 22.2 + 2.0 * DS
C   D2 = 76.2
C   ZETAO = ASIN(DS/D2)
C   D3 = D1 - 2.0 * D2 * SIN(ZETAO)
C   WRITE(LU, '("//" OMEGAT THETAB (deg.)"/')
C   DO INC = 0, 255
C     OMEGAT = 1.0 * INC / 256 * 2.0 * PI
C     DZETA = DZETAS + DZETAD * COS(OMEGAT)
C     AC = SQRT((D1**2) + (D2**2) - 2. * D1 * D2 * SIN(ZETAO + DZETA))
C     BAC = ASIN(D2 * COS(ZETAO + DZETA) / AC)
C     DAC = ACOS(((AC**2) + (D2**2) - (D3**2)) / (2.0 * AC * D2))
C     DAE = BAC + DAC - 1.570796327
C     AF = D2 * COS(DAE)
C     BG = D2 * COS(ZETAO + DZETA)
C     THETAB(INC+1) = ASIN((AF - BG) / D3)
C     IF((INC/5) * 5.EQ. INC) THEN
C       WRITE(LU, ' (F9.5, 2X, F9.5)') OMEGAT, THETAB(INC+1) * (180. / PI)
C     ENDIF
C   ENDDO
C
C   DO FFT
C
C   CALL RFFT(THETAB, 256, -1)
```

C PRINT DATA
WRITE(LU, '("//FREQUENCY DOMAIN (deg.)"/)')
DO I=1,10
WRITE(LU,*) (THETAC(I)/128.0)*180./PI

ENDDO

C
C
POLAR TO RECTILINEAR FOR FUNDAMENTAL COMPONENT
AMP=SQRT((THETAB(3)/128.0)**2+(THETAB(4)/128.0)**2)*180./PI
PHASE=ATAN(THETAB(4)/THETAB(3))*180./PI
WRITE(LU, '("//AMPLITUDE = "F9.5,/"PHASE (deg.) = "F9.5)') AMP, PHASE
STOP
END
END\$

Appendix B

DETAILS OF STATIC CALIBRATION

Calibration Equipment

The entire balance/suspension/sting assembly was mounted in a fixture (on a "surface table" outside the wind tunnel) which allowed it to be rotated about its longitudinal axis through 360°, thus allowing the application of positive and negative loads to the balance in the normal force or sideforce plane. Since it was not practical to calibrate with the actual aircraft model, a cylindrically shaped "calibrating body" that permitted loading the balance in the same manner as the model was used. It was machined to specification for this investigation and has precisely located holes "stations" drilled into its surface along the longitudinal axis for positioning of calibration weights. An inclinometer for measurement of angular deflection, and a meter for measurement of vertical deflection were mounted on the calibrating body.

Note: Calibration weights and measuring devices were available in U.S. customary units only, therefore, to preserve the accuracy of the calibration all calculations were carried out to completion using the U.S. system.

Procedure

- (1) The apparatus was positioned in the plunging configuration and the balance leveled at zero load conditions.
- (2) Weights were hung from the calibrating body through a range of 50 lbs (in 5-10 lb increments) at 5 separate stations (equivalent to forward and aft distances from the apparatus reference center (RC)). The apparatus was rolled 180° and the procedure repeated so that normal force loads were applied in both positive and negative directions.
- (3) A cross bar and pulley arrangement on the calibrating body enabled the balance to be subjected to positive and negative rolling moment. Weights were applied through a range of 5 lbs (in 1 lb increments) at 3.16 in from the RC along the balance lateral

axis.

- (4) Procedures (1) and (2) were repeated with the balance rolled precisely 90° for loading in the sideforce plane. Due to electrical wiring constrictions, only negative sideforce loads were applied.

During all loadings, simultaneous readings of angular deflection (V/deg), vertical deflection (in) and strain-gauge outputs (mV) from the cantilever beam and each flexure were recorded.

Results

Figures B-1 to B-9 display the results of the static calibration. Data points represent the change in output or deflection with change in load for each flexure at a particular distance from the reference center. An "in-house" two-dimensional plotting program (*GRAFIT*), capable of linear regression analysis, was used to accurately define the calibration slopes and graphically display the results.

In Figures B-1 to B-4, the slope of the straight line through data points indicates the sensitivity of the cantilever or particular flexure. For the cases of the cantilever calibration (Fig. B-1) and roll flexure calibration (Fig. B-4), sensitivities were obtained from the average of the slopes for positive and negative applied force. It was also necessary to ascertain the effective location of the pitch and yaw flexure strain-gauges, called the "electrical center". This is the point where the variation of output with applied force is zero; i.e., the x -axis intercept at zero load. From Figure B-2, the electrical center of the pitch flexure coincides with the apparatus RC, necessary for accurate determination of pitching moment. The exact locations of forward and aft yaw flexures (Fig. B-3) are required to resolve the two yaw moments into a single moment about the RC.

All load data exhibit a high degree of linearity and repeatability, in particular for the roll calibration (Fig. B-4) where the scatter at each load (all loads applied twice) is negligible and remains unnoticed.

The vertical deflection measured at each applied normal force was used to determine the stiffness (k) of the apparatus in the primary DOF (Fig. B-5). From the graph;

$$k_{\text{primary}} = \frac{1}{\text{slope}} = 451.36 \text{ lb/in}$$

The change in balance deflection angles with change in load at a particular station were used to determine the stiffness in the secondary DOFs. From Figure B-6;

$$k_{\text{pitch}} = \frac{1}{\text{slope}(\text{pitch DOF})} \times \frac{180}{\pi} = 14,000 \text{ lb in/rad}$$

Similarly,

$$k_{yaw} = 7,800 \text{ lb in/rad}$$

Note: An error analysis was not computed since only approximate stiffnesses were required.

Balance interactions were determined during the calibration procedure. The interactions of loadings in the normal force plane on the roll and yaw gauges (Fig. B-7), of loadings in the sideforce plane on the roll and pitch gauges (Fig. B-8) and of rolling moment on the yaw gauges (Fig B-9) are linear.

Interaction percentages are defined as the moment that produces the interaction signal in terms of the actual moment or force applied in the interacting DOF, for example, interaction in roll due to sideforce (from Figs. B-8 and B-4)

$$= \frac{2.21 \times 10^{-2} \text{ mV/lb in}}{2.445 \text{ mV/(lb in)}} = 0.90\%$$

Similarly, pitch interaction with sideforce = 0.18%; forward yaw, aft yaw and roll interaction with normal force are 1.1%, 0.12% and 0.32% respectively, and forward and aft yaw interaction with rolling moment are 1.0% and 2% respectively.

A drive force calibration was obtained by applying a known load to the calibrating body (in the normal force plane) which resulted in a deflection. The DC current to the drive coil required to restore the calibrating body to its original position constituted the desired calibration. As the emphasis in this experiment was on moment derivatives and not force derivatives, the calibration procedure was repeated for only 2 applied loads and therefore the calibration is approximate.

The above findings have been converted to SI units for compatibility with the data acquisition system and are referred to in SI units throughout the text.

Notes

- (1) Loads were applied at +0.03 in from RC
- (2) Average slope = 0.1835 mV/lb
- (3) Uncertainty band falls within displayed symbol size

Fig. B-1 Cantilever beam calibration

Notes

- (1) Each individual ordinate value represents the slope of the pitch flexure output with 11 separately applied loads.
- (2) Uncertainty band falls within displayed symbol size

Fig. B-2 Pitch flexure calibration

Notes

- (1) Each individual ordinate value represents the slope of the yaw flexure output with 11 separately applied loads
- (2) Uncertainty band falls within displayed symbol size

Fig. B-3 Yaw flexure calibration

Notes

- (1) Average slope = $2.445 \text{ mV}/(\text{lb in})$
- (2) Uncertainty band falls within displayed symbol size

Fig. B-4 Roll flexure calibration

Notes

- (1) Loads were applied at +0.08 in from RC
- (2) Magnitude of average slope = 2.215×10^{-3} in/lb
- (3) Uncertainty band falls within displayed symbol size

Fig. B-5 Stiffness in primary degree of freedom

Notes

- (1) Each individual ordinate value for each of the above curves represents the resulting slope of the inclinometer output with 11 separately applied loads.
- (2) Magnitude of average slope for pitch DOF = 4.12×10^{-3} deg/(lb in)
- (3) Uncertainty band falls within displayed symbol size

Fig. B-6 Stiffness in pitch and yaw degrees of freedom

Notes

- (1) Each individual ordinate value represents the slope of the flexure output with 11 separately applied loads
- (2) Uncertainty band falls within displayed symbol size.

Fig. B-7 Roll and yaw interaction with normal force

Notes

- (1) Each individual ordinate value represents the slope of the flexure output with 11 separately applied loads
- (2) Uncertainty band falls within displayed symbol size

Fig. B-8 Roll and pitch interaction with sideforce

Notes

- (1) Each individual ordinate value represents the slope of the yaw flexure output with 11 separately applied loads
- (2) Uncertainty band falls within displayed symbol size

Fig. B-9 Yaw interaction with rolling moment

Appendix C

DETAILS OF DATA REDUCTION

C.1 Calculation Sequence for Dynamic Derivatives

This Appendix deals with the subroutine *AERO* which is part of the standard signal processing main program (used in forced-oscillation experiments at UAL) which must be rewritten for the particular experiment. *AERO* provides preliminary data evaluation of each test point from a single wind tunnel run utilizing the data reduction equations ((6.30) to (6.33)) and expressions ((6.16), (6.17), (6.26) to (6.29)) derived in Chapter 5. The subroutine makes use of an "apparatus matrix" in all calculations, part of which contains the quantities determined from static and dynamic calibration of the apparatus. The remaining portions of the matrix are calculated and filled by the main program during each wind tunnel run.

The following contains a copy of the apparatus matrix, a listing of subroutine *AERO* and an example data output for a single test run in the translational oscillation test series:

PULSING. →		LATERAL →		START CHANNEL	SCAN INTERVAL (ms)	SCANS/RECORD	NUMBER OF RECORDS	MAXIMUM LAG NUMBER	CALIBRATION CENTER FREQUENCY
				10	4	280	4	30	45
				NUMBER OF CHANNELS	TOTAL NUMBER OF WORDS		RUN REPEAT NUMBER	TARE PERIOD(S)	WIND-ON PERIOD(S)
				6					
				SENSITIVITY	STIFFNESS	MOMENT OF INERTIA	DISTANCE FROM REFERENCE CENTER	MASS	
				$4.11 \times 10^{-5} \text{ V/N}$	79,080 N/m				
				$1.02 \times 10^{-2} \text{ V/N}$					
				$6.41 \times 10^{-2} \text{ V/N-m}$					
				$6.88 \times 10^{-2} \text{ V/N-m}$			0.04755 m		
				$7.39 \times 10^{-2} \text{ V/N-m}$			0.01572 m		
				$2.40 \times 10^{-2} \text{ V/N-m}$					

CHANNEL →

Table C-1 Apparatus Matrix

FTN7X

```

C SUBROUTINE AERO(PO,TO,P,SIGMA,PHIP,MODE,IREC,DUM2)
C ELECTROMECHANICAL TRANSLATIONAL OSCILLATION APPARATUS ON
C SIDE SUPPORT
C STANDARD DYNAMICS MODEL
C SI UNITS
C MODE 6=PLUNGING++ OSC. MODE 5=LATERAL OSC.
C * * * * *
C
C PURPOSE: Subroutine AERO provides preliminary data evaluation
C of each test point from a single wind tunnel run.
C AUTHOR: J.Kuzina
C DATE: June,1985
C
C * * * * *
C GLOSSARY OF SYMBOLS
C ALPHA angle of attack,body axes system
C AMPLT,AMPLW tare.wind-on amplitude of primary oscillation
C AMPLWD AMPLW(degrees),delta ALPHA or delta BETA
C AMPLWM AMPLW(mm)
C AMTRX apparatus matrix
C B wingspan
C BETA angle of sideslip,body axes system
C CBAR wing mean aerodynamic chord
C CNS,CND constants for non-dimensionalization of static,
C dynamic stability derivatives
C CORD matrix of voltages measured (inphase,quad.form)
C C(1)-C(10) stability derivatives
C C(11) reduced frequency
C FREQT,FREQW tare.wind-on frequency of oscillation
C ID identification number of test conditions(test matrix)
C IREC run number(IREC less than 0 for calibration)
C MACH freestream Mach number
C MOMT matrix of moments,forces measured(inphase,quad.form)
C MU viscosity
C NCH number of channels
C PHIP aerodynamic angle of bank
C PO,P stagnation,static pressure
C Q freestream dynamic pressure
C RE Reynolds number per meter(*E-6)
C REL Reynolds number based on reference length(*E-6)
C RHO freestream density
C RL reference length
C S wing plan area
C SIGMA aerodynamic angle of attack
C TO,T stagnation,static temperature
C PREFIX:
C M indicates quantity converted to SI units
C D indicates quantity to data storage
C
C DATA STORAGE
C NASSDP data file for plunging oscillation
C NASSDL data file for lateral oscillation
C DATA(1) IREC
C DATA(2) ID
C DATA(3) ALPHA
C DATA(4) BETA
C DATA(5) SIGMA
C DATA(6) PHIP
C DATA(7) to DATA(17) C(1) to C(11)
C DATA(18) FREQW
C DATA(19) AMPLWM
C DATA(20) AMPLWD
C DATA(21) MACH
C DATA(22) REL
C DATA(23) V

```

⁺⁺ For simplicity,'plunging' refers to translational osc. experiments along z body axis in all computer programs.

C
C
C
C
C
C
C

DATA(24) Q
DATA(25) P
DATA(26) PO
DATA(27) TO

* * * * *

REAL MACH,MU,DATA(40),C(11)
COMPLEX CORD(8,2),MOMT(8,2)
CHARACTER YESNO*1
DIMENSION IDC(144),NAM7(3)
COMMON CORD,AMTRX(10,6)
EQUIVALENCE (CORD,MOMT)
EQUIVALENCE (DATA(1),DIREC),(DATA(2),DID),(DATA(3),ALPHA),
1 (DATA(4),BETA),(DATA(5),SIG),(DATA(6),PHI),(DATA(7),C(1)),
2 (DATA(18),FREQW),(DATA(19),AMPLWM),(DATA(20),AMPLWD),
3 (DATA(21),MACH),(DATA(22),REL),(DATA(23),V),(DATA(24),Q),
4 (DATA(25),DPM1),(DATA(26),DPOM1),(DATA(27),DTOM)

DATA NAM7/'NASSDP',S,CBAR,B/3*.1/

DATA RTD/57.2958/,PI/3.14159/

IF(MODE.EQ.5)THEN

NAM7(3)=NAM7(3)-4 ! change NASSDP to NASSDL

ENDIF

DIREC=IREC

SIG=SIGMA

PHI=PHIP

WRITE(1,('Enter I.D. number from test matrix:'))

READ(1,*) ID

WRITE(6,5) ID

DID=ID

5 FORMAT(/3X,"TRANSLATIONAL OSC. DATA REDUCTION"/2X,

1 " MODEL-SDM ID NUMBER = "I3)

C
C

STANDARD DYNAMICS MODEL PARAMETERS

S=174.12E-4

B=.2286

CBAR=.0863

C
C

COMPUTE TUNNEL CONDITIONS:

(a) CONVERT SIGMA,PHIP FROM AERODYNAMIC AXES TO ALPHA,BETA IN BODY
AXES SYSTEM

IF(PHIP.NE.0.)THEN

ALPHA=ATAN(TAN(SIGMA/RTD)*COS(PHIP/RTD))*RTD

BETA=ASIN(SIN(SIGMA/RTD)*SIN(PHIP/RTD))*RTD

ELSE

ALPHA=SIGMA

BETA=PHIP

END IF

C
C

(B) CALCULATE FLOW PARAMETERS

MACH=(P/PO)**(-2./7.)

MACH=SQRT(5.*(MACH-1.))

NCH=AMTRX(2,1)

POM=PO*101325./760.

Q=0.7*MACH**2*(1.+MACH**2/5.)*(-3.5)*POM

T=TO/(1.+MACH**2/5.)

TM=T*5./9.

V=20.07*SQRT(TM)*MACH

PM=P*101325./760.

TOM=TO*5./9.

RHQ=PM/287.6/TM

MU=1.448E-6*TM**1.5/(TM+110.4)

RE=1.E-6*RHO*V/MU

IF(MODE.EQ.6)THEN

RL=CBAR

ELSE

RL=B

END IF

REL=RE*RL

```

PM1=PM/1000.
DPM1=PM1
POM1=POM/1000.
DPOM1=POM1
DTOM=TOM
WRITE(6, '("AMTRX CONTENTS")')
WRITE(6, *) (AMTRX(I,1), I=3,6)
IF(IREC.LT.0) GO TO 10
C
C
C
10 TRANSFORM VOLTAGES INTO PHYSICAL QUANTITIES
AMPLT=CABS(CORD(1,1))/AMTRX(3,1)/AMTRX(3,2)
AMPLW=CABS(CORD(1,2))/AMTRX(3,1)/AMTRX(3,2)
DO 20 I=2,NCH
    MOMT(I,1)=CORD(I,1)/AMTRX(I+2,1)
    MOMT(I,2)=CORD(I,2)/AMTRX(I+2,1)
20 AMPLTM=AMPLT*1000.
    AMPLWM=AMPLW*1000.
    AMPLWD=AMPLW*2.*PI/AMTRX(2,6)/V*RTD
C
C
C
C COMPUTE FREQUENCY OF OSCILLATION
FREQT=1./AMTRX(2,5)
FREQW=1./AMTRX(2,6)
C
C
C COMPUTE DIRECT DERIVATIVES
CNS=V/(Q*S)
CND=CNS*2.*V
C(9)=CND*(AMTRX(3,2)*((AMTRX(2,5)/AMTRX(2,6))**2-1.)*
1REAL(MOMT(2,2))/AMPLW-REAL(MOMT(2,1))/AMPLT)*(AMTRX(2,6)/(2*PI))
2**2
IF(MODE.EQ.6) THEN
    C(9)=C(9)/CBAR
ELSE
    C(9)=C(9)/B
END IF
C(10)=-CNS*(AMTRX(2,6)*AIMAG(MOMT(2,2))/AMPLW
1-AMTRX(2,5)*AIMAG(MOMT(2,1))/AMPLT)/(2.*PI)
C
C COMPUTE FORCE FOR DIAGNOSTIC PURPOSES
MOMT(1,1)=CMPLX(C(10)*AMPLW/CNS*2.*PI/AMTRX(2,6), C(9)*AMPLW/CND*
1CBAR*(2.*PI/AMTRX(2,6))**2)
C
C
C COMPUTE AERODYNAMIC VECTORS AND STORE IN MOMT
DO 30 I=3,NCH
30 MOMT(I,1)=MOMT(I,2)-MOMT(I,1)*AMPLW/AMPLT
C
C
C
C RESOLVE MOMENTS MEASURED BY FORWARD & AFT YAW/PITCH FLEXURES
C WRT REFERENCE CENTER:
C (a) DETERMINE FORCE AND LOCATION WRT REFERENCE CENTER
C SIDE FORCE IF PLUNGING OSC., NORMAL FORCE IF LATERAL OSC.
C STORE FORCE IN MOMT(2,2). STORE LOCATION IN MOMT(3,2)
MOMT(2,2)=(MOMT(5,1)-MOMT(4,1))/(AMTRX(6,4)-AMTRX(7,4))
MOMT(3,2)=(MOMT(5,1)*AMTRX(6,4)-MOMT(4,1)*AMTRX(7,4))/
1(MOMT(5,1)-MOMT(4,1))
C
C
C
C (B) COMPUTE MOMENT ABOUT REF. CENTER, STORE IN MOMT(1,2)
C YAWING IF PLUNGING OSC., PITCHING IF LATERAL OSC.
MOMT(1,2)=MOMT(2,2)*MOMT(3,2)
WRITE(6,40)
40 FORMAT(/5X,"MOMENT MATRIX")
WRITE(6,50) ((MOMT(I,J), J=1,2), I=1,NCH)
50 FORMAT(/,2(F10.4,F10.4,5X))
C
WRITE(6,60) FREQT, AMPLTM, FREQW, AMPLWM, AMPLWD
60 FORMAT(/3X,"TARE FREQ=", F7.3, 10X, "TARE AMPL=", F7.4,
1/3X, "WIND ON FREQ=", F7.3, 7X, "WIND ON AMPL(MM)=", F7.4,
2/3X, "WIND ON AMPL(DEG.)=", F7.4/)
WRITE(6,70) ALPHA, BETA, SIGMA, PHIP, TO, PO, P, V, MACH, Q
70 FORMAT(3X, "ALPHA=", F4.1, 4X, "BETA=", F4.1, 4X, "SIGMA=", F5.2, 4X,
1"PHIP=", F5.2/6X, "TO=", F5.1, 4X, "PO=", F5.1, 4X, "P=", F5.1/6X, "V=",
2F6.1, 4X, "MACH=", F4.3, 4X, "Q=", F7.0/)

```

```

80      WRITE(6,80)S,CBAR,B
      FORMAT(/3X,"AREA=",F7.5,5X,"CHORD=",F7.5,5X,"SPAN=",F7.5)
C
C      COMPUTE CROSS DERIVATIVES
      IF(MODE.EQ.5)GO TO 110
C      CROSS DERIVATIVES FOR PLUNGING OSCILLATION
      C(1)=- (CND/(CBAR**2))*REAL(MOMT(3,1))/AMPLW*(AMTRX(2,6)/
1(2.*PI))**2
      C(2)= (CNS/CBAR)*AMTRX(2,6)*AIMAG(MOMT(3,1))/(AMPLW*2.*PI)
      C(3)= - (CND/B/CBAR)*REAL(MOMT(1,2))/AMPLW*(AMTRX(2,6)/(2.*PI))**2
      C(4)= (CNS/B)*AMTRX(2,6)*AIMAG(MOMT(1,2))/(AMPLW*2.*PI)
      C(5)= - (CND/B/CBAR)*REAL(MOMT(6,1))/AMPLW*(AMTRX(2,6)/(2.*PI))**2
      C(6)= (CNS/B)*AMTRX(2,6)*AIMAG(MOMT(6,1))/(AMPLW*2.*PI)
      C(7)= - (CND/CBAR)*REAL(MOMT(2,2))/AMPLW*(AMTRX(2,6)/(2.*PI))**2
      C(8)= CNS*AMTRX(2,6)*AIMAG(MOMT(2,2))/(AMPLW*2.*PI)
      C(11)=FREQW*CBAR*PI/V
      WRITE(6,90)
90      FORMAT(/7X,"PLUNGING OSCILLATION"/)
      WRITE(6,100)(C(I),I=1,10)
100     FORMAT(/7X,"CMADOT=",F8.3,10X,"CMALFA=",F8.3//7X,"CNADOT=",F8.3,
110X,"CNALFA=",F8.3,10X,"CLADOT=",F8.3,10X,"CLALFA=",F8.3,10X,
2"CYADOT=",F8.3,10X,"CYALFA=",F8.3//7X,"CZADOT=",F8.3,10X,
3"CZALFA=",F8.3//)
      GOTO 140
C
C      CROSS DERIVATIVES FOR LATERAL OSCILLATION
110     C(1)=- (CND/(B**2))*REAL(MOMT(3,1))/AMPLW*(AMTRX(2,6)/(2.*PI))**2
      C(2)= (CNS/B)*AMTRX(2,6)*AIMAG(MOMT(3,1))/(AMPLW*2.*PI)
      C(3)= - (CND/B/CBAR)*REAL(MOMT(1,2))/AMPLW*(AMTRX(2,6)/(2.*PI))**2
      C(4)= (CNS/CBAR)*AMTRX(2,6)*AIMAG(MOMT(1,2))/(AMPLW*2.*PI)
      C(5)= - (CND/(B**2))*REAL(MOMT(6,1))/AMPLW*(AMTRX(2,6)/(2.*PI))**2
      C(6)= (CNS/B)*AMTRX(2,6)*AIMAG(MOMT(6,1))/(AMPLW*2.*PI)
      C(7)= - (CND/B)*REAL(MOMT(2,2))/AMPLW*(AMTRX(2,6)/(2.*PI))**2
      C(8)= CNS*AMTRX(2,6)*AIMAG(MOMT(2,2))/(AMPLW*2.*PI)
      C(11)=FREQW*B*PI/V
      WRITE(6,120)
120     FORMAT(/7X,"LATERAL OSCILLATION"/)
      WRITE(6,130)(C(I),I=1,10)
130     FORMAT(/7X,"CNBDOT=",F8.3,10X,"CNBETA=",F8.3,10X,"CMBDOT=",F8.3,
110X,"CMBETA=",F8.3,10X,"CLBDOT=",F8.3,10X,"CLBETA=",F8.3,10X,
2"CZBDOT=",F8.3,10X,"CZBETA=",F8.3,10X,"CYBDOT=",F8.3,10X,
3"CYBETA=",F8.3//)
C
C      DATA STORAGE TO DISK
C
140     CONTINUE
      IF(IREC.GT.0) THEN
        IRSBJ=IREC*4-3
        ! convert run # to record #
        OPEN(99,FILE=NAME7,IOSTAT=IER,ERR=9000,ACCESS='DIRECT',
+          STATUS='OLD',RECL=100,MAXREC=2000)
        WRITE(99,'(10F10.4)',IOSTAT=IER,ERR=9000,REC=IRSBJ)
+          (DATA(ISBJ),ISBJ=1,10)
        WRITE(99,'(10F10.4)',IOSTAT=IER,ERR=9000,REC=IRSBJ+1)
+          (DATA(ISBJ),ISBJ=11,20)
        WRITE(99,'(10F10.4)',IOSTAT=IER,ERR=9000,REC=IRSBJ+2)
+          (DATA(ISBJ),ISBJ=21,30)
        WRITE(99,'(10F10.4)',IOSTAT=IER,ERR=9000,REC=IRSBJ+3)
+          (DATA(ISBJ),ISBJ=31,40)
        CLOSE(99)
      ENDIF
      RETURN
      ! main return
C
9000    CLOSE(99)
      IF(IER.EQ.462.OR.IER.EQ.506) THEN
        WRITE(1,('File "3A2" does not exist!"/"Do you want it ",
+          "created for you? (Y/N)")) NAM7
        READ(1,'(A1)') YESNO

```

```

IF(YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN ! create + fill with 0
  OPEN(99,FILE=NAME,Iostat=IER,ERR=9000,ACCESS='DIRECT',
    STATUS='NEW',RECL=100,MAXREC=2000)
  + ZERO=0.0
  DO IRSBJ=1,2000
    WRITE(99,'(10F10.4)',REC=IRSBj) (ZERO,I=1,10)
  ENDDO
  CLOSE(99)
  IRSBJ=IREC*3-2 ! convert run # to record #
  OPEN(99,FILE=NAME,Iostat=IER,ACCESS='DIRECT',
    STATUS='OLD',RECL=100,MAXREC=2000)
  + WRITE(99,'(10F10.4)',Iostat=IER,REC=IRSBj)
    (DATA(ISBJ),ISBJ=1,10)
  + WRITE(99,'(10F10.4)',Iostat=IER,REC=IRSBj+1)
    (DATA(ISBJ),ISBJ=11,20)
  + WRITE(99,'(10F10.4)',Iostat=IER,REC=IRSBj+2)
    (DATA(ISBJ),ISBJ=21,30)
  + WRITE(99,'(10F10.4)',Iostat=IER,REC=IRSBj+3)
    (DATA(ISBJ),ISBJ=31,40)
  + CLOSE(99)
ENDIF
ELSE
  WRITE(1,("I/O error #"I3" in file access! "))IER
ENDIF
RETURN ! error return
END
END$

```


INPUT PARAMETERS: 63 52 6 0 0
 CAL. FREQ. 44.922
 CALIBRATION MATRIX FOR RUN # 63

Z	PHI	Z	PHI	Z	PHI
503.243	-.004	491.971	.002	478.159	.000
96.271	-.055	93.425	-.050	90.239	-.051
419.628	-.012	407.424	-.009	395.746	-.005
200.415	-.059	188.079	-.054	183.066	-.055
206.322	-.093	201.599	-.095	189.227	-.099
517.339	-.114	500.923	-.117	485.932	-.119

RUN # 63- 0

2:59 PM FRI., 18 OCT., 1985

RUNNING AVERAGE OF CORRELATIONS

TARE		WIND ON	
.4108E+01	.1203E-02	.3981E+01	.2542E-02
-.1454E-01	.7782E+00	-.1704E+00	.2815E+01
.2580E+01	.6147E-01	.4011E+01	.1340E+01
.4916E+00	.1012E-01	.5444E+00	.3121E+00
-.8130E-01	.7539E-01	-.2623E-01	.3422E+00
.2484E+00	-.4265E-01	.1335E+00	-.3507E-01
TARE		WIND ON	
.4108E+01	.1203E-02	.3979E+01	-.4762E-02
-.1454E-01	.7782E+00	-.1691E+00	.2852E+01
.2580E+01	.6147E-01	.4106E+01	.1328E+01
.4916E+00	.1012E-01	.5731E+00	.2825E+00
-.8130E-01	.7539E-01	-.3381E-02	.3137E+00
.2484E+00	-.4265E-01	.1530E+00	-.3893E-01
TARE		WIND ON	
.4108E+01	.1203E-02	.3977E+01	-.1110E-02
-.1454E-01	.7782E+00	-.1689E+00	.2839E+01
.2580E+01	.6147E-01	.4122E+01	.1276E+01
.4916E+00	.1012E-01	.5677E+00	.3041E+00
-.8130E-01	.7539E-01	-.9333E-02	.3331E+00
.2484E+00	-.4265E-01	.1596E+00	-.3277E-01
TARE		WIND ON	
.4108E+01	.1203E-02	.3976E+01	-.5710E-02
-.1454E-01	.7782E+00	-.1666E+00	.2822E+01
.2580E+01	.6147E-01	.4137E+01	.1244E+01
.4916E+00	.1012E-01	.5229E+00	.3218E+00
-.8130E-01	.7539E-01	-.4929E-01	.3515E+00
.2484E+00	-.4265E-01	.1674E+00	-.5595E-01

TRANSFER FUNCTIONS AT TARE AND W-O FREQS.

489.702030	1.025146	489.679930	1.025513
92.562897	-4.626801	92.557663	-4.626389
404.022280	-3.205141	404.002080	-3.202433
185.958500	-9.954651	185.945800	-9.953796
198.448970	-19.008369	198.431880	-19.007805
492.451900	-58.218910	492.425290	-58.217979

VOLTAGES AT SOURCE

.008388	-.000015	.008119	-.000029
-.000576	.008379	-.003316	.030327
.005384	.000203	.010215	.003160
.002633	.000195	.002712	.001876
-.000442	.000338	-.000414	.001732
.000508	-.000027	.000349	-.000072

TRANSLATIONAL OSC. DATA REDUCTION
MODEL-SDM ID NUMBER = 18
AMTRX CONTENTS
4.112E-5 1.02E-2 6.4E-2 6.882E-2

MOMENT MATRIX

-2.1780	-.2371	-.0009	.0163
-.0565	.8214	-.0687	-.1729
.0630	.0462	-.0796	-.0368
.0024	.0245	.0394	.0273
.0002	.0190	-.0056	.0234
-.0059	-.0019	.0145	-.0030

TARE FREQ= 45.407 TARE AMPL= 2.5795
WIND ON FREQ= 45.410 WIND ON AMPL(MM)= 2.4970
WIND ON AMPL(DEG.)= .4743

ALPHA=30.0 BETA= 0.0 SIGMA=30.00 PHIP= 0.00
T0=530.0 P0=731.9 P=700.4
V= 86.1 MACH=.251 Q= 4134.

AREA= .01741 CHORD= .08630 SPAN= .22860

PLUNGING OSCILLATION

CMADOT=	-8.559	CMALFA=	.899
CNADOT=	.046	CNALFA=	.120
CLADOT=	.305	CLALFA=	-.014
CYADOT=	.806	CYALFA=	-.290
CZADOT=	-2.781	CZALFA=	-3.655

C.2 Calculation Sequence for Static Derivatives

As outlined in Chapter 5, a Fortran computer program (*REDIT*) was written for handling of raw dynamic data and calculation of static data upon completion of the wind tunnel test series. (Similar programs were generated for the z axis and y axis translational oscillation cases, but only the z axis case is dealt with here). The raw voltages from chart recorder traces were entered manually and static aerodynamic coefficients for each test condition were calculated. The program is well commented, however, two particulars are elaborated:

- (i) Balance interactions which were not negligible ($>0.3\%$) have been incorporated as cross-sensitivities and corrections have been applied to the raw pitch, roll and yaw voltages. The corrections are of standard form in accordance with Pope (Ref. 44), for example;

Roll correction

$$\begin{aligned} \delta = & \left[Y \times \text{distance from roll flexure} \right] \times (\text{cross sensitivity due to } Y) \\ & + \left[Z \times \text{distance from roll flexure} \right] \times (\text{cross sensitivity due to } Z) \end{aligned} \quad (C.1)$$

where Z and Y are the normal and sideforce, respectively.

- (ii) The apparatus deflects due to static aerodynamic loads experienced under wind-on conditions, thus, the model angular position was not correctly measured. The amount of deflection was determined by the magnitude and location of the normal force and sideforce. Corrections were applied to the mean values of the aerodynamic angles when $\alpha \neq 0$, $\beta \neq 0$, and were of the form;

$$\alpha_o = Z \left(\frac{\text{pitch deflection}}{\text{load}} \right)_{(\text{at RC})} + \frac{M}{k_M} \quad (C.2)$$

$$\beta_o = Y \left(\frac{\text{yaw deflection}}{\text{load}} \right)_{(\text{at RC})} + \frac{N}{k_N} \quad (C.3)$$

where the slopes at the RC and apparatus stiffnesses have been determined in the static calibration (Fig. B-6).

The following contains the program listing:

FTN7X
\$FILES 1,1
\$EMA /BIGARR/
PROGRAM REDIT

C
C PURPOSE: This program was developed for data handling at the end
C of the plunging oscillation test series or for data
C inspection whenever required.

C AUTHOR: J.Kuzina
C DATE: Aug./85
C Standard dynamics model
C SI units

C *****
C GLOSSARY OF SYMBOLS

C ACOR ALPHA correction for static offset
C ALPHA angle of attack, body axes system
C AVGCNT average counter(# of elements needed for average)
C AVGDT array of raw data from AERO(500 runs,40 elements/run)
C AVGDYP plot file for dynamic data, plunging oscillation
C AVGSTP plot file for static data, plunging oscillation
C B wingspan
C BCOR BETA correction for static offset
C BETA angle of sideslip, body axes system
C CBAR wing mean aerodynamic chord
C D distance of Y from model reference center
C DA distance of NA from model reference center
C DATLOC data location
C DF distance of NF from model reference center
C DR distance of L from model reference center
C ELEMNT one of 40 elements stored/run
C ID identification number of test conditions(test matrix)
C IREC
C L roll (moment or flexure)
C M pitch (moment or flexure)
C N yaw (moment)
C NA aft yaw (moment or flexure)
C NASSDP dynamic data file(direct access)for plung. osc.(from AERO)
C NASSSP static data file(direct access) for plunging oscillation
C NF forward yaw (moment or flexure)
C RAWDT array of raw data from AERO (500 runs,40 elements/run)
C RHO freestream density
C RUNNUM run number
C S wing plan area
C TEMP temporary variable
C Y side force
C Z normal force

C
C PREFIXES: SUFFIX:
C C coefficient C corrected voltage or angle
C COR correction to voltage
C CS cross-sensitivity
C G gain
C K stiffness
C S sensitivity
C V voltage

C *****

C IMPLICIT NONE

C REAL RAWDT,AVGDT

C COMMON /BIGARR/ RAWDT(500,40),AVGDT(100,40)
C
C

```

      CALL RNASSD
      CALL ORDER
      WRITE(1,('("DONE ORDER")'))
      CALL AVERAG
      CALL STATIC
C
      END
C*****
C
C      Subroutine RNASSD to read AEROT data file NASSDP
C*****
$EMA /BIGARR/
      SUBROUTINE RNASSD
      IMPLICIT NONE
C
      INTEGER IER,ISBJ,IRSBJ,IREC
      REAL RAWDT,AVGDT
C
      CHARACTER FILNAM*6
C
      COMMON /BIGARR/ RAWDT(500,40),AVGDT(100,40)
C
      FILNAM='NASSDP'
      OPEN(99,FILE=FILNAM,IOSTAT=IER,ERR=9000,ACCESS='DIRECT',
+        STATUS='OLD',RECL=100,MAXREC=2000)
      DO IREC=1,500
        WRITE(1,*) IREC
        IRSBJ=IREC*4-3
        READ(99, '(10F10.4)',IOSTAT=IER,ERR=9000,REC=IRSBJ)
+        (RAWDT(IREC,ISBJ),ISBJ=1,10)
        READ(99, '(10F10.4)',IOSTAT=IER,ERR=9000,REC=IRSBJ+1)
+        (RAWDT(IREC,ISBJ),ISBJ=11,20)
        READ(99, '(10F10.4)',IOSTAT=IER,ERR=9000,REC=IRSBJ+2)
+        (RAWDT(IREC,ISBJ),ISBJ=21,30)
        READ(99, '(10F10.4)',IOSTAT=IER,ERR=9000,REC=IRSBJ+3)
+        (RAWDT(IREC,ISBJ),ISBJ=31,40)
      ENDDO
      CLOSE(99)
      RETURN
C
C      ! main return
C
C      Error handling
C
C      9000 WRITE(1,('I/O error #"I3" in access of "A6"!!!')) IER,FILNAM
      RETURN
      END
C*****
C
C      Subroutine ORDER to sort data in order of ID number
C*****
$EMA /BIGARR/
      SUBROUTINE ORDER
      IMPLICIT NONE
C
      INTEGER RUNNUM,ELEMNT,DATLOC
      INTEGER IRSBJ,ISBJ,IER,IREC
      REAL RAWDT,AVGDT,TEMP,NEWVALUE
      LOGICAL DONE,EDITDONE
      CHARACTER YESNO*1,FILNAM*6
C
      COMMON /BIGARR/ RAWDT(500,40),AVGDT(100,40)
C
C      Set unused ID # locations to 9999 (to go to bottom).
C
      FILNAM='NASSDP'
      DO RUNNUM=1,500
        IF(RAWDT(RUNNUM,2).EQ.(0.0)) THEN
          RAWDT(RUNNUM,2)=9999.

```

ENDIF
ENDDO

Sort data (bubble sort is crude but it works)

DONE=.FALSE.
DO WHILE(.NOT.DONE)
 DONE=.TRUE.
 DO DATLOC=1,499
 IF(RAWDT(DATLOC,2).GT.RAWDT(DATLOC+1,2)) THEN
 DONE=.FALSE.
 DO ELEMNT=1,40
 TEMP=RAWDT(DATLOC,ELEMNT)
 RAWDT(DATLOC,ELEMNT)=RAWDT(DATLOC+1,ELEMNT)
 RAWDT(DATLOC+1,ELEMNT)=TEMP
 ENDDO
 ENDIF
 ENDDO

To print after sort and before average

WRITE(1,('DO YOU WANT A PRINTOUT OF THE RAW DATA?'))
READ(1,('A1')) YESNO
IF (YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN
 IRSBJ=1
 DO WHILE((RAWDT(IRSBJ,1).NE.0).AND.(IRSBJ.LE.500))
 WRITE(40,('10F10.4'))
 + (RAWDT(IRSBJ,ISBJ),ISBJ=1,10)
 + WRITE(40,('10F10.4'))
 + (RAWDT(IRSBJ,ISBJ),ISBJ=11,20)
 + WRITE(40,('10F10.4'))
 + (RAWDT(IRSBJ,ISBJ),ISBJ=21,30)
 + WRITE(40,('10F10.4'))
 + (RAWDT(IRSBJ,ISBJ),ISBJ=31,40)
 IRSBJ=IRSBJ+1
 ENDDO
END IF

EDIT RAW DATA !!!! NOTE, THIS IS PERMANENT

WRITE(1,('DO YOU WANT TO EDIT RAW DATA? Y/N'))
READ(1,('A1')) YESNO
IF (YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN
 EDITDONE=.FALSE.
 DO WHILE (.NOT.EDITDONE)
 WRITE(1,('ENTER RUN NUMBER TO EDIT'))
 READ(1,*) RUNNUM
 IRSBJ=1
 DONE=.FALSE.
 DO WHILE(.NOT.DONE.AND.IRSBJ.LE.500)
 IF (RAWDT(IRSBJ,1).EQ.RUNNUM) THEN
 DONE=.TRUE.
 WRITE(1,('8F10.4, " "'))
 + (RAWDT(IRSBJ,ISBJ),ISBJ=1,8)
 WRITE(1,('8F10.4, " "'))
 + (RAWDT(IRSBJ,ISBJ),ISBJ=9,16)
 WRITE(1,('8F10.4, " "'))
 + (RAWDT(IRSBJ,ISBJ),ISBJ=17,24)
 WRITE(1,('8F10.4, " "'))
 + (RAWDT(IRSBJ,ISBJ),ISBJ=25,32)
 WRITE(1,('8F10.4, " "'))
 + (RAWDT(IRSBJ,ISBJ),ISBJ=33,40)
 WRITE(1,('IS THIS IT? Y/N'))
 READ(1,('A1')) YESNO
 IF (YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN
 WRITE(1,('WHICH ELEMENT? 2 TO 40'))
 READ(1,*) ELEMNT

```

      IF (ELEMNT.GE.2.AND.ELEMNT.LE.40) THEN
        WRITE(1, '("ELEMENT TO CHANGE=", F10.4)')
        RAWDT(IRSBJ, ELEMNT)
        WRITE(1, '("IS THIS THE ELEMENT? Y/N")')
        READ(1, '(A1)') YESNO
        IF (YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN
          WRITE(1, '("ENTER NEW VALUE")')
          READ(1, *) NEWVALUE
          WRITE(1, '("ORIGINAL=", F10.4, " NEW VALUE=", F10.4)')
          RAWDT(IRSBJ, ELEMNT), NEWVALUE
          RAWDT(IRSBJ, ELEMNT)=NEWVALUE
        ENDIF
      ELSE
        WRITE(1, '("INVALID ELEMENT NUMBER=", I6)') ELEMNT
      ENDIF
    ENDIF
  ENDIF
  IRSBJ=IRSBJ+1
END DO
WRITE(1, '("MORE EDITING? Y/N")')
READ(1, '(A1)') YESNO
IF (YESNO.EQ.'N'.OR.YESNO.EQ.'n') THEN
  WRITE(1, '("WRITING MODIFIED DATA TO DISK")')
  OPEN(99, FILE=FILNAM, IOSTAT=IER, ERR=9000, ACCESS='DIRECT',
    STATUS='OLD', RECL=100, MAXREC=2000)
C
C DATA STORAGE TO DISK
C
  DO IREC=1,500
    WRITE(1, *) IREC
    IRSBJ=IREC+4-3
    WRITE(99, '(10F10.4)', IOSTAT=IER, ERR=9000, REC=IRSBJ)
      (RAWDT(IREC, ISBJ), ISBJ=1,10)
    WRITE(99, '(10F10.4)', IOSTAT=IER, ERR=9000, REC=IRSBJ+1)
      (RAWDT(IREC, ISBJ), ISBJ=11,20)
    WRITE(99, '(10F10.4)', IOSTAT=IER, ERR=9000, REC=IRSBJ+2)
      (RAWDT(IREC, ISBJ), ISBJ=21,30)
    WRITE(99, '(10F10.4)', IOSTAT=IER, ERR=9000, REC=IRSBJ+3)
      (RAWDT(IREC, ISBJ), ISBJ=31,40)
  END DO
  CLOSE(99)
  EDITDONE=.TRUE.
ENDIF
END DO
ENDIF
RETURN
C
C ERROR HANDLING
C
9000 WRITE(1, '("I/O ERROR #", I3, " IN WRITING OF ", A6, "!!!")') IER, FILNAM
      RETURN
      END
C*****
C
C Subroutine AVERAG to average data for each ID number
C
C*****
$EMA /BIGARR/
SUBROUTINE AVERAG
IMPLICIT NONE
C
  INTEGER DATLOC, AVGCNT, ELEMNT, ID
  REAL RAWDT, AVGDT, OLDDID, NEWID
  LOGICAL DONE
  CHARACTER YESNO*1
C
  COMMON /BIGARR/ RAWDT(500,40), AVGDT(100,40)
C
C Set up variables

```

```

C      OLDID=0.0
      DATLOC=1
      DONE=.FALSE.
C
C      Do until ID numbers run out
C
DO WHILE(.NOT.DONE)
  NEWID=RAWDT(DATLOC,2)      ! get new ID #
  WRITE(1,*) DATLOC,RAWDT(DATLOC,2)
  IF(NEWID.NE.9999) THEN
C
C      Running average
C
    IF(NEWID.NE.OLDID) THEN
      AVGCNT=1
    ELSE
      AVGCNT=AVGCNT+1
    ENDIF
    OLDID=NEWID
    DO ELEMENT=1,40
      AVGDT(OLDID,ELEMENT)=AVGDT(OLDID,ELEMENT)*(1.0-1.0/AVGCNT)
      +      +RAWDT(DATLOC,ELEMENT)*(1.0/AVGCNT)
    ENDDO
    ELSE
      DONE=.TRUE.
    ENDIF
    DATLOC=DATLOC+1      ! increment data location counter
  ENDDO
C
C      Option to print
C
  WRITE(1,('Do you want a printout of the averaged data?'))
  READ(1,('A1')) YESNO
  IF(YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN
10    WRITE(40,10)
      FORMAT(///35X,"SDM PLUNGING OSCILLATION DERIVATIVES(AVERAGED"
      +      " DATA)"/35X,"ID      ALPHA      BETA      CMADOT      CMA      "
      +      "CNADOT      CNA      CLADOT      CLA      CZADOT      CZA      "
      +      "FREQ      AMPL"/)
      DO ID=1,100
        IF(NINT(AVGDT(ID,2)).EQ.ID) THEN
          WRITE(40,20) AVGDT(ID,2),AVGDT(ID,3),AVGDT(ID,4),
          +      AVGDT(ID,7),AVGDT(ID,8),AVGDT(ID,9),AVGDT(ID,10),
          +      AVGDT(ID,11),AVGDT(ID,12),AVGDT(ID,15),
          +      AVGDT(ID,16),AVGDT(ID,17),AVGDT(ID,20)
20        FORMAT(6X,F5.0,2X,F6.2,3X,F5.2,3X,F6.3,3X,F5.3,3X,F6.3,
          +      3X,F5.3,3X,F6.3,3X,F5.3,3X,F5.3,3X,F5.3,2X,F5.3,
          +      3X,F5.3)
        ENDIF
      ENDDO
    ENDIF
    RETURN
  END
C*****
C
C      SUBROUTINE STATIC TO ENTER STATIC DATA FOR EACH ID NUMBER
C
C*****
$EMA /BIGARR/
SUBROUTINE STATIC
IMPLICIT NONE
CHARACTER INPUT*1,YESNO*1
C
  INTEGER ID,I,IER
  REAL VZ,VM,VNF,VNA,VL,GZ,GM,GNE,GNA,GL,ALPHAC,BETAC
  REAL CM,CL,CN,CZ,CY,CLADT,AVGDT,TEMPOR(8),ZERO
  LOGICAL DONE,FIRST

```



```

C      COMMON/BIGARR/RAWDT(500,40),AVGDT(100,40)
C
C      Open file
C
C      OPEN(99,FILE='NASSSP::300',IOSTAT=IER,ERR=1000,ACCESS='DIRECT',
+      STATUS='OLD',RECL=100,MAXREC=100)
C      GOTO 2000
1000  CLOSE(99)
C
C      Create file if it doesnt exist and fill with zeros
C
C      OPEN(99,FILE='NASSSP::300',ACCESS='DIRECT',STATUS='NEW',RECL=100,
+      MAXREC=100)
C      ZERO=0.0
C      DO ID=1,100
C          WRITE(99,'(8(F10.4,2X))',REC=ID) (ZERO,I=1,8)
C      ENDDO
C
C      Read static data file
C
C      2000  DO ID=1,100
C          READ(99,'(8(F10.4,2X))',REC=ID) (TEMPOR(I),I=1,8)
C          IF(TEMPOR(1).EQ.ID) THEN
C              DO I=2,8
C                  AVGDT(ID,28+I)=TEMPOR(I)
C              ENDDO
C          ENDIF
C      ENDDO
C      CLOSE(99)
C
C      FIRST=.TRUE.
C      DONE=.FALSE.
C      WRITE(1,'("DO YOU WISH TO ENTER STATIC DATA?")')
C      READ(1,'(A1)')INPUT
C      IF (INPUT.EQ.'Y'.OR.INPUT.EQ.'y')THEN
C          DO WHILE(.NOT.DONE)
C              WRITE(1,'("ENTER ID NUMBER,9999 TO STOP")')
C              READ(1,*)ID
C              IF (ID.NE.9999)THEN
C                  WRITE(1,'("ENTER VOLTAGES(V):CANTILEVER,PITCH,YAWF,YAWA,"
+                  "ROLL")')
C                  READ(1,*)VZ,VM,VNF,VNA,VL
C                  IF(.NOT.FIRST) THEN
C                      WRITE(1,'("Do you wish to enter new gains?")')
C                      READ(1,'(A1)') YESNO
C                  ELSE
C                      YESNO='Y'
C                      FIRST=.FALSE.
C                  ENDIF
C                  IF(YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN
C                      WRITE(1,'("ENTER GAINS:CANTILEV,PITCH,YAWF,YAWA,ROLL")')
C                      READ(1,*)GZ,GM,GNF,GNA,GL
C                  ENDIF
C                  CALL CALSTD(VZ,VM,VNF,VNA,VL,GZ,GM,GNF,GNA,GL,
+                  ALPHAC,BETAC,CM,CL,CN,CZ,CY,ID)
C                  AVGDT(ID,28)=ALPHAC
C                  AVGDT(ID,29)=BETAC
C                  AVGDT(ID,30)=CM
C                  AVGDT(ID,31)=CN
C                  AVGDT(ID,32)=CL
C                  AVGDT(ID,33)=CZ
C                  AVGDT(ID,34)=CY
C              ELSE
C                  DONE=.TRUE.
C              END IF
C          END DO
C      END IF

```

```

WRITE(1,('Do you wish to print static data?'))
READ(1, '(A1)') YESNO
IF(YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN
  WRITE(40,30)
  30  FORMAT(///9X,"TABLE ? - STATIC AERODYNAMIC COEFFICIENTS"
+      " (PLUNGING OSCILLATION)"///6X,"ALPHA  BETA  CM"
+      "      CN      CL      CZ      CY      MACH  RE"/)
  DO ID=1,100
    IF(NINT(AVGDT(ID,2)).EQ.ID) THEN
      WRITE(40,40) AVGDT(ID,3),AVGDT(ID,4),AVGDT(ID,30),
+      AVGDT(ID,31),AVGDT(ID,32),AVGDT(ID,33),
+      AVGDT(ID,34),AVGDT(ID,21),AVGDT(ID,22)
  40  FORMAT(4X,F6.2,3X,F5.2,3X,F6.4,2X,F6.4,2X,F6.4,3X,
+      F6.4,3X,F7.4,3X,F5.3,3X,F5.3)
    ENDIF
  ENDDO
ENDIF
C
C  Open file and write current static data
C
OPEN(99,FILE='NASSSP::300',ACCESS='DIRECT',RECL=100,MAXREC=100)
DO ID=1,100
  IF(NINT(AVGDT(ID,2)).EQ.ID) THEN
    WRITE(99,'(8(F10.4,2X))',REC=ID) ID,(AVGDT(ID,I),I=28,34)
  ENDIF
ENDDO
CLOSE(99)
C
C  Open file and write to plot file
C
OPEN(99,FILE='AVGSTP::300')
DO ID=1,100
  IF(NINT(AVGDT(ID,2)).EQ.ID) THEN
    WRITE(99,'(8(F10.4,2X))',ID,(AVGDT(ID,I),I=28,34)
  ENDIF
ENDDO
CLOSE(99)
C
C  Save to disc(plot file)
C
OPEN(99,FILE='AVGDYP::300')
DO ID=1,100
  IF(NINT(AVGDT(ID,2)).EQ.ID) THEN
    WRITE(99,25) AVGDT(ID,2),AVGDT(ID,28),AVGDT(ID,29),
+      AVGDT(ID,7),AVGDT(ID,8),AVGDT(ID,9),AVGDT(ID,10),
+      AVGDT(ID,11),AVGDT(ID,12),AVGDT(ID,15),
+      AVGDT(ID,16),AVGDT(ID,17),AVGDT(ID,20)
  25  FORMAT(6X,F5.0,2X,F6.2,3X,F5.2,3X,F6.3,3X,F5.3,3X,F6.3,
+      3X,F5.3,3X,F6.3,3X,F5.3,3X,F5.3,3X,F5.3,2X,F5.3,
+      3X,F5.3)
    ENDIF
  ENDDO
CLOSE(99)
RETURN
END
C*****
C
C  SUBROUTINE CALSTD to calculate static coefficients and correct
C  equilibrium angles ALPHA,BETA for static deflections
C
C*****
$EMA /BIGARR/
SUBROUTINE CALSTD(VZ,VM,VNF,VNA,VL,GZ,GM,GNF,GNA,GL,
+      ALPHAC,BETAC,CM,CL,CN,CZ,CY,ID)
  IMPLICIT NONE
C
  INTEGER ID
  REAL SM,SNF,SNA,SL,SZ,DF,DA,DR,CSNFL,CSNAL,CSLY,CSNFZ,KM,KN,M,L

```

REAL NF,NA,Y,D,N,Z,CORM,CORNF,CORNA,CORL,VMC,VNFC,VNAC,VLC,ACOR
REAL BCOR,ALPHAC,BETAC,S,B,CBAR,CNST,CM,CL,CN,CZ,CY,RHO,RAWDT
REAL AVGDT,VZ,VM,VNF,VNA,VL,GZ,GM,GNF,GNA,GL,RTD

COMMON/BIGARR/RAWDT(500,40),AVGDT(100,40)

Fix sensitivities,electrical centers,cross-sensitivities and
stiffnesses obtained from static calibration

SM=4.51E-2
SNF=3.25E-2
SNA=3.49E-2
SL=2.17E-2
SZ=4.11E-5
DF=0.04755
DA=0.01572
DR=0.0279
CSNFL=-3.41E-4
CSNAL=7.15E-4
CSLY=-0.196E-3
CSNFZ=0.353E-3
KM=1580
KN=880

Calculate static moments,forces and relevant distances
uncorrected for cross-sensitivities(for each ID)

M=VM/(SM*GM)
L=VL/(SL*GL)
NF=VNF/(SNF*GNF)
NA=VNA/(SNA*GNA)
Y=(NA-NF)/(DF-DA)
D=((NA*DF)-(NF*DA))/(NA-NF)
N=Y*D
Z=VZ/(SZ*GZ)

Correct voltages for cross-sensitivities of flexure(for each ID)
Determine voltage corrections

CORM=0.0
CORNF=(CSNFL*L)+(CSNFZ*Z*(M/Z-DF))
CORNA=(CSNAL*L)
CORL=CSLY*Y*(D-DR)

Apply voltage corrections

VMC=VM-CORM
VNFC=VNF-CORNF
VNAC=VNA-CORNA
VLC=VL-CORL

Recompute static moments and forces(for each ID number)

M=VMC/(SM*GM)
L=VLC/(SL*GL)
NF=VNFC/(SNF*GNF)
NA=VNAC/(SNA*GNA)
Y=(NA-NF)/(DF-DA)
D=((NA*DF)-(NF*DA))/(NA-NF)
N=Y*D

Compute ALPHA,BETA corrections(for each ID)

RTD=67.2958
BCOR=(M/KM)*RTD
BCOR=-((-5.8E-4*Y)/RTD + N/KN)*RTD
ALPHAC=AVGDT(ID,3)+ACOR
BETAC=AVGDT(ID,4)+BCOR

C
C
C

Nondimensionalize moments and forces(for each ID)

S=174.12E-04
B=0.2286
CBAR=0.0863
RHO=2.0*AVGDT(ID,24)/(AVGDT(ID,23)**2)
CNST=0.5*RHO*(AVGDT(ID,23)**2)*S
CM=M/(CNST*CBAR)
CL=L/(CNST*B)
CN=N/(CNST*B)
CZ=Z/CNST
CY=Y/CNST
RETURN
END

END

FIN

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