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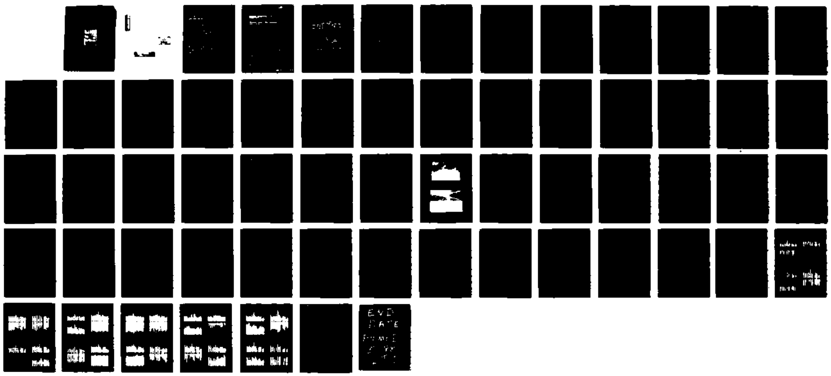

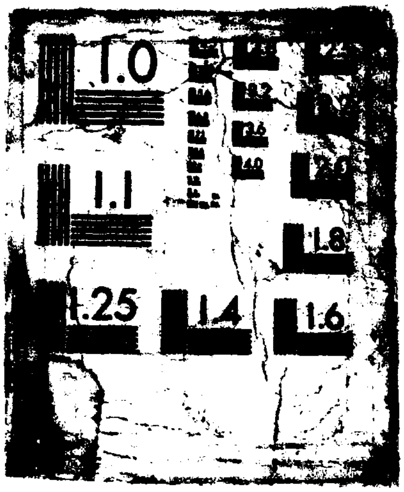
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THE APPLICATION OF LINEAR MAXIMUM LIKELIHOOD ESTIMATION OF AERODYNAMIC DERIVATIVES FOR THE BELL-205 AND BELL-206

APPLICATION DE LA MÉTHODE DU MEILLEUR ESTIMATEUR LINÉAIRE POUR LE CALCUL DES DÉRIVÉES AÉRODYNAMIQUES DU BELL-205 ET DU BELL-206

by/par

J.H. deLeeuw
Institute for Aerospace Studies, University of Toronto

and/et

K. Hui
National Aeronautical Establishment

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S.R.M. Sinclair, Head/Chef
Flight Research Laboratory/
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G.F. Marsters
Director/Directeur

ABSTRACT

Parameter identification from flight test data of fixed-wing aircraft is currently a common procedure for application to aircraft development work, validation of simulation, flight simulator verification, flight control systems synthesis, aircraft handling qualities, flight envelope expansion and airplane certification. Similar work on the identification of the more complex helicopter system is currently still in the research stage. This report describes a number of flight test experiments involving the application of parameter estimation techniques to helicopters in order to determine the stability and control derivatives and to obtain information to identify improvements in the structure of the helicopter model.

RÉSUMÉ

L'identification des paramètres à partir des données d'essai en vol est une méthode couramment appliquée à la mise au point des avions, à la validation des simulations, à la vérification des simulateurs de vol, à la synthèse des systèmes de commande de vol, à la maniabilité des avions, à l'extension de l'enveloppe de vol et à l'homologation des avions. Des travaux semblables portant sur le cas plus complexe des hélicoptères sont actuellement au stade de la recherche.

Ce rapport traite de l'usage des techniques d'estimation de paramètre dans le but de déterminer les dérivés de stabilité et d'obtenir de meilleurs modèles mathématiques pour la description des hélicoptères en tant que système. Dans ce contexte, les données de plusieurs essais en vol sont présentées.

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NOMENCLATURE

a_x, a_y, a_z	longitudinal, lateral, and vertical accelerometer output
F_x, F_y, F_z	aerodynamic forces
$f(.)$	system state function
GG^*	measurement noise covariance
$g(.)$	system observation function
I_x, I_y, I_z	moments of inertia
I_{xy}, I_{xz}, I_{yz}	cross products of inertia
$J(.)$	cost function
m	mass of aircraft
L, M, N	aerodynamic rolling, pitching and yawing moments respectively
L', N'	decoupled aerodynamic rolling and yawing moments respectively
p	roll rate
q	pitch rate
r	yaw rate
u	body x-axis small perturbation velocity
v	body y-axis small perturbation velocity
w	body z-axis small perturbation velocity
x	system state
X	x-axis acceleration
Y	y-axis acceleration
Z	z-axis acceleration
ϕ	roll angle
θ	pitch angle
ψ	yaw angle
δ_a	lateral roll cyclic
δ_e	longitudinal pitch cyclic
δ_r	lateral yaw input
δ_p	longitudinal heave collective

1.0 INTRODUCTION

The field of aircraft stability and control exemplifies many successful applications of system identification technology. For fixed wing aircraft, these techniques to determine stability and control derivatives are used frequently and with confidence. The application of the same techniques to helicopters is not as far advanced because the complex helicopter is more difficult to describe with relatively simple mathematical models and flight test measurements exhibit high levels of vibration noise originating from the rotor.

For the work at the Flight Research Laboratory (FRL), a maximum likelihood method based on a linear aircraft model was chosen to serve as the initial algorithm to analyze flight test data obtained with Bell-205 and Bell-206 helicopters (see Figures 1 and 2). The first applications of the maximum likelihood estimators were based on simplified 3 degree of freedom models for uncoupled longitudinal and lateral modes. This work was similar to the earlier work at FRL by Gould and Hindson,[1] who performed flight tests on the Bell-205 with separate longitudinal and lateral control inputs. They employed uncoupled models in which the stability and control derivatives were determined.

The two most common techniques used to analyze the rigid body stability and control derivatives are the so-called output error and equation error methods. In the former, the coefficients of the adopted model are adjusted to get a best fit for the measured variables. This method will result in coefficient values that are unbiased if only measurement noise is present. The alternative equation error method, provides estimates for the coefficients by a regression analysis of the equations formulating the aerodynamic forces and moments in terms of the aircraft states and control inputs. In these equations, the state variables have to be estimated from measurements (possibly assisted by the use of flight path reconstruction techniques) and will, therefore, exhibit measurement errors. The coefficients obtained in this very computation-efficient method will be biased in the presence of this measurement noise even if no other noise sources are present.

2.0 PARAMETER ESTIMATION METHOD

Any parameter estimation method requires first the selection of a mathematical model describing the evolution of the state of the system. A judgement of which state variables are significant in a particular application has to be made. For conventional fixed-wing aircraft a six degree of freedom (6DOF) model, involving only the rigid body states, u, v, w, p, q and r, and even the simpler uncoupled longitudinal and lateral subsystems, has been remarkably successful. For dynamically more complex aircraft, including helicopters, it is anticipated that additional states may be required to provide a satisfactory description. Examples are the states describing the flexible modes of the wings and those representing the dynamics of the rotor system. Fortunately, in many cases of practical interest, the natural frequencies associated with these additional states are considerably higher than those from the rigid body modes such that, for sufficiently gradual control applications, these additional states behave in a quasi-static manner and a model based on the rigid body states can still give useful results.

For the work described in this report, the model adopted for the helicopter is a linear, fully coupled six degree of freedom rigid body system. The estimation of the parameter values that occur in this model, i.e. the stability and control derivatives, is done on the basis of flight tests in which the helicopter is excited by suitable control inputs. The parameter values are found by algorithms which minimize the differences between the calculated response of the model equations and those of the measurements. The major algorithm used for this purpose was the maximum likelihood method as embodied in the modified maximum likelihood estimator; version 3 (MMLE3)[2,3] program developed by NASA. In addition, regression calculations were performed, partly to obtain reasonable starting guesses for the iterative maximum likelihood process and partly to serve as alternative estimates for the coefficients for comparison with the maximum likelihood estimator (MLE) results.

2.1 Description of MMLE3

The aircraft is a dynamic system continuous in time while the measurements are made at discrete time intervals for analysis on a digital computer. The formulation of the model when process noise is absent in MMLE3 (see Figure 3 for overall flowchart) terminology is defined by:

$$x(t_o) = x_o(\xi) \tag{1a}$$

$$\dot{x}(t) = f[x(t)\ ,\ u(t)\ ,\ \xi\ ,\ t\ ] \tag{1b}$$

$$z(t_i) = g[x(t_i),\ u(t_i),\ t_i,\ \xi_i]+\eta(t_i) \tag{1c}$$

where
- x state vector
- u input vector
- z measured observation vector

η measurement noise vector, a sequence of independent Gaussian vectors with zero mean and covariance GG*

ξ vector of unknown parameters

The MMLE3 program does contain a Kalman filter option which can be invoked to reduce the effects of process noise, such as the effects of gusts. For most of our flight tests rigorously gust free conditions were selected to avoid the complications of random gust inputs. This means that the process "noise" consists primarily of residual modelling errors and rotor vibrations that are assumed to be small, and the Kalman filter option was not used.

The forms of the f and g functions were postulated to be linear. The parameter estimation process then consists of the determination of the values of the unknown elements of the parameter vector ξ. In our linear six degree of freedom model, ξ contains the stability and control derivatives but also contains a number of unknown offsets representing, for instance, instrument biases.

The cost function is defined by

$$J(\xi) = \frac{1}{2} \sum_{i=1}^{N} \left[z(t_i) - \tilde{z}_\xi(t_i)\right]^* (GG^*)^{-1} \left[z(t_i) - \tilde{z}_\xi(t_i)\right]$$

where

$\tilde{z}_\xi$ is the computed response associated with the specific values of ξ

GG* is the weighting function which theoretically should be the covariance matrix of the measurement noise

The value of ξ that minimizes the cost funcion is determined iteratively by a Newton-Raphson algorithm. This method also generates estimated information about the accuracy of the estimates of the individual parameters. The MMLE3 program provides this information in the form of the Cramer-Rao bounds as well as the correlation coefficients of particular parameter pairs. If our repeats of a particular experiment are available, then the scatter in the parameter values estimated for each experiment can be used as an explicit indication of the quality of the estimation process.

If only measurement error is present, i.e., if the model is perfect, then the covariance matrix of the noise should be expressible in terms of the residual error between the predictions of the best model and the measurements:

$$GG^* = \frac{1}{N}\sum_{i=1}^{N} [\tilde{z}_{\xi}(t_i) - z(t_i)][\tilde{z}_{\xi}(t_i) - z(t_i)]^* \qquad (2)$$

To determine G as well as the other unknowns the following two step algorithm is available in MMLE3: 1) one iteration of the Gauss-Newton algorithm is used to improve all the unknowns of the ξ vector for a fixed value of G obtained in the previous step; and 2) Equation (2) is used with $\tilde{z}_{\xi}$ evaluated at the revised value of ξ to obtain a new estimate of G. Then steps (1) and (2) are repeated until convergence is obtained. It should be noted that doing this with flight test data has been found to result in unrealistic GG*'s because of modelling errors and the possible presence of correlated noise. Consequently, in practice, a diagonal representation of GG* was used with values adjusted to result in fits of roughly comparable quality for each of the elements of the observation vector.

2.2 Equations of Motion

It is customary in the analysis of the dynamics of helicopters, to describe the system equations in a reference frame fixed to the fuselage body. In this frame the velocity components are u, v and w and the angular rates are given by p, q and r. Since gravity makes a contribution to the total force acting on the body, through the orientation of the body frame with respect to earth-fixed axes, the attitude angles θ, ϕ and ψ are also of interest.

A further simplification is appropriate because mirror symmetry of the helicopter can be assumed with respect to the x-y plane. Consequently $I_{xy} = I_{yz} = 0$. The equations then take the familiar form (see Etkin's "Dynamics of Flight")

$$F_x - mg \sin\theta = m(\dot{u} + qw - rv)$$
$$F_y + mg \cos\theta \sin\phi = m(\dot{v} + ru - pw)$$
$$F_z + mg \cos\theta \cos\phi = m(\dot{w} + pv - qu)$$
$$\tag{3}$$
$$L = I_x\dot{p} - I_{xz}\dot{r} + qr(I_x - I_y) - I_{xy}pq$$
$$M = I_y\dot{q} + rp(I_x - I_z) + (p^2 - r^2)I_x$$
$$N = -I_{xz}\dot{p} + I_z\dot{r} + pq(I_y - I_x) - rpI_{xz}$$

They are supplemented with relations between the orientation angles (Euler angles) and the angular rates in the body frame of reference:

$$p = \dot{\phi} - \dot{\psi} \sin\theta$$
$$q = \dot{\theta} \cos\phi + \dot{\psi} \cos\theta \sin\phi \tag{4}$$
$$r = \dot{\psi} \cos\theta \cos\phi - \dot{\theta} \sin\phi$$

or, alternatively

$$\dot{\phi} = p + q \tan\theta \sin\phi + r \tan\theta \cos\phi$$
$$\dot{\theta} = q \cos\phi - r \sin\phi$$
$$\dot{\psi} = (q \sin\phi + r \cos\phi) \sec\theta.$$

It is seen that the equations for L and N are cross coupled, even when all second order terms are neglected. Further simplifications are often made in the equations for helicopter dynamics by introducing modified quantities L' and N', as follows:

$$L' = \frac{1}{I_x} \frac{L + \frac{I_{xz}}{I_x} N}{1 - \frac{I_{xz}^2}{I_x I_z}}$$

$$N' = \frac{1}{I_z} \frac{N + \frac{I_{xz}}{I_z} L}{1 - \frac{I_{xz}^2}{I_x I_z}} \tag{6}$$

or

$$L = I_x L' - \frac{I_{xz} I_z}{I_x} N'$$

$$N = I_z N' - \frac{I_{xz} I_x}{I_z} L' \tag{7}$$

When the second order terms in the L and N equations are ignored this modification results in uncoupled equations for L' and N'. When we introduce the notation

$$X = \frac{F_x}{m} \qquad Y = \frac{F_y}{m} \qquad Z = \frac{F_z}{m} \tag{8}$$

then the equations take the following form

$$\begin{aligned}
X - g \sin\theta &= \dot{u} + qw - rv \\
Y + g \sin\phi \cos\theta &= \dot{v} + ru - pw \\
Z + g \cos\phi \cos\theta &= \dot{w} + pv - qu \\
L' &= \dot{p} \\
M' &= \dot{q} \\
N' &= \dot{r} \\
\dot{\phi} &= p + q \tan\theta \sin\phi + r \cos\phi \tan\theta \\
\dot{\theta} &= q \cos\phi - r \sin\phi.
\end{aligned} \tag{9}$$

It is worth repeating that the stated equations for L', M' and N' are only valid when second order products of angular rates can be neglected. The other equations, however, are not in a linearized form. In fact, the MMLE3 program is structured to make it easy to calculate the terms as given in the equations above from measured quantities; and in the analysis in this project this was done since the flight manoeuvres were kept to reasonable magnitudes. It is not expected that serious non-linear effects in the kinematic equations were present, so that this inconsistency in the equations is not of practical importance.

3.0 BACKGROUND

For the first phase of the work described in this report, uncoupled longitudinal and lateral analyses were made of the experiments. This constituted a simple first look at the problem of parameter estimation and followed the approach in earlier work at the Flight Research Laboratory by Gould and Hindson. They treated the longitudinal and lateral cases separately. Also, their basic independent unknown parameters were not the conventional stability and control derivatives, but a more fundamental set of parameters where use was made of estimated geometric interrelationships between the various derivatives.

In the current program, this uncoupled model was applied in three experiments namely: 1) Bell-205 flights covering a carpet of flight conditions (See Figure 4); and 2) Bell-206 flights for a carpet of flight conditions (See Figure 5).

3.1 Bell-205 Flights

The purpose of the first experiments was to validate the instrumentation and the format of data flow by checking the self consistency of the measurements, and to test the NASA MMLE3 with uncoupled three degree of freedom (3DOF) linear models on flight test data.

3.2 Bell-206 Flights

Since most aircraft do not have computer commanded control inputs, as were available for the Bell 205 experiments, a set of flight tests with manual control inputs was performed with the Bell 206 over a wide range of flight conditions. The preferred control excitation was a sequence of constant amplitude pulses of varying length and alternating sign (specifically, 3-2-1-1, where the numbers represent the pulse lengths in seconds). Although the flying conditions were not perfectly smooth in all flights, a reasonable data base with replications of each manoeuvre was obtained from which stability derivatives were obtained. Also, from these experiments the internal consistency of the results can be determined when the manual control inputs were inherently variable between tests.

4.0 DESCRIPTION OF EXPERIMENTAL EQUIPMENT

4.1 Bell-205 System

The Bell-205 used for these experiments is the airborne simulator at the Flight Research Laboratory of the National Research Council Canada. It is well instrumented and possesses a complete data acquisition system.

4.1.1 Control inputs

The control inputs were programmed by the on-board computer and then used to drive the hydraulic control actuators to excite the aircraft in a well-controlled, repeatable manner. The commanded inputs were measured by linear variable digital transducer (LVDT) sensors at the actuators and provided explicitly as information on cyclic, longitudinal, lateral and tail rotor control inputs.

4.1.2 Anti-aliasing filters

All raw data were filtered by anti-aliasing filters with a cutoff frequency of 10 Hz and critical damping of 0.7, as represented by the transfer function,

$$Z_B(S) = \frac{(62.83)^2}{S^2 + 1.4 * 62.83S + (62.83)^2} \tag{16}$$

4.1.3 Notch filter

Also, a notch filter at 10.8 Hz (blade passage frequency) with the transfer function

$$Z_N(S) = \frac{S^2 + (67.86)^2}{S^2 + 0.4 * 67.87S + (67.86)^2} \tag{17}$$

was used on the accelerometer and angular rate channels to minimize the effect of vibration at the blade frequency.

4.1.4 Sample rate

The data was collected at a sampling rate of 128 Hz. The reference outputs of the accelerometers were reset to zero during the trim condition before each experiment.

4.1.5 Complementary filter

Values for u, v, w were calculated by a complementary mixing filter, which combined low frequency information from the air data system with high frequency information from the accelerometers.

4.1.6 Instrument location

The accelerometers were not located exactly at the center of gravity of the helicopter but had an estimated vertical offset of about 17.5 inches to 22.5 inches below the C. of G.

4.1.7 Data transfer

The data was recorded on a streamer tape which can hold up to 50 megabytes. This streamer tape data was transcribed to a 9-track tape at a density of 800 bpi in IBM compatible format and this tape was used to transfer the raw data to the IBM main frame computer for analysis.

4.2 Bell-206 System

The Bell-206 does not have a built-in instrumentation system like that of the Bell-205. A flight test instrumentation package from the University of Toronto Institute for Aerospace Studies (UTIAS) was used in the experiments.

4.2.1 UTIAS package

This package consists of three orthogonal linear accelerometers, three orthogonal angular rate gyros and a vertical gyro to measure pitch and bank angles. It also provides two measurements related to static pressure: One indicates the absolute pressure while the other offers a higher resolution measurement of the changes in static pressure during a manoeuvre by a differential measurement with reference to a constant pressure reservoir. Dynamic pressure is also measured.

4.2.2 Anti-aliasing filter and sample rate

Butterworth anti-aliasing filters are used in the package, to precondition the signals for a sampling rate of 40 Hz.

4.2.3 Data transfer

The flight data are recorded on special tape cassettes. Special provisions were made on a PDP-11 computer to transcribe this information to standard 9-track tapes for transfer to the IBM mainframe computer.

4.2.4 Flight path reconstruction

As a first step in the data analysis, a six degree of freedom non-linear compatibility check (flight path reconstruction) technique was used to eliminate any instrumental offsets and to calculate u, v, and w.

5.0 RESULTS WITH THE 3DOF MODEL

5.1 Bell-205 Initial Flights

5.1.1 Methodology of analysis

A multistep control input was used for separate excitation of pitch, heave, roll and yaw (4 axes) over a speed range of 40 to 90 kts at intervals of 10 kts. The steps were separated by 1-1-2-3 seconds and the control input alternated between opposite levels of equal magnitude. At each speed an experiment was performed in level flight, followed by experiments in a fast climb, in a moderate climb, in a moderate descent and a fast descent. Each sequence was closed off by another level flight experiment. Climb and descent rates of ±500 ft/min, and ± 1000 ft/min were used. A three degree of freedom, decoupled longitudinal or lateral mathematical model was used to analyze the data from each manoeuvre, separately. The fits obtained were generally very good but the parameter values obtained from roll and yaw experiments did not match well. A similar conclusion was reached from a comparison of heave and pitch experiments.

5.1.2 Conclusions

From the analysis of the experiments for the carpet of flight conditions, the following observations can be made:

1) Good starting guesses for the initial values of the derivatives were required to attain convergence and self-consistent results. Even so, a minor proportion of the yaw and heave experiments did not lead to converged results;
2) The weighting factor in the cost function had to be adjusted to balance the fits for each of the major measured variables on the basis of subjective judgment of the time history plots;
3) Strong correlation between roll and yaw derivatives was indicated by the MMLE analysis;
4) Different values were obtained for the parameter values estimated from experiments using roll excitation as compared to those from tests with yaw excitation. Similar differences were noted for the parameters determined from tests with pitch and heave excitation. It may therefore be necessary to analyze manoeuvres with different control inputs together and a 6DOF model may be required as well.
5) The data were collected under mild to fair turbulent conditions and there were no repeat experiments. To obtain a cursory impression of the self-consistency of the estimated parameters, a statistical analysis was made using the additional cases of climbing and descending flight conditions.

5.2 Bell-206 Flights

5.2.1 Description of Bell-206 flights

Manual control inputs were used with a (3-2-1-1 sec) multistep single axis input. Flight conditions covered an extensive carpet of forward speeds and vertical velocities. Many replications of each experiment were performed, in order to be able to average out any residual effects of turbulence and to assess the differences between the results from individual control inputs which vary from case to case with manual control inputs. Also, an explicit quantitative indication of repeatability is obtained this way. In the data analysis, all raw data were passed through the compatibility check program (flight path reconstruction) to eliminate any instrumental offsets and to obtain u, v, and w. As was done in the analysis of the Bell-205 experiments, the MMLE procedures were used to interpret the results for single axis control excitation.

5.2.2 Observations

On the basis of the analysis of the entire set of experiments for the Bell 206 the following observations can be made:

1) Sometimes one experiment converged but a direct repeat did not. It is believed that inter-axis coupling effects which are not modelled may have varied as a result of differences in the manual control inputs.
2) As was found to be the case in the Bell-205 experiments, good initial values for the derivatives were needed to obtain convergence.
3) As for the Bell 205, the weighting factors in the cost function had to be adjusted to get appropriately balanced fits for each of the major measured variables.

4) In general, better convergence was observed than in the case of the Bell-205 experiments. This could be due to smoother flying conditions or smaller cross-coupling effects.

5) Because of the need for interactive changes in initial values and weighting factors in the analysis procedures to produce convergence, the total effort in man hours and computing time to obtain the results of the Bell-205 and Bell-206 flights was considerable.

Note: The initial stability and control derivatives for the MMLE3 program were obtained either from results of other experiments or from NASA CR 3144[26]. Engineering judgement was required to establish the weighting fucntions and several techniques were employed. For example, in some cases interventions were made following each of the initial few cycles of the MMLE3 process and adjustments made to the weights; in others, the program was allowed to iterate while several derivatives that had strong correlation coefficients were fixed in value. The resulting set of derivatives was then used as the new "initial values" for a second MMLE3 iteration. This step had to be repeated several times before convergence was obtained in a substantial number of cases. Sometimes, despite this effort, no convergence was obtained. Also, appropriate adjustments to the weighting factors were made to assist in the process of obtaining convergent results with reasonable fits for all measured quantities. This very tedious process was characteristic of the three degree of freedom analyses of separate, single control, manoeuvres. Fortunately, when the six degree of freedom model was used with simultaneous analysis of various manoeuvres and starting values for the derivatives obtained from a regression analysis were used, these difficulties were largely overcome.

6.0 EVOLUTION OF FLIGHT TEST PROCEDURES AND 6DOF DATA ANALYSIS

Following the three sets of experiments described in the previous section, a number of modifications were introduced in the data analysis methodology. This involved the adaptation of the MMLE program to handle models with six degrees of freedom, the attention to the subject of control input optimization and a decision to use stepwise regression to obtain comparison values for the derivatives as well as to provide starting values for the parameters in the MMLE analysis.

6.1 Scope of Flight Conditions and Format of Experiments

It was decided to perform flight tests, with the Bell 205, only under turbulence free conditions. An altitude of 3,000 ft was selected to optimize the chances of finding favourable, smooth flying conditions. Each manoeuvre was preceded by a steady trim condition, which was set up with the assistance of the stability augmentation system of the Bell 205. Each manoeuvre started with a 2 sec period of trimmed flight without stability augmentation before the application of the control inputs. Also, the period of control excitation was followed by a substantial "trailer"[5] which allowed the measurement of the aircraft's natural modes, especially the lowest frequency phugoid mode. There were repeats in both directions of each control input. Also, a variety of input shapes were used to study the relative effectiveness of different control inputs.

6.2 Input Optimization

In order to assist in the selection of control input time histories, with suitable frequency content[6], Bode plots (see Figures 6, 7, 8, and 9) for a six degree of freedom model using published values (NASA CR3144) of the stability derivatives were made showing the contribution made by each derivative to the response. It was found that a sinusoidal control input at high enough frequencies will result in an aircraft response that is dominated by the control derivatives. Specifically, a 4 rad/sec sinusoidal input will create a response that is largely determined by the control derivatives and which is only minimally affected by the stability derivatives. Hence, a 4 rad/sec sinusoidal input was selected as one of the control inputs for this flight test.

Following the recommendations of references 5 and 7 on limiting the high frequency content of control inputs to minimize the excitation of the rotor modes, appropriate multistep sequences were selected. A 3-2-1-1 (sec) step was selected as applicable because its frequency content above 5 rad/sec is small. Also, a modified 3-2-1-1 multistep (see Figure 10: Input optimization) was employed in which the average value of control deflection is zero, unlike that of the normal (equal step) 3-2-1-1 sequence. This selection was made in the hope that the extreme values of the states of the aircraft for the modified 3-2-1-1 input would be better balanced and so would permit a larger overall amplitude of the input without causing concern about non-linear effects.

All of the above inputs are generated by the computer commanded control system. However, because in many flight test situations such computer control is not available, a number of cases with manual control inputs were also flown. These manual inputs were of two types: 1) inputs where the pilot attempts multistep inputs similar to the computer commanded inputs and 2) inputs for which the pilot was asked to use his own judgement in providing control motion of longitudinal and lateral cyclic that cause good excitation of the phugoid and dutch roll modes.

6.3 Prefiltering

A second-order digital Butterworth, no-phase-shift filter,[8] with cutoff at 4 Hz was developed to filter all channels in the post-flight analysis in order to use an effective sampling frequency of 16 Hz in the data analysis. Also, a differentiator/smoother[9] filter with a cutoff at 4 Hz (256 terms) was tested and developed to obtain the angular accelerations that were used to transform the measurements at the instrumentation centre to the center of gravity of the aircraft. The cutoff frequency of 4 Hz was chosen because the anticipated fuselage modes have frequencies well under 1 Hz. Any measured input above 1 Hz will be largely noise from sources such as engine/rotor vibration and instrumentation noise etc.

6.4 Stepwise Regression

A stepwise regression analysis was used to provide good starting values for the derivatives required in the MLE procedure. This method is capable of analyzing combined manoeuvres. (For example, a choice can be made to analyze one control excitation at a time, or two, three or all four segments simultaneously). The stepwise nature of the regression method also indicates the relative importance of the various stability derivatives by the order of selection of the regression variables. It also calculates the estimated error in the values of the individual stability derivatives. In addition, information on the F-ratios and the overall error of the regression is provided for each analysis. It is noted that the angular accelerations as provided by the differentiator/smoother are necessary in this regression analysis. When the stepwise regression results are similar to those obtained with the MLE method, confidence that the model gives reasonable results is obviously reinforced.

6.5 **Summary of the Improved Methodology and Test Plan** (see Figure 12)

1) A six degree of freedom linear quasi-static, rigid body mathematical model was used.
2) The Stability Augmentation System (SAS) was used to establish trimmed flight. Each experiment started with a leader of 2 sec of trimmed flight (hands off, no SAS) which was followed by the initiation of the manoeuvre.
3) A considerable number of repeats and both parities (+/-) for the inputs were part of the experimental program.
4) A modified 3-2-1-1 input was used to allow the aircraft to have a larger signal input while still performing manoeuvres within the small perturbation region.
5) A 4 rps (radian per second) oscillating input was used for the robust determination of control derivatives .
6) Pilot's manual inputs were selected to excite the short period, phugoid and dutch roll modes of the aircraft based on the real-time judgment of the pilot.

7) Various magnitudes for the computer commanded modified 3-2-1-1 sequences were used to obtain information about small perturbation limits and to determine a preferred amplitude range.

8) A stepwise regression analysis was performed to obtain good starting values and to offer alternative estimates of the derivatives for comparison with the MLE values.

7.0 OBVSERVATIONS BASED ON FEBRUARY 1987 BELL-205 FLIGHTS

1) The simultaneous analysis of four manoeuvres (one for each control) gave consistent convergence and allowed estimation of all stability and control derivatives.

2) The stepwise regression results were very similar to the MLE results.

3) Over a speed range from 20-100 kts, the MLE procedure has no convergence problem.

4) Starting values were provided by stepwise regression and their use resulted in relatively quick convergence of the MLE process. (Typically fewer than 7 iterations).

5) From the inputs at different magnitudes it was found that the Cramer-Rao (C-R) bounds were large both for small signal amplitudes and for high signal amplitudes. A preferred range of control amplitudes could therefore be indicated. (See Appendix I.)

6) The sinusoidal excitation at 4 rps input gives the best control derivatives (lowest C-R bounds) as anticipated, but gave still quite credible values for the stability derivatives. (See Appendix 1.)

7) One manually performed modified 3-2-1-1 input was performed at too large a magnitude and it was also deficient in that it showed

significant over-shoots. Therefore, the analysis of this case showed high Cramer-Rao bounds. The rest of the manually flown modified 3-2-1-1 results showed similar C-R bounds as compared with the results obtained using computer generated inputs. (See Appendix 1)

8) In this series of flight tests, none of the hover cases was analysed since in all cases the control inputs produced perturbations beyond the small disturbance amplitudes.

9) Root locus plots (see Figure 13) for flights at 60 kts and for various inputs were made. The stability roots compared reasonably well but showed variations on different control inputs.

10) The estimates of the stability derivatives can be compared with the published values such as those in the NASA CR3144 "Green Book". While these values were found to be similar, not surprisingly the time history fits (see Figures 14 and 15) are better when the derivatives determined by the Parameter Estimation method were used rather than the book values.

8.0 THE SHORTCOMINGS IN THE PRESENT MLE

1) Only linear models can be accepted. Non-linear effects can only be treated as pseudo-controls, and time delay effects cannot be modelled. A non-linear version[11] would be a substantial asset.

2) No rotor dynamics[12-18] (rotor modes) are allowed, other than the implicit quasi-steady effects. The introduction of higher order models incorporating rotor dynamics will most likely require additional measured information related to the rotor state to assist in the determination of the many additional parameters.

3) The modelling of inflow dynamics[17-21] (induced flow) will affect the thrust, pitching moment and rolling moment equations. Again, additional measurements may be necessary to use more complex models incorporating these effects.

4) The modelling of the engine governor system[22-23] may also be required if model improvements are sought.

5) Significant non-linear effects[24] may exist at practical levels of control inputs.

9.0 RECOMMENDATIONS FOR FUTURE RESEARCH

1) A non-linear version of the maximum likelihood estimator will expand the capability of the identification technique.

2) A low pass filter with a cutoff frequency of 1 Hz placed just before the control actuators may inhibit the excitation of the rotor modes.

3) An augmentation system using attitude feedback (θ, ϕ) may possibly facilitate the testing at hover and low speed conditions. If rate feedback is a necessity to obtain satisfactory manoeuvres, then the estimation process may be hampered.

4) An investigation into the use of higher order terms in the model for the aerodynamic forces and moments, e.g. the body accelerations and control rate terms[25], may show improvement.

5) A rotorcraft simulation program, e.g. C81, will provide the rotor and body aerodynamic derivatives and time histories. These time histories will be used to validate the linear, 6 degree of freedom mathematical model.

6) Further development in the two-step method, which is a flight path reconstruction program, followed by a stepwise regression analysis may improve the identification procedure.

10.0 REFERENCES

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FIG. 1: THE NAE BELL-205 AIRBORNE SIMULATOR

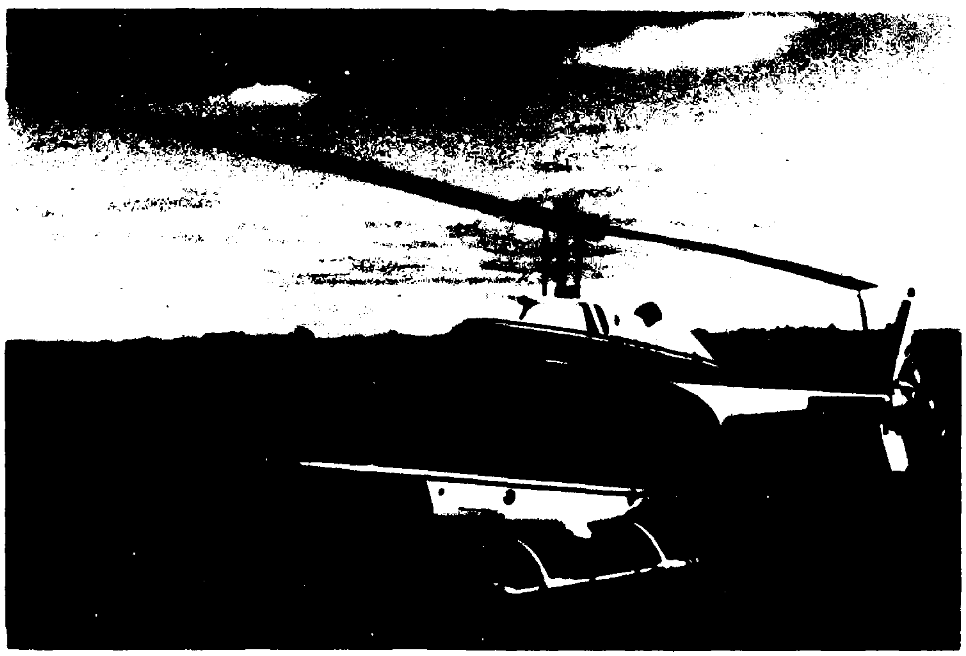


FIG. 2: THE NAE BELL-206 JET RANGER

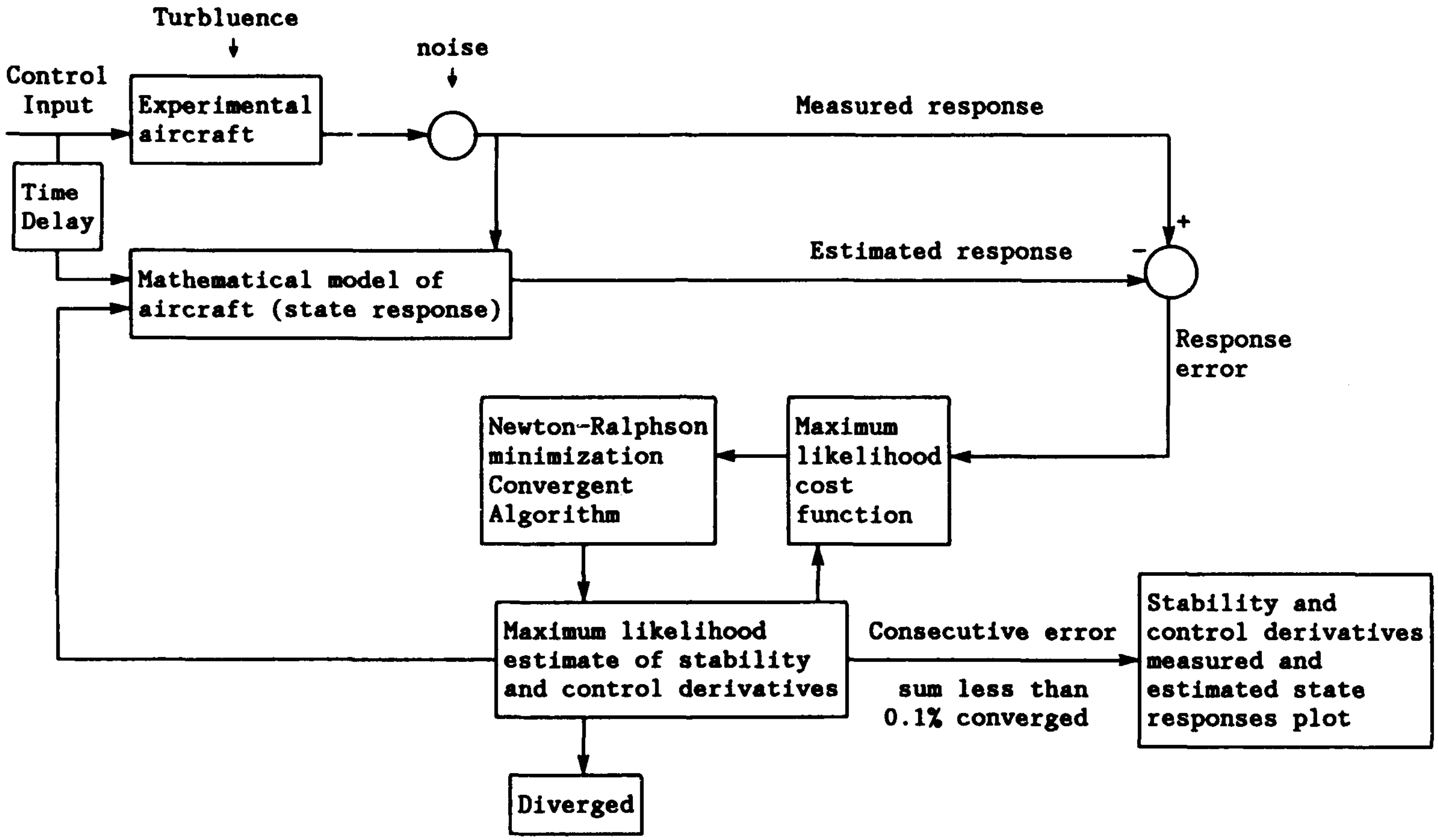


Figure 3: MMLE3 Overall Flowchart

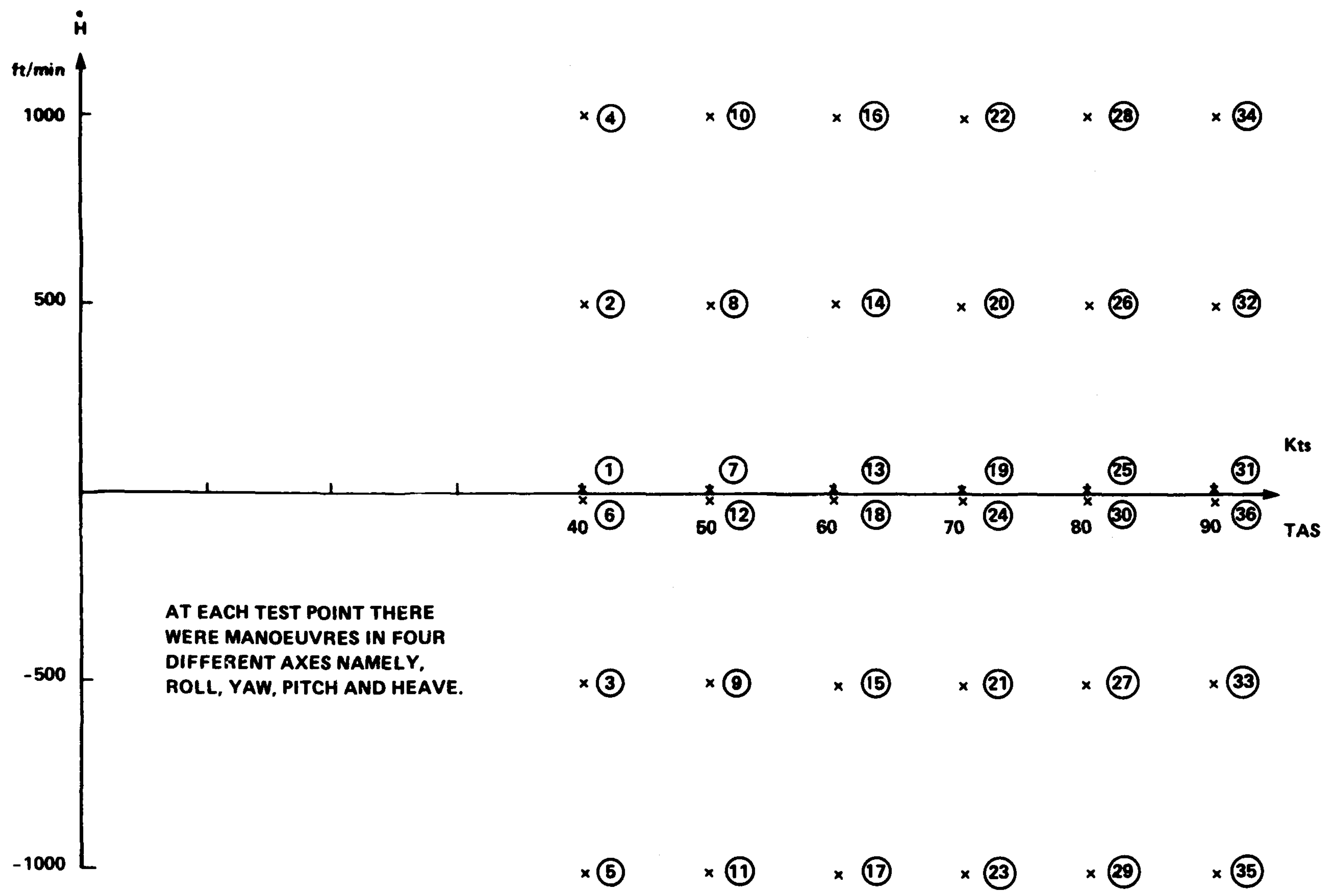
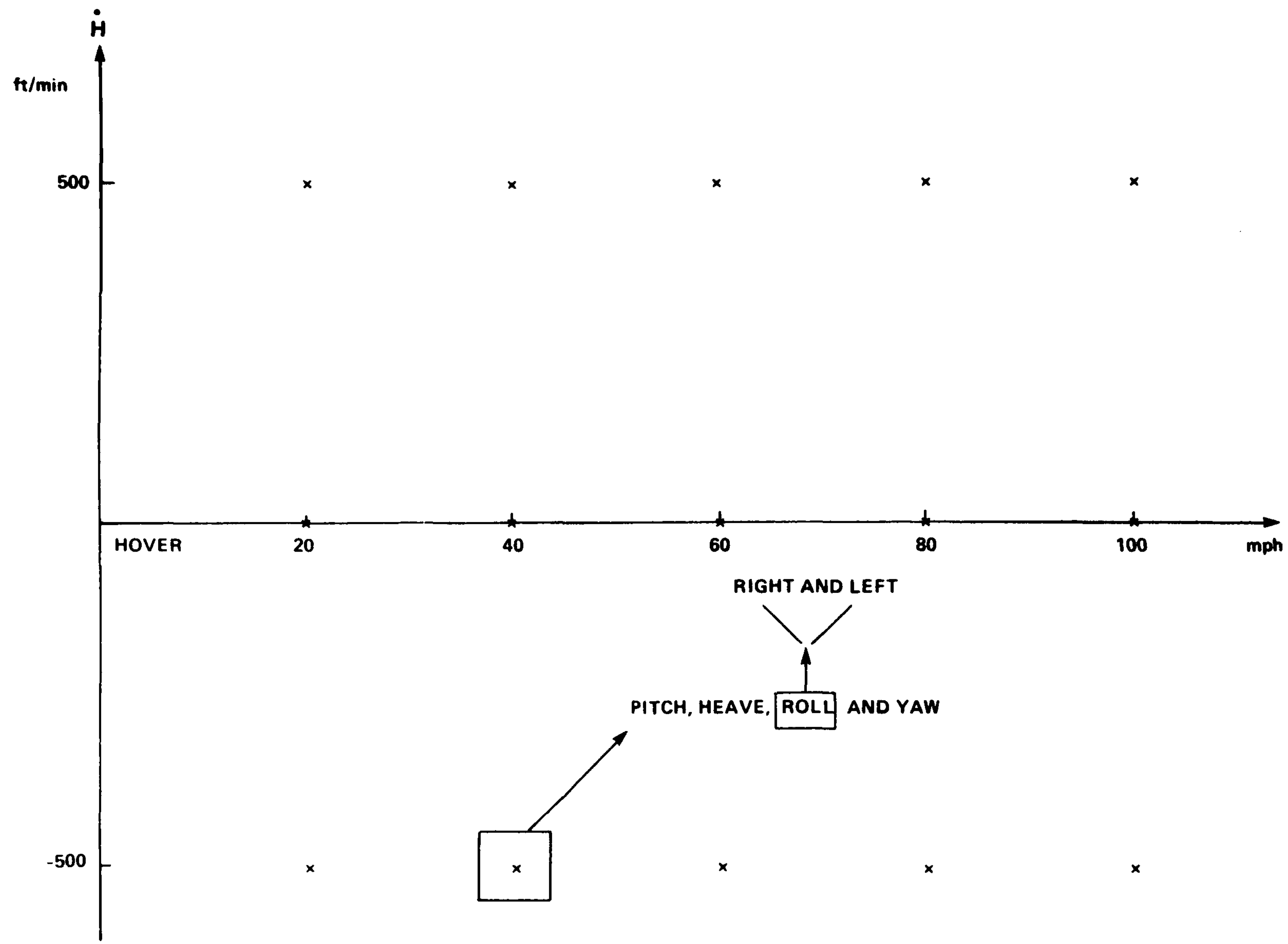


FIG. 4: BELL-205 FLIGHTS COVERING A CARPET OF FLIGHT CONDITIONS



AT EACH TEST POINT, THERE WERE MANOEUVRES IN FOUR DIFFERENT AXES NAMELY ROLL, YAW, PITCH AND HEAVE.
AT EACH MANOEUVRE, THERE ARE EXCITATIONS IN BOTH DIRECTIONS.

FIG. 5: BELL-206 FLIGHTS COVERING A CARPET OF FLIGHT CONDITIONS

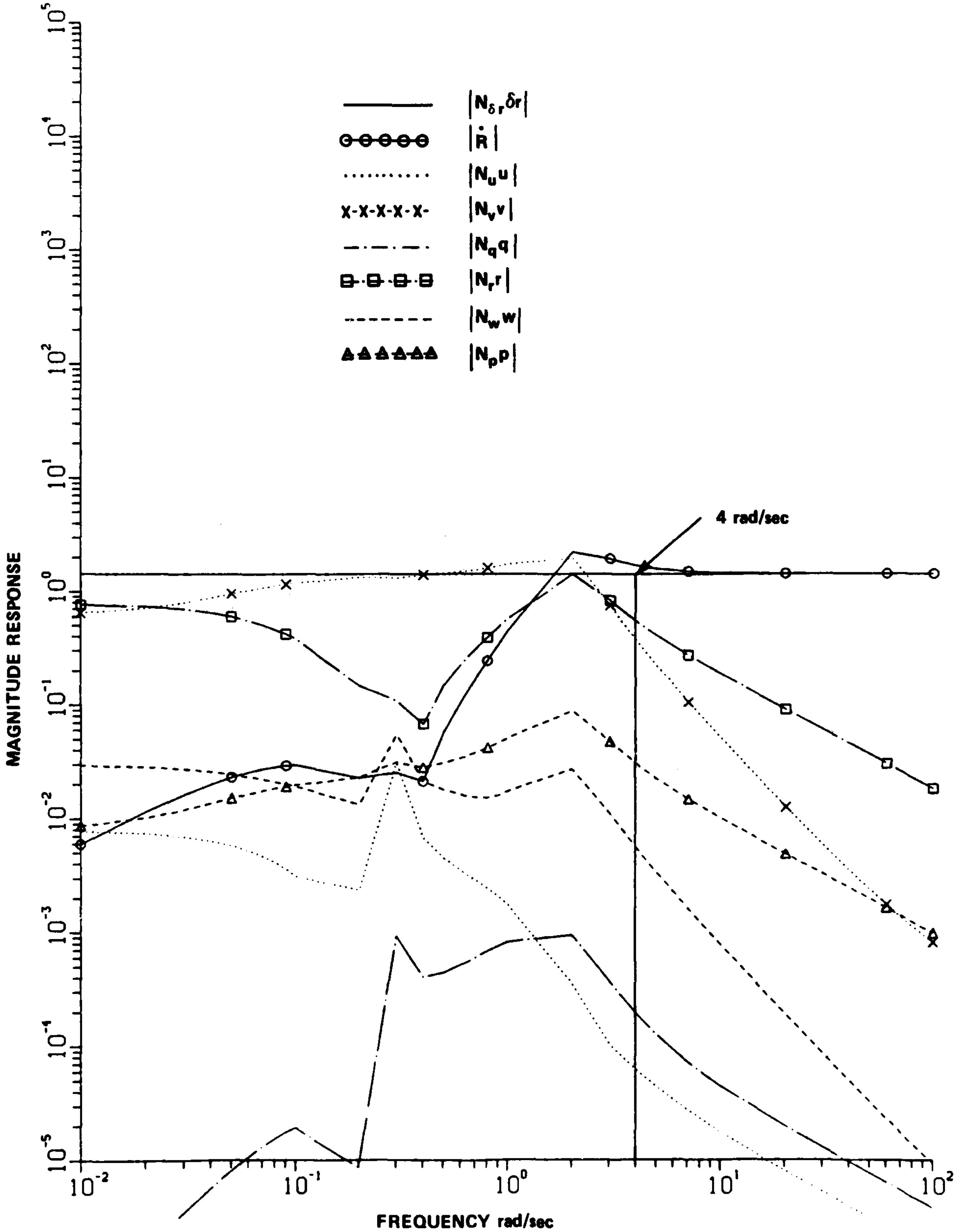


FIG. 6: BODE PLOT – YAW RESPONSE

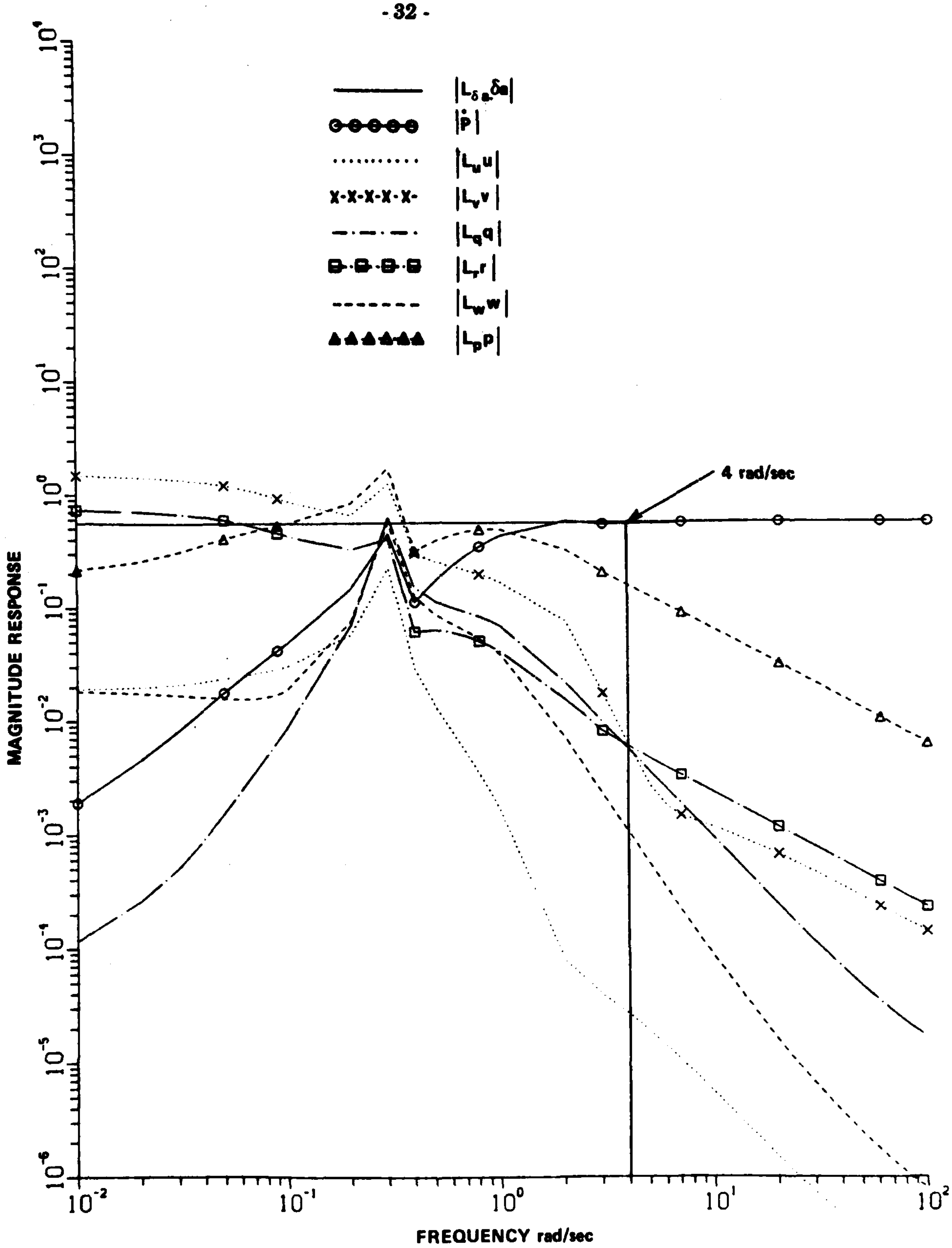


FIG. 7: BODE PLOT – ROLL RESPONSE

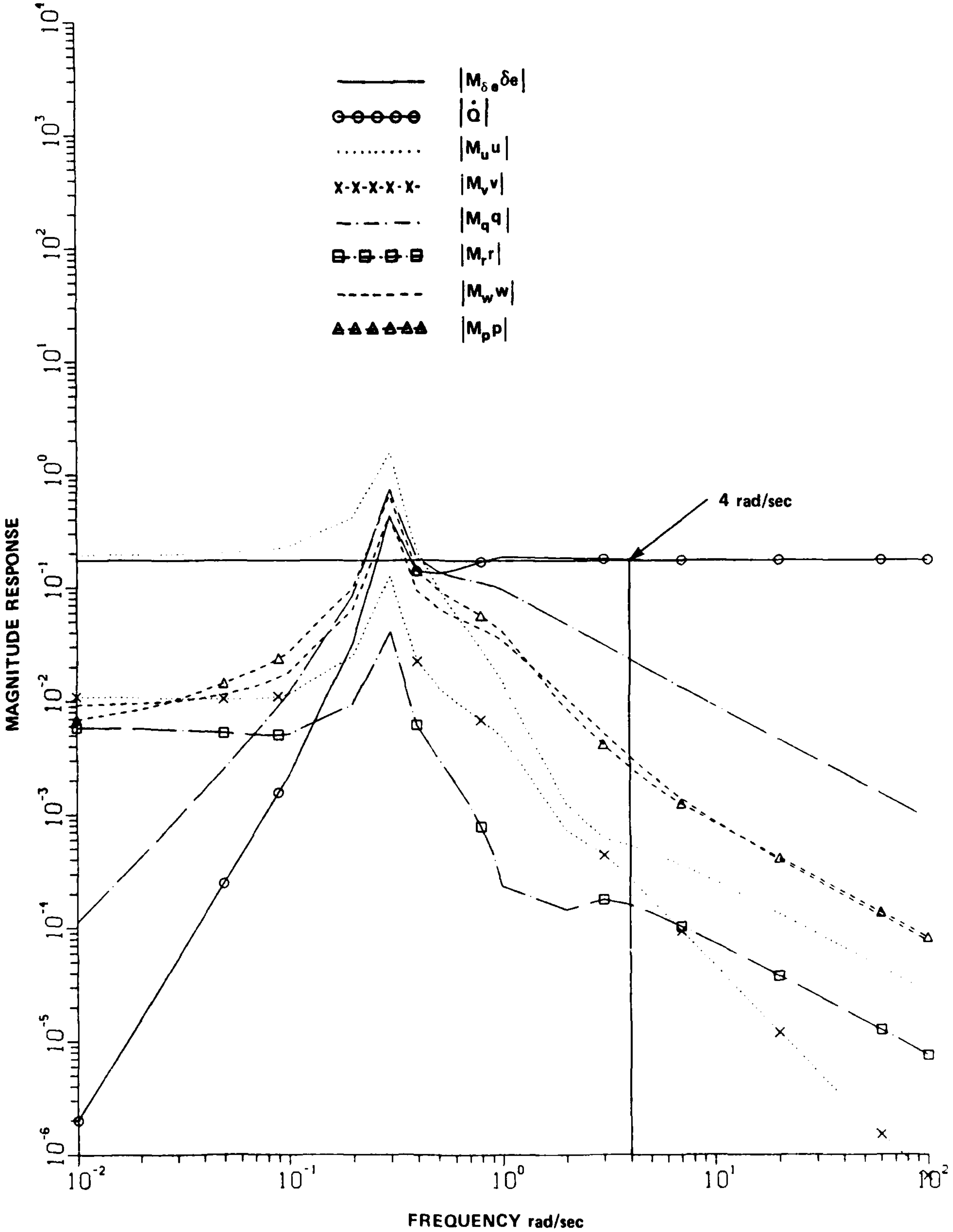


FIG. 8: BODE PLOT — PITCH RESPONSE

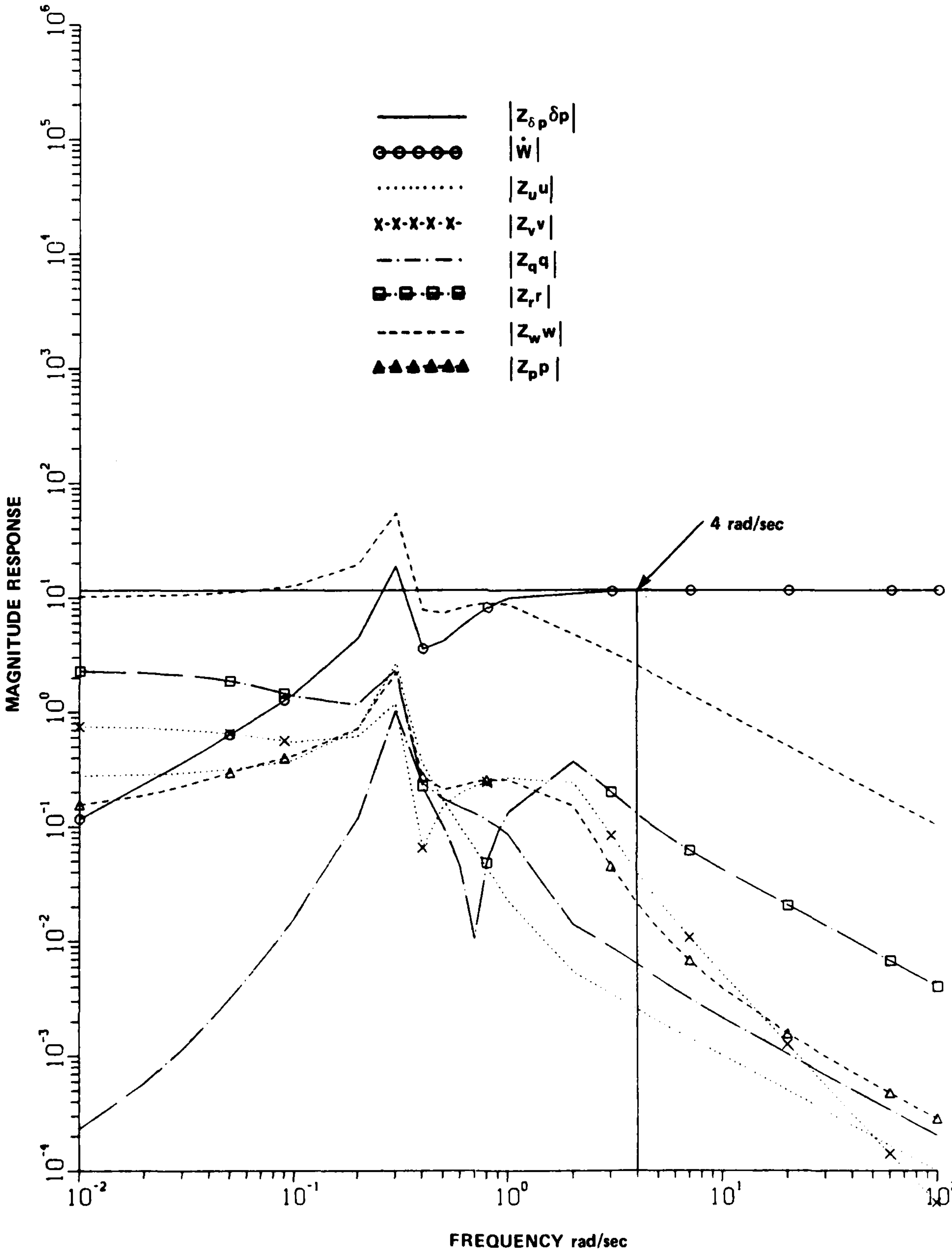


FIG. 9: BODE PLOT – HEAVE RESPONSE

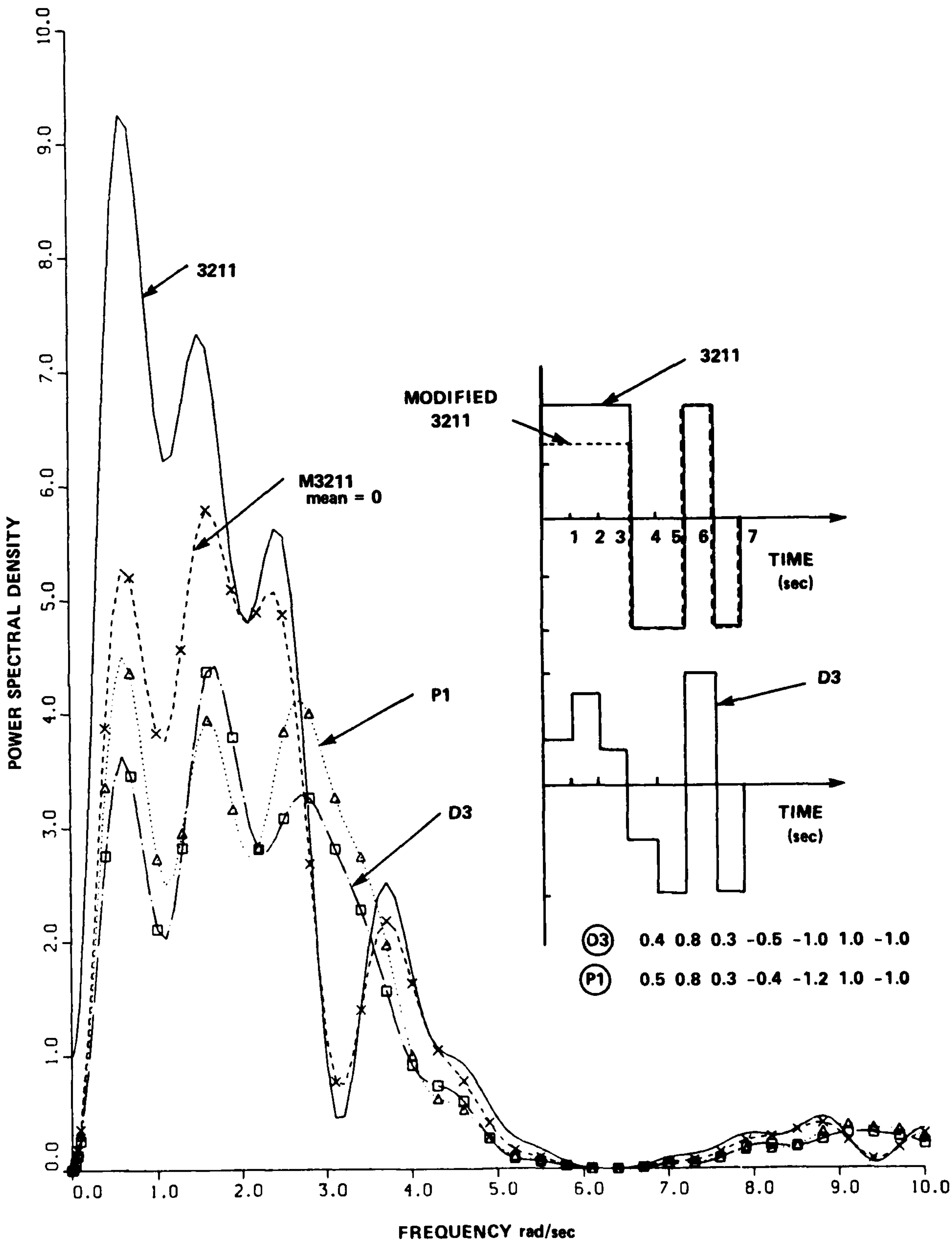
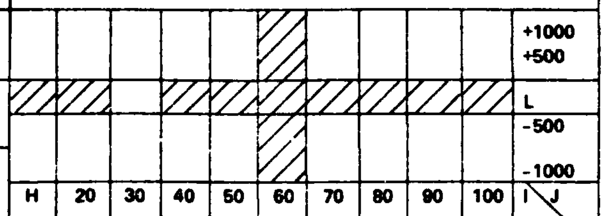


FIG. 10: INPUT OPTIMIZATION

Climb / Descend	Input	Direction	Channels	Times	Flight No.	(Kts) Speed	Pilot	Remarks
Level	M3211	2	4	3	005	60	Bob - Keith	Baseline case — a carpet of flight conditions for an optimized input modified 3-2-1-1 and compared to 4 rad/sec sinusoid input.
± 500	M3211	2	4	2	006	60	Bob - Keith	
± 1000	M3211	2	4	2	007	60	Bob - Keith	
Level	M3211	2	4	2	012	40,80 100	Bob - Keith	
Level	M3211	2	4	2	019	50,90 100	Bob – Keith	
Level	M211 OSC	2 1	4	2	021	70 70,100	Dan - Murray	
Level	Doublet	2	4	2	009	20	Bob - Keith	
Level	Doublet OSC	2 1	4	2	008	Hover	Bob - Keith	I – Electronic M3211
Level	Multi-axes	2	4	2	024	Hover	Bob - Keith	J – OSC 4 rad/sec
Level	OSC	1	4	2	013	20	Bob - Keith	
Level	OSC	1	4	2	011	40 → 90	Dan - Murray	
Level ± 500 ± 1000	OSC	1	4	2	010	60	Dan - Murray	
Level	Elect-M3211 Manual M3211	2	4	2	022	60	Bob - Keith	Pilot's manual M3211 versus Electronic M3211
Level	M3211	2	4	4/5	017	60	Bob - Keith	Various input magnitudes signal/noise ratio compared to nonlinear effects.
Level	Pilot's Choice	2	4	2	025	60	Bob - Keith	Pilot tries to excite phugoid and Dutch roll modes

Flight number 001 - 004 were used to integrate the instrumentation.
Flight number 012, 014, 015 etc. were used for practice run for less stable conditions and small perturbation limits.

FIG. 11: FLIGHT TEST MATRIX BELL-205 FOR SIX DEGREE OF FREEDOM MMLE3

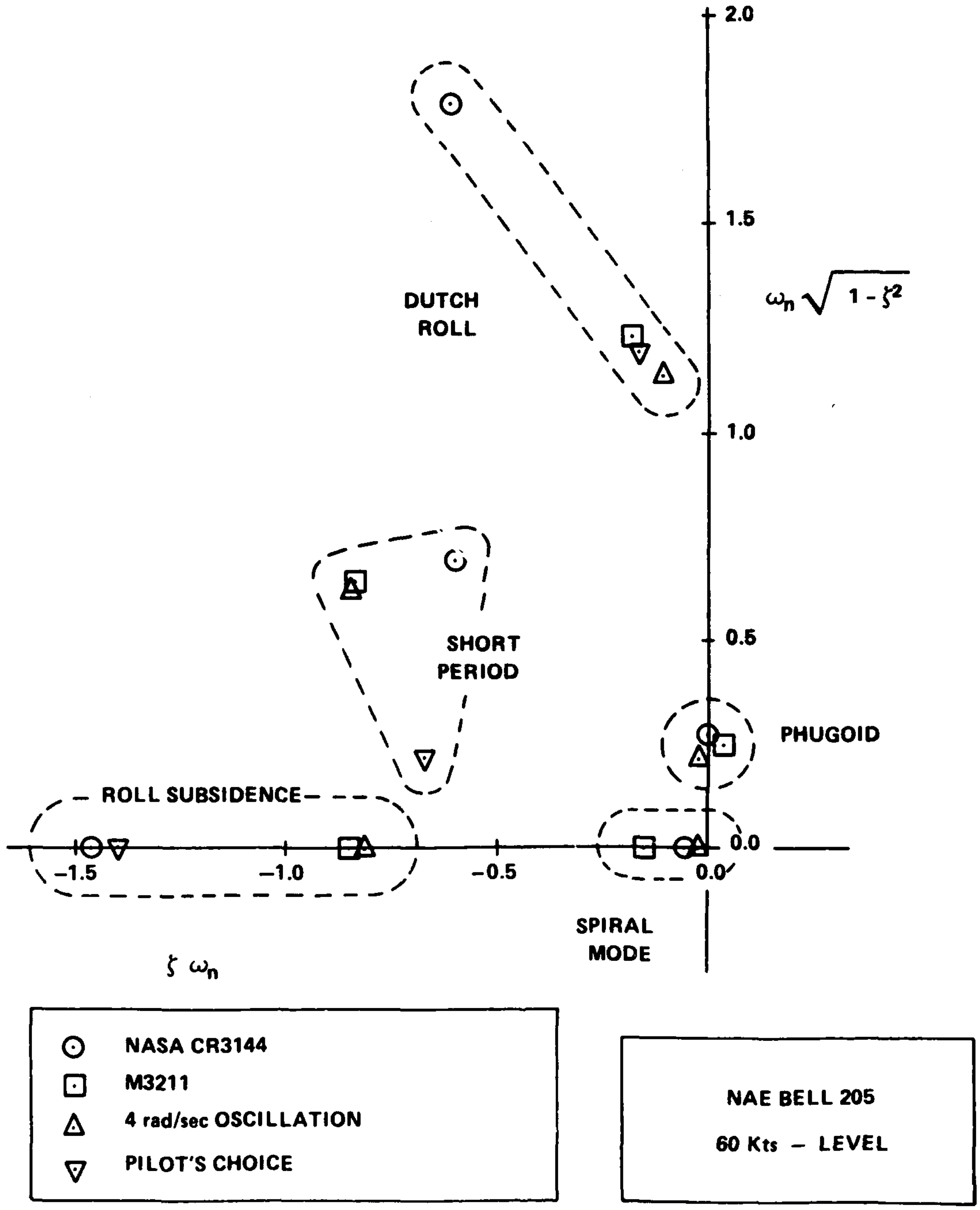


FIG. 12: PLOT OF STABILITY ROOTS

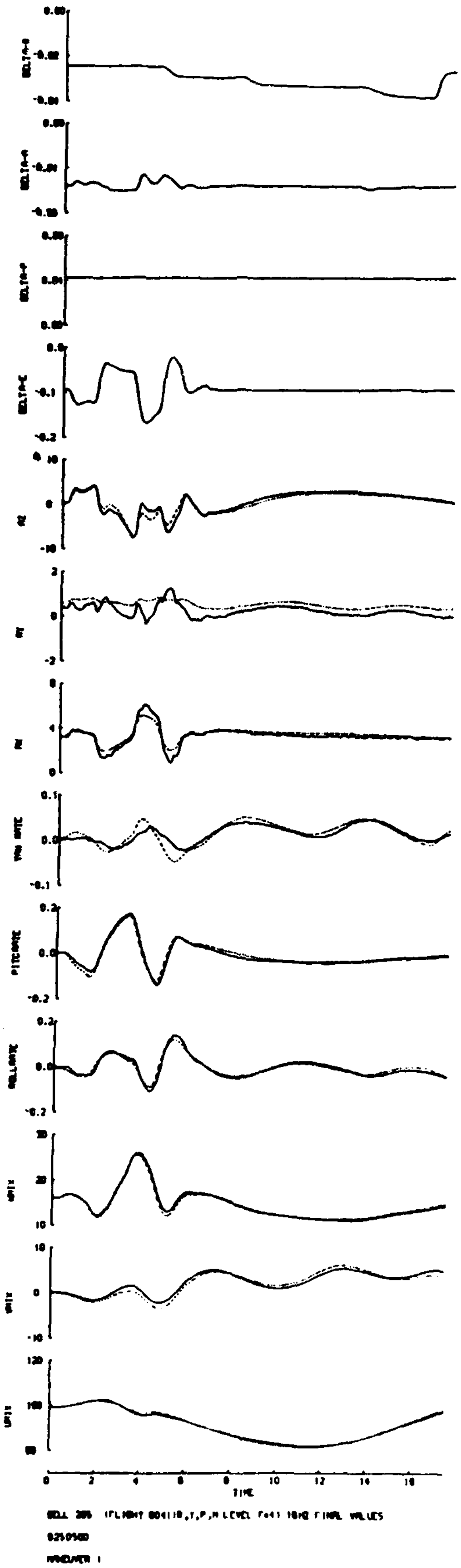


FIG. 13a: TIME HISTORY FIT (USING IDENTIFIED DERIVATIVES)

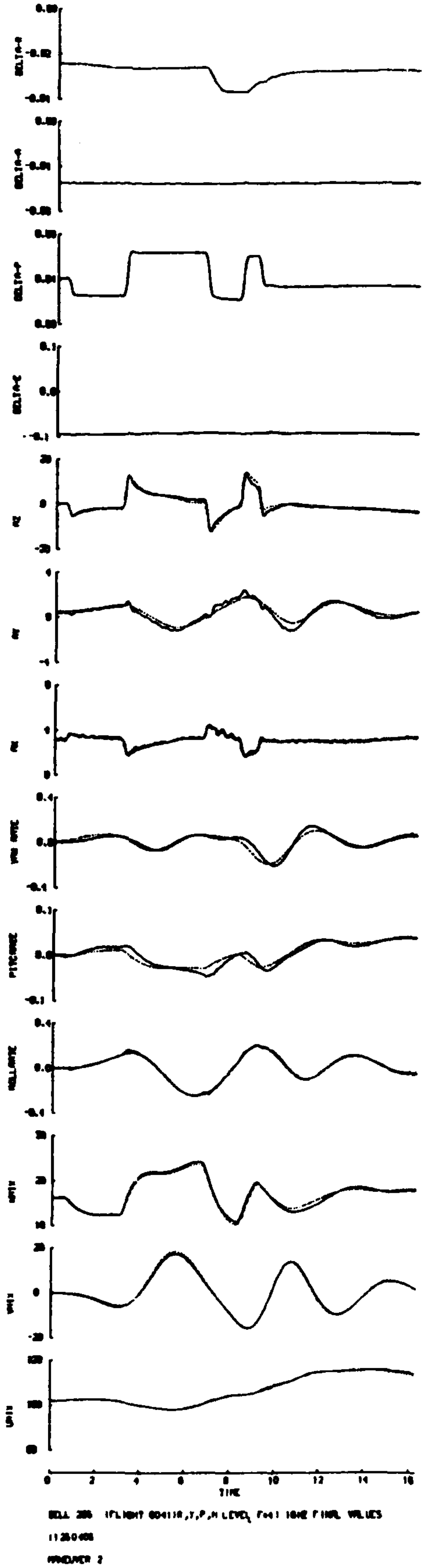


FIG. 13b: TIME HISTORY FIT (USING IDENTIFIED DERIVATIVES)

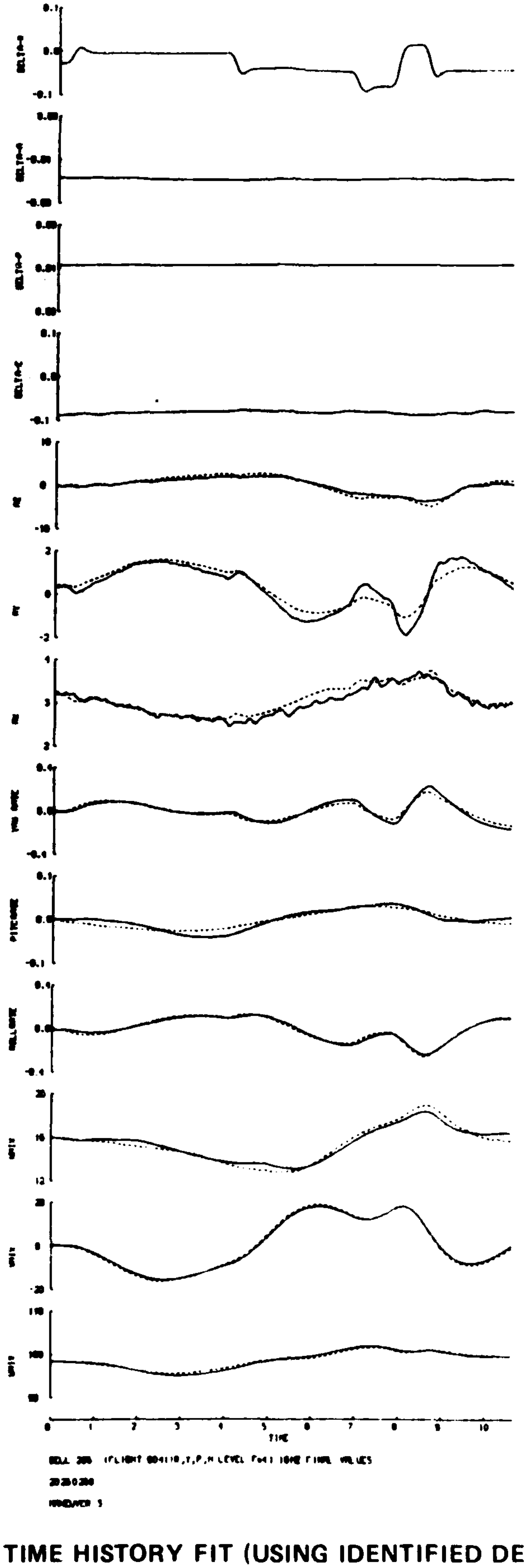


FIG. 13c: TIME HISTORY FIT (USING IDENTIFIED DERIVATIVES)

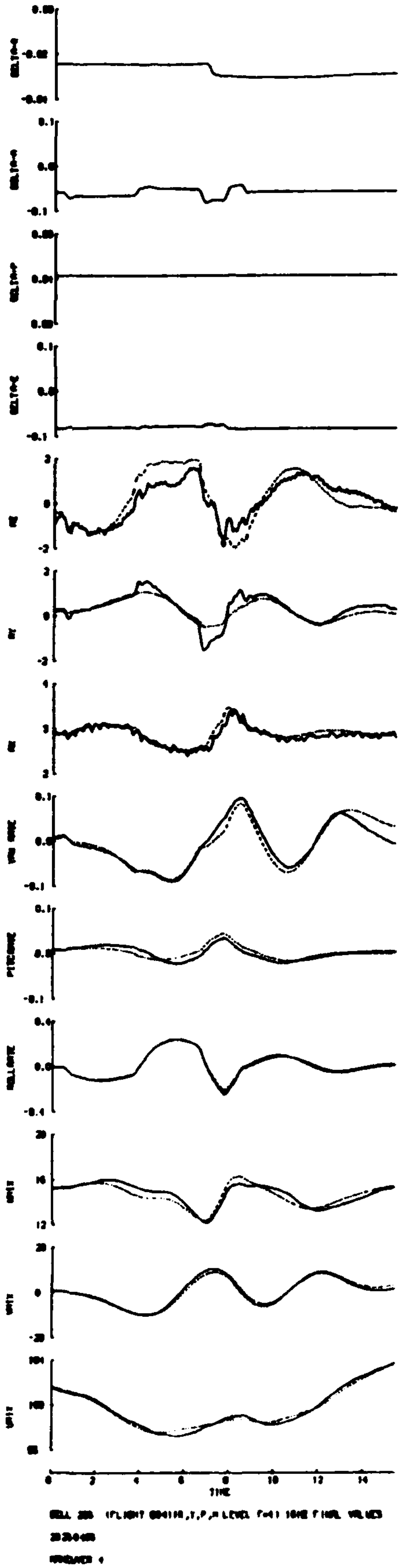


FIG. 13d: TIME HISTORY FIT (USING IDENTIFIED DERIVATIVES)

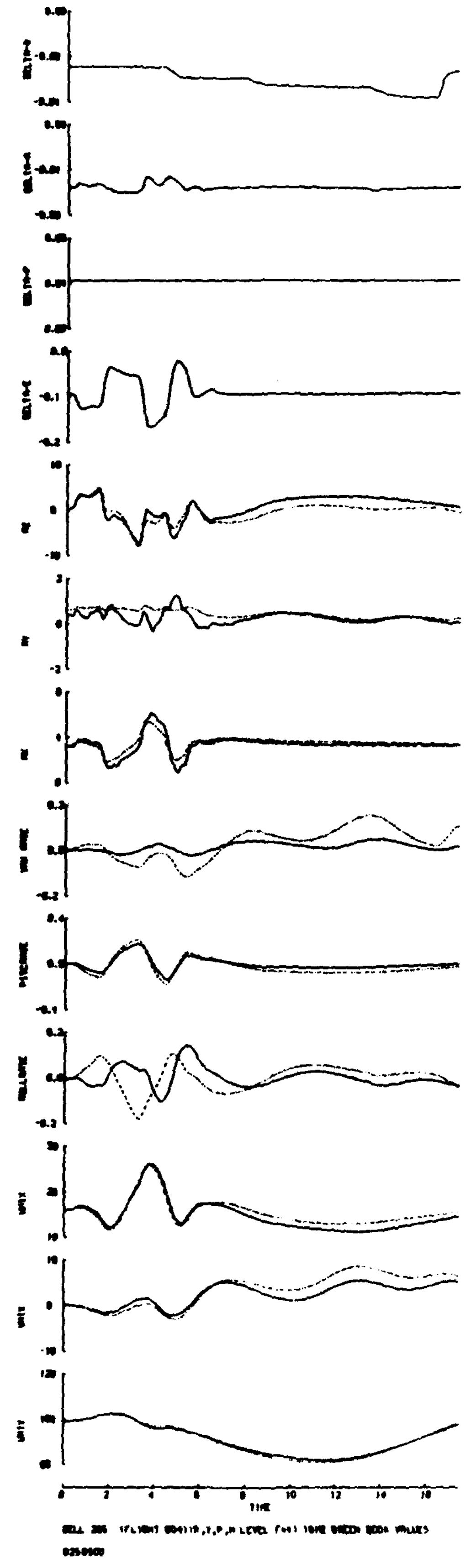


FIG. 14a: TIME HISTORY FIT (USING CR3144 DERIVATIVES)

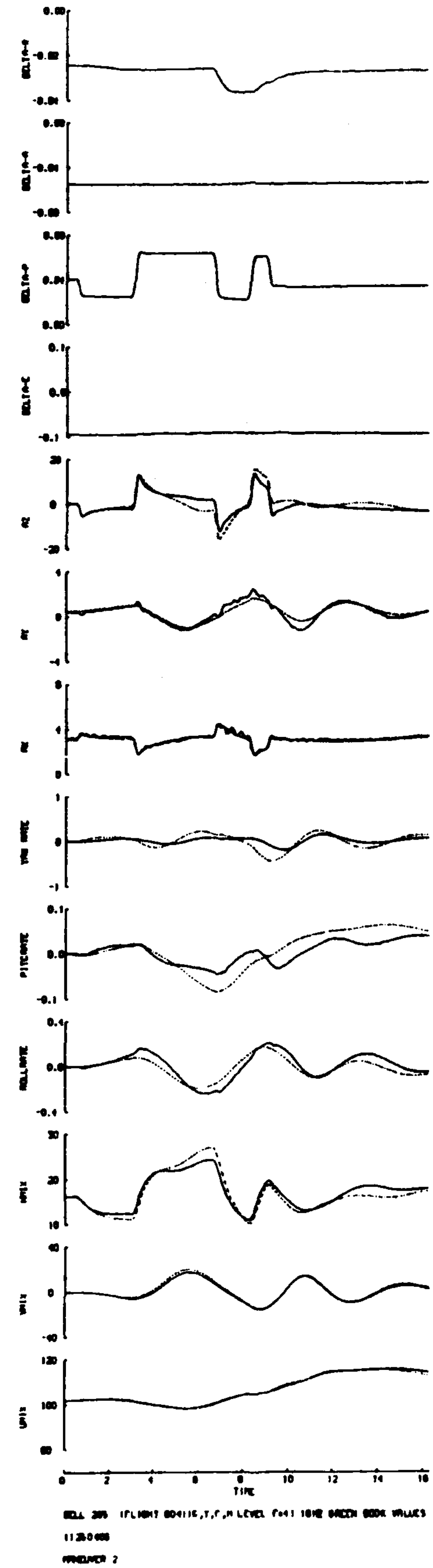


FIG. 14b: TIME HISTORY FIT (USING CR3144 DERIVATIVES)

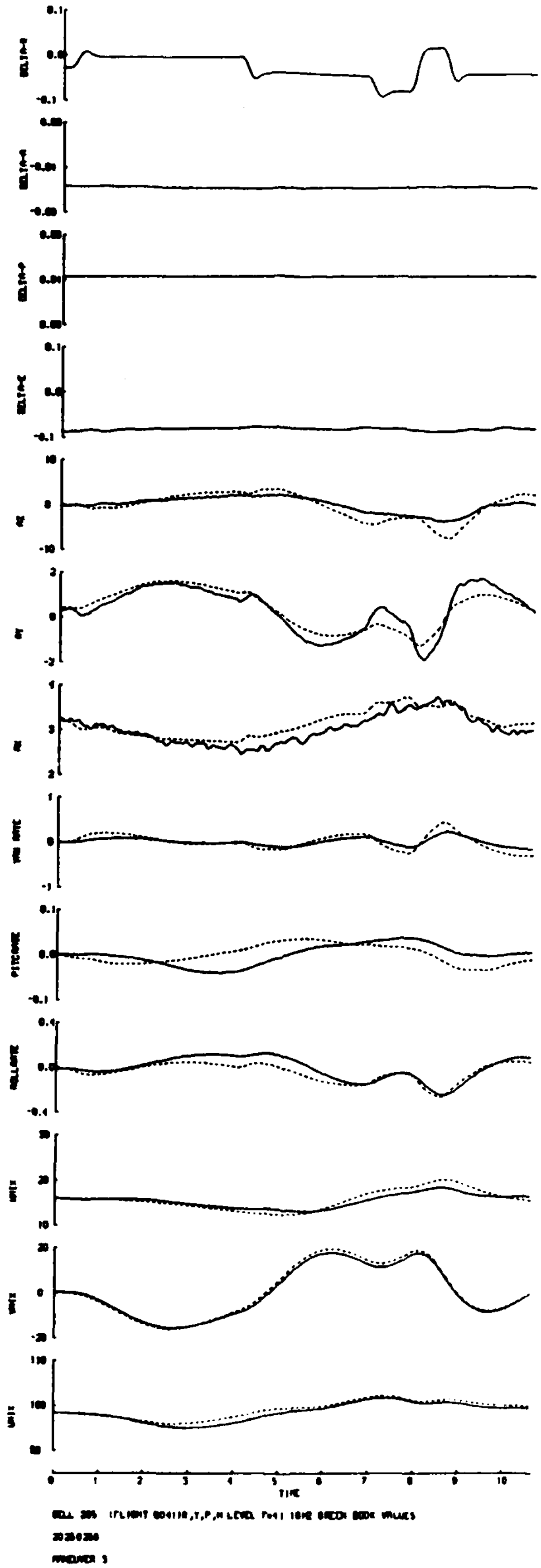


FIG. 14c: TIME HISTORY FIT (USING CR3144 DERIVATIVES)

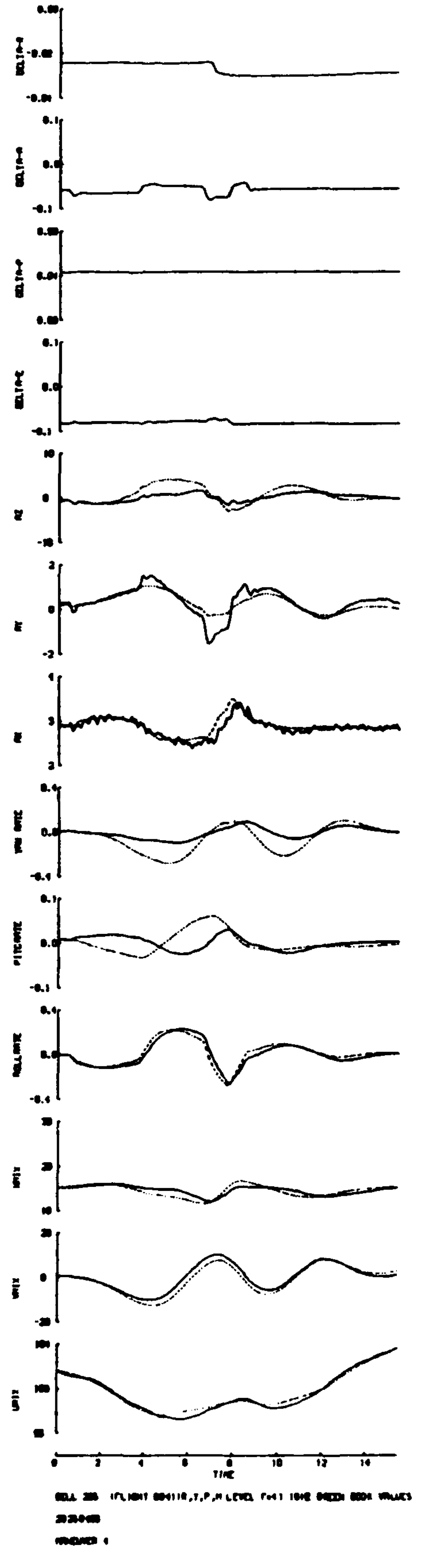
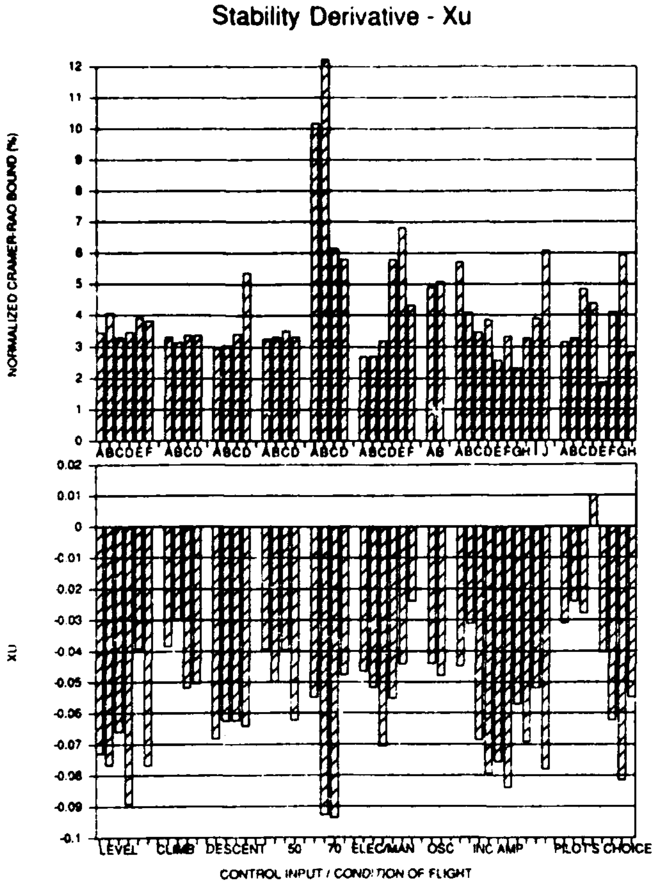
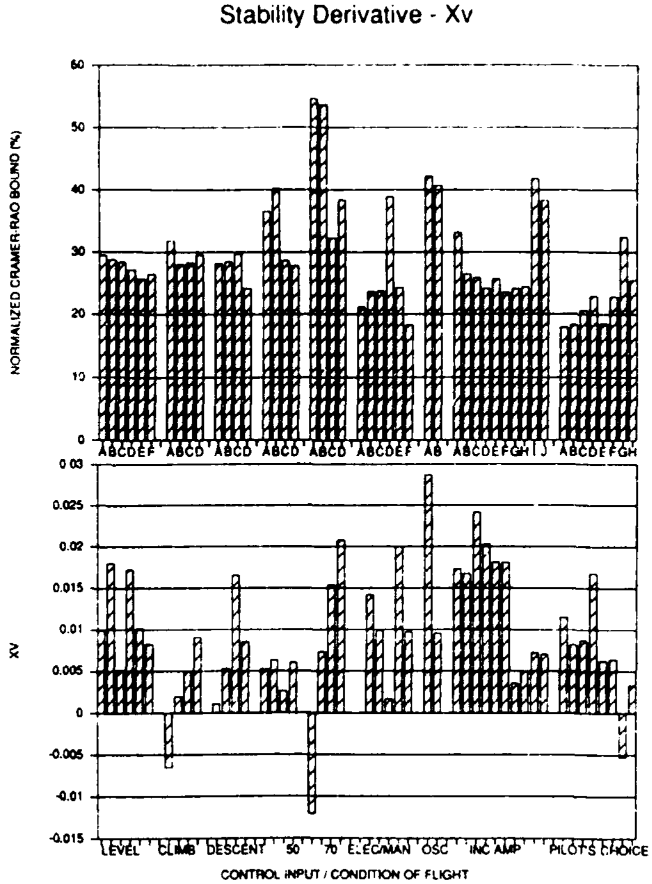


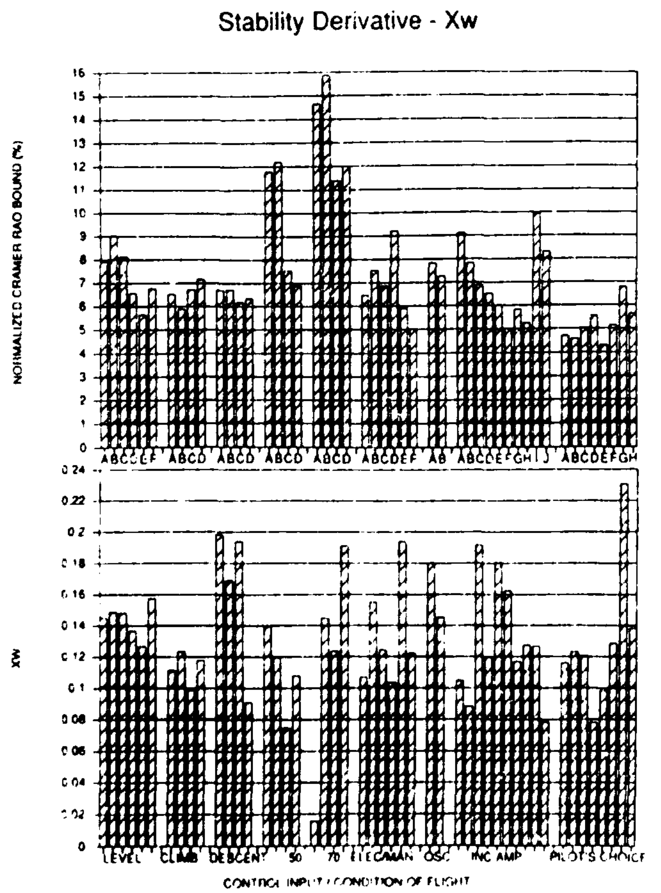
FIG. 14d: TIME HISTORY FIT (USING CR3144 DERIVATIVES)

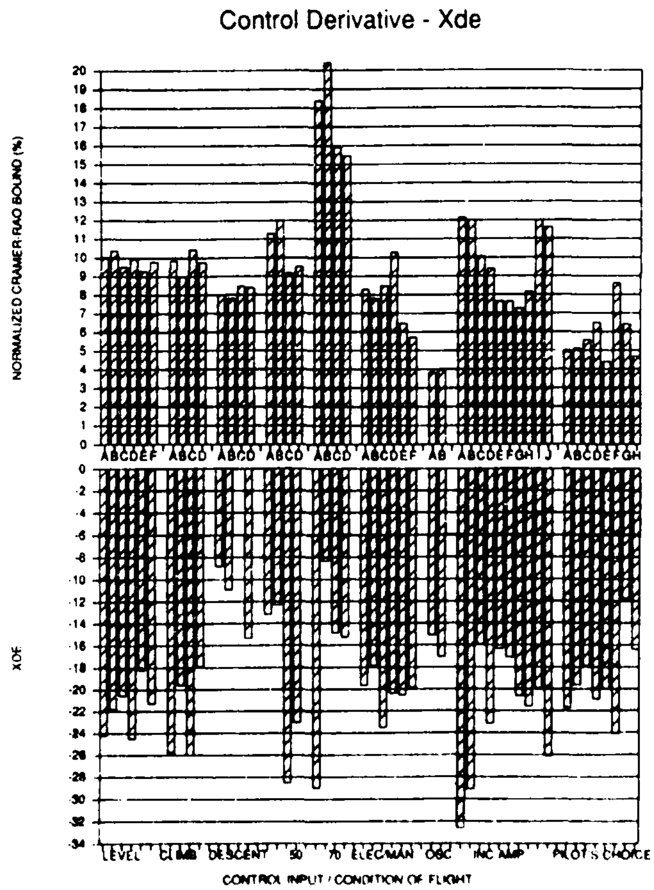
APPENDIX

Stability and control derivatives and their Cramer Rao (CR) bounds are presented for 60 knots level flight, climbing and descending, for 50 knots level flight, and for 70 knots level flight cases. Data are also plotted for the 60 knots level flight case, following control input excitations: electronic modified 3-2-1-1 inputs versus manual attempts at the same modified 3-2-1-1 inputs; oscillation input at frequency of 4 rps; 5 different magnitudes of modified 3-2-1-1 inputs and the "pilot's choice" input. (Designations for these various flight and excitation cases appear on the horizontal axis of the plotted data).



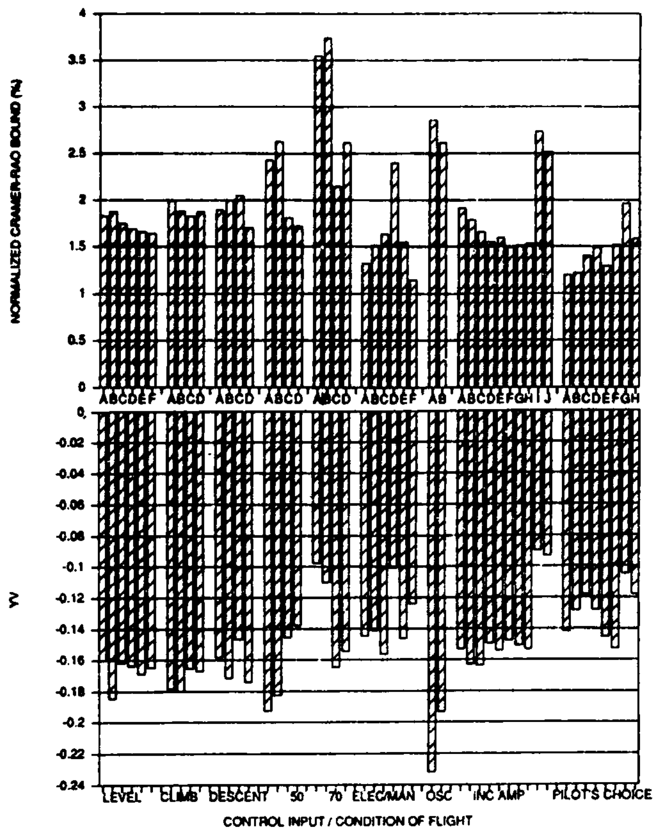




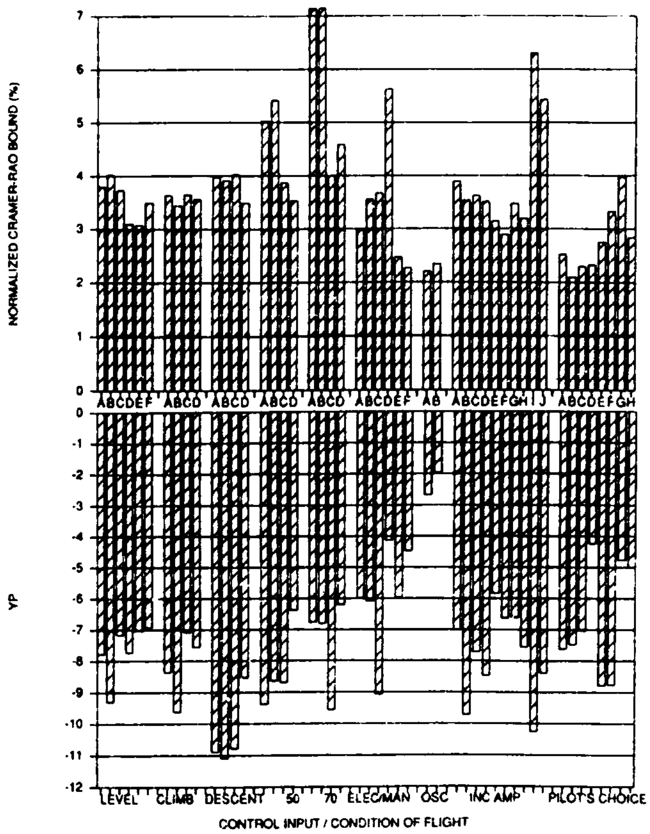


THE PLOT OF FOUR TYPICAL X-DERIVATIVES

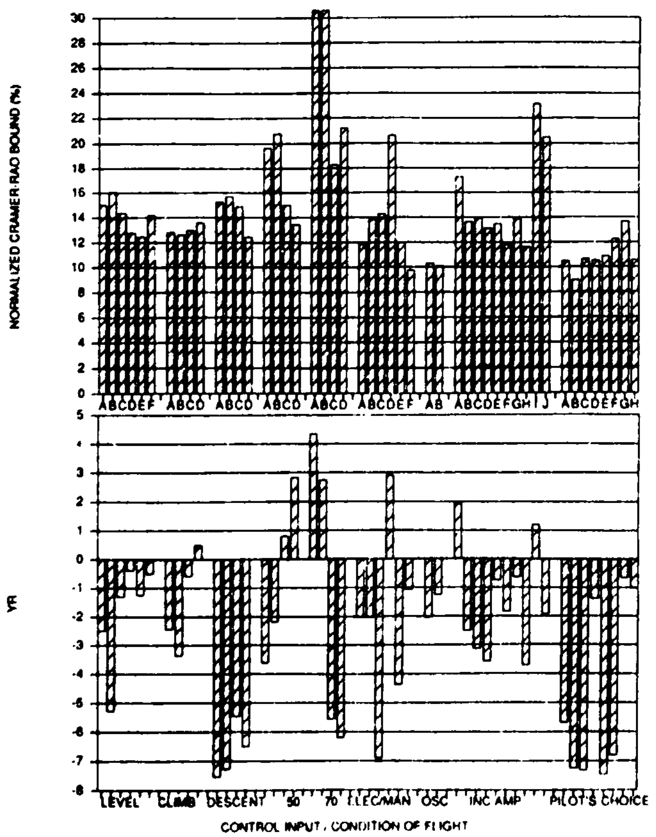
Stability Derivative - Yv



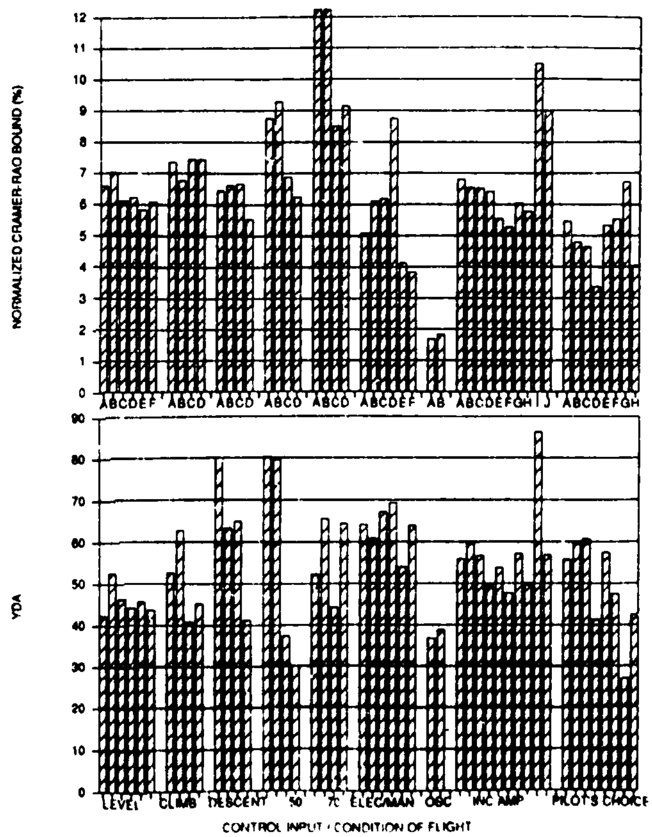
Stability Derivative - Yp



Stability Derivative - Yr

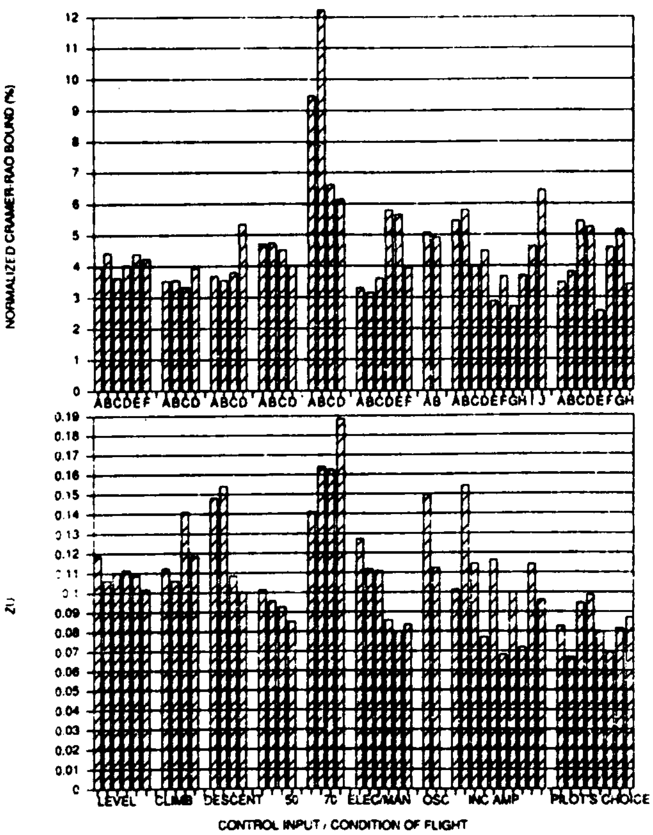


Control Derivative - Yda

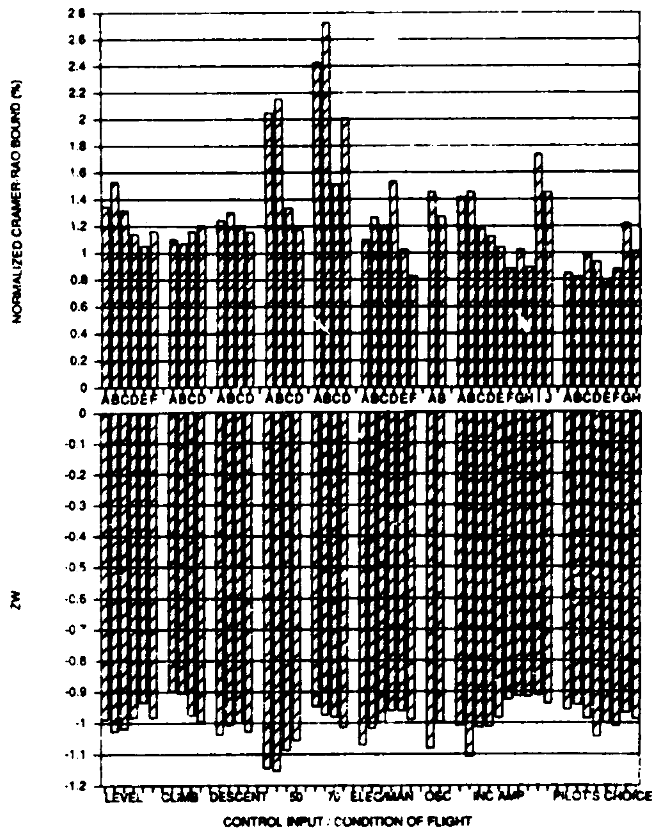


THE PLOT OF FOUR TYPICAL Y-DERIVATIVES

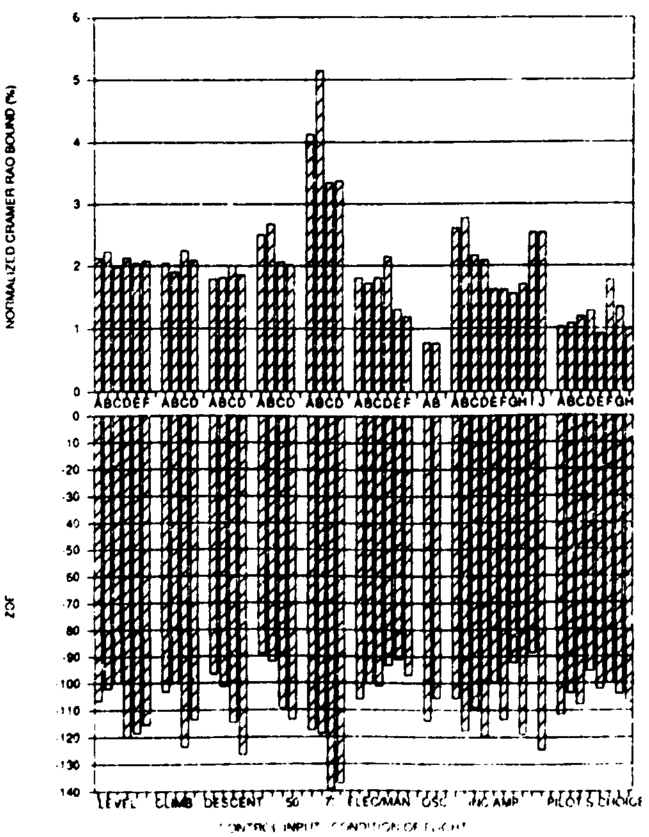
Stability Derivative - Zu



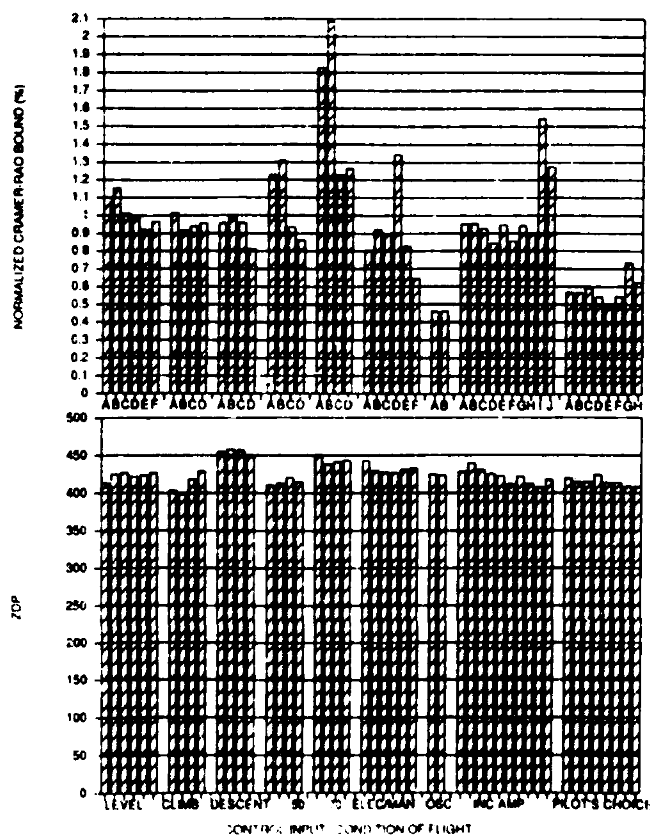
Stability Derivative - Zw



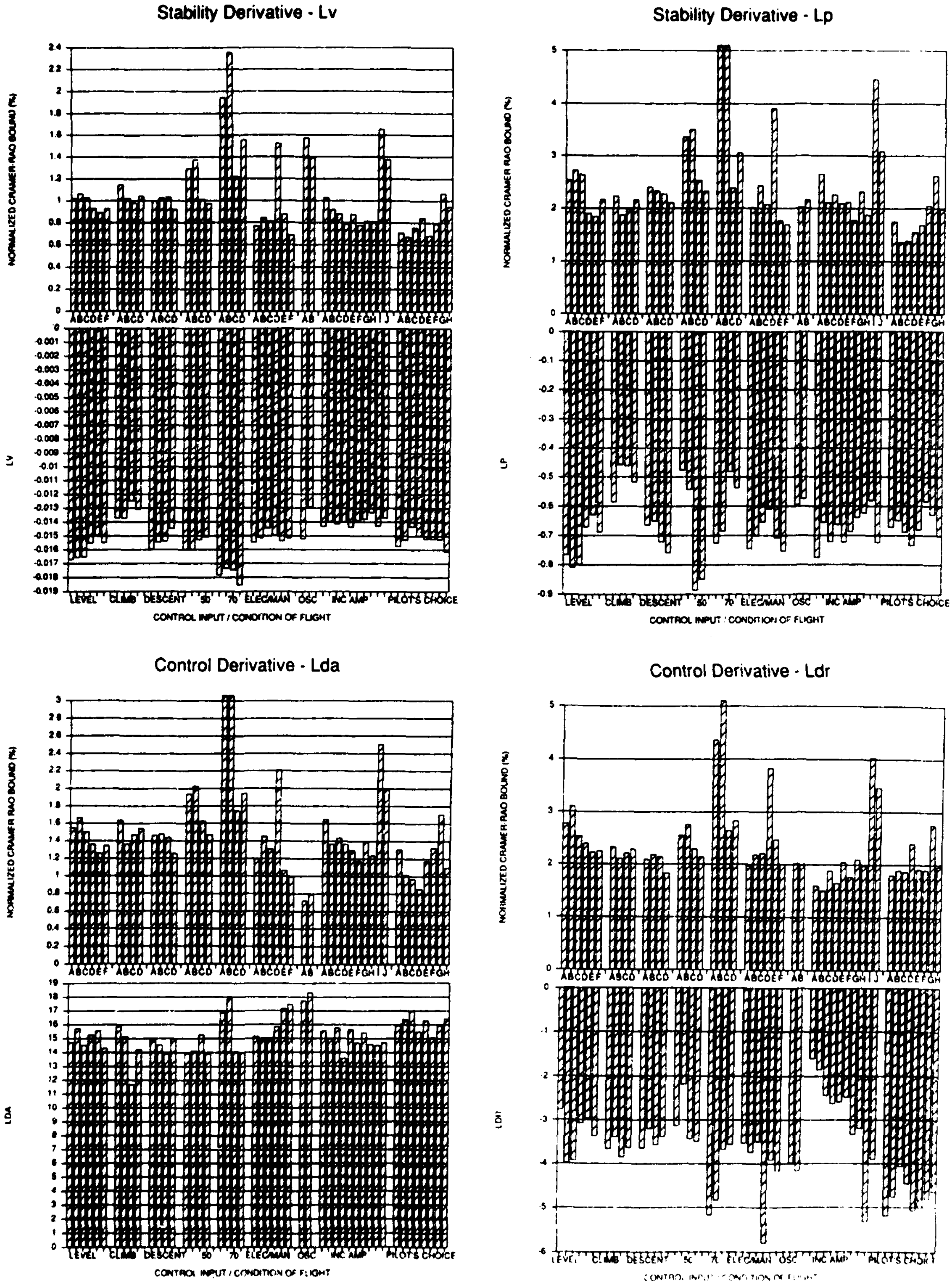
Control Derivative - Zde



Control Derivative - Zdp

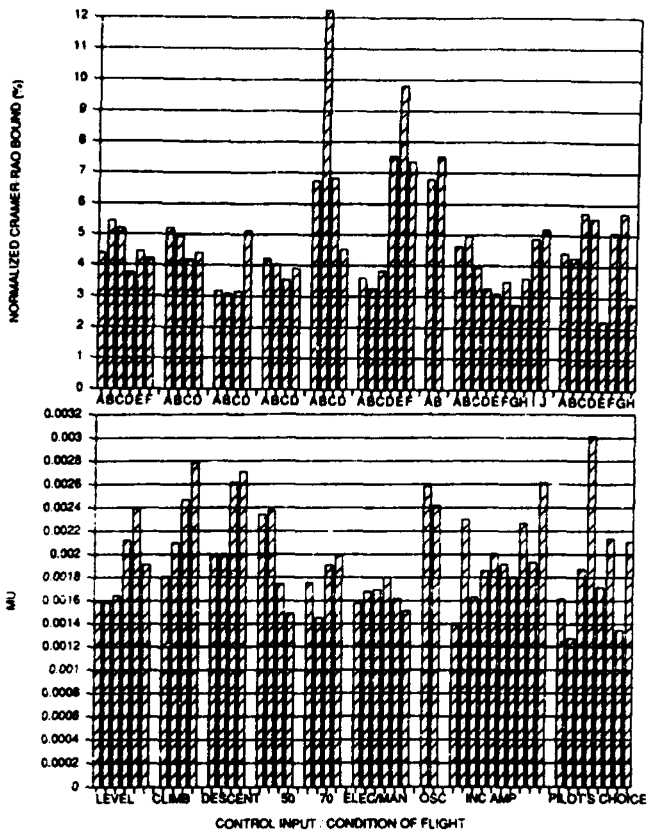


THE PLOT OF FOUR TYPICAL Z-DERIVATIVES

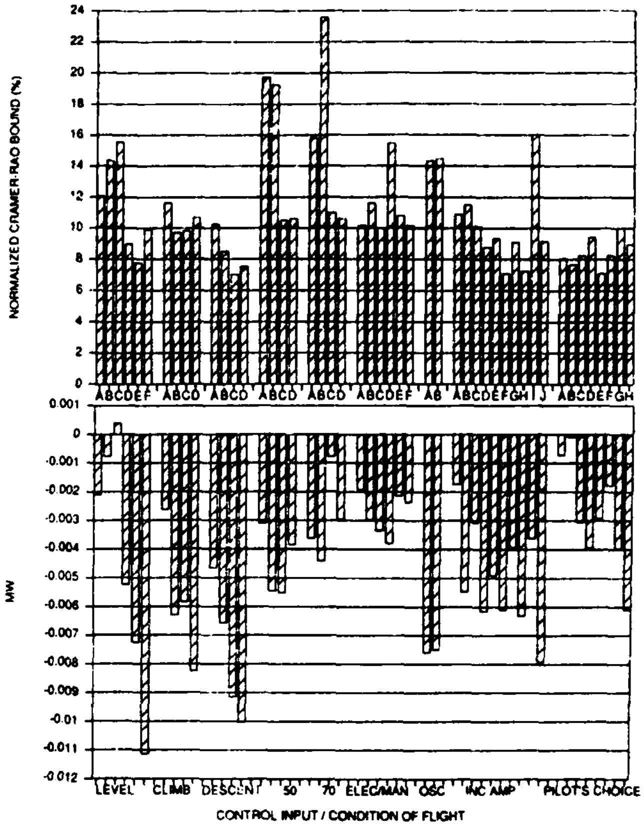


THE PLOT OF FOUR TYPICAL L-DERIVATIVES

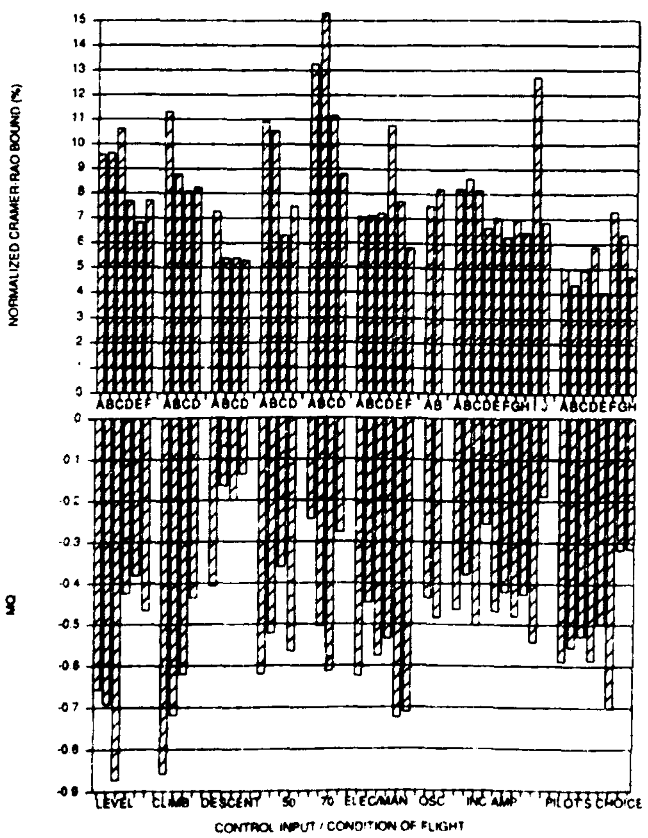
Control Derivative - Mu



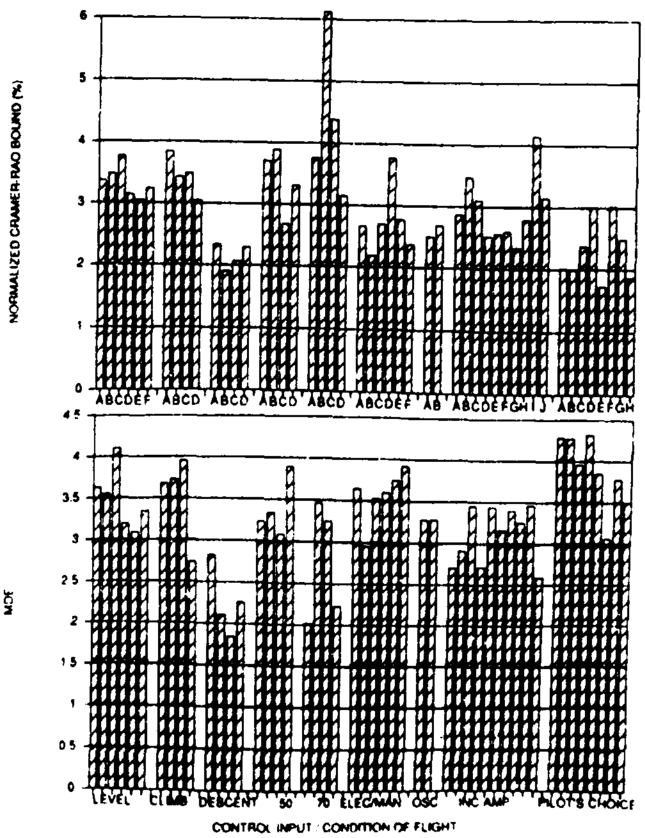
Stability Derivative - Mw



Stability Derivative - Mq

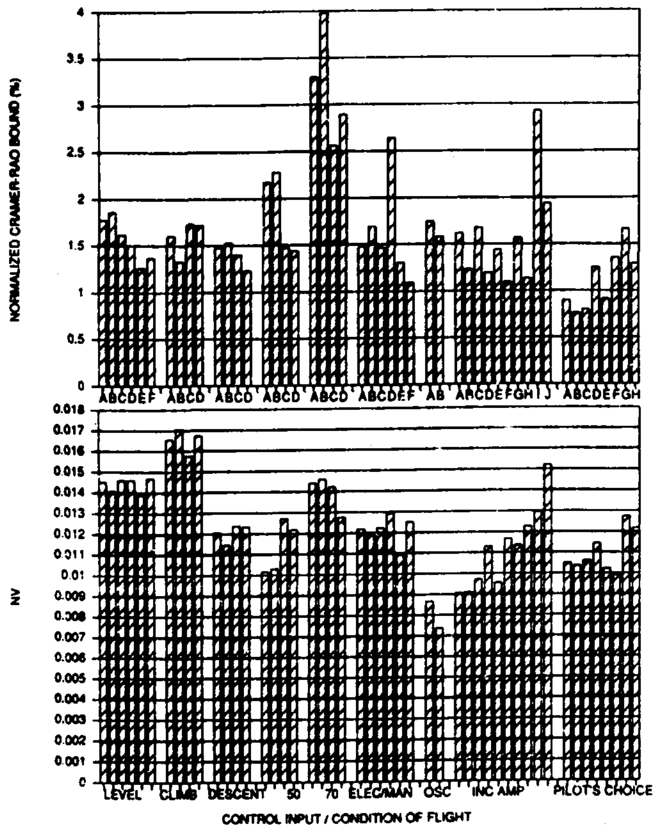


Control Derivative - Mde

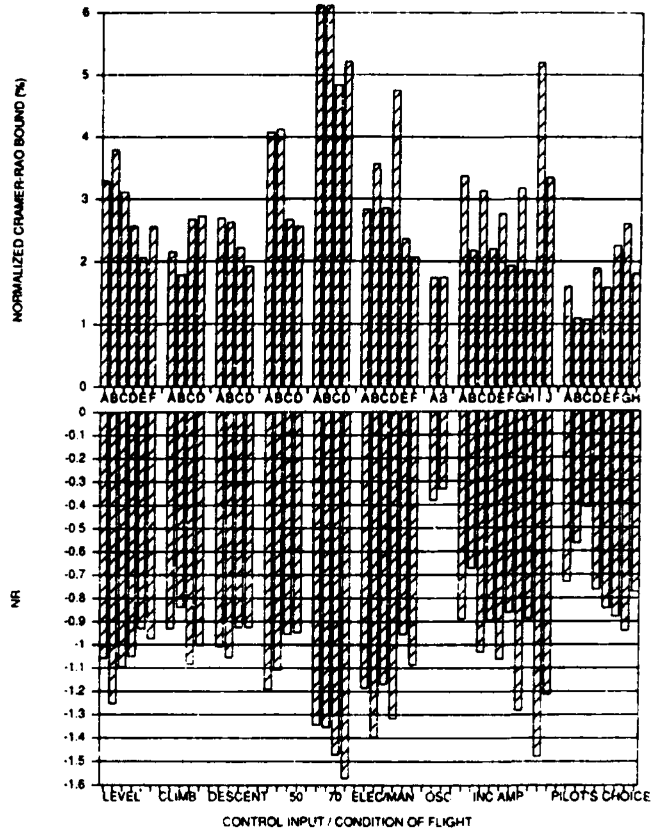


THE PLOT OF FOUR TYPICAL M-DERIVATIVES

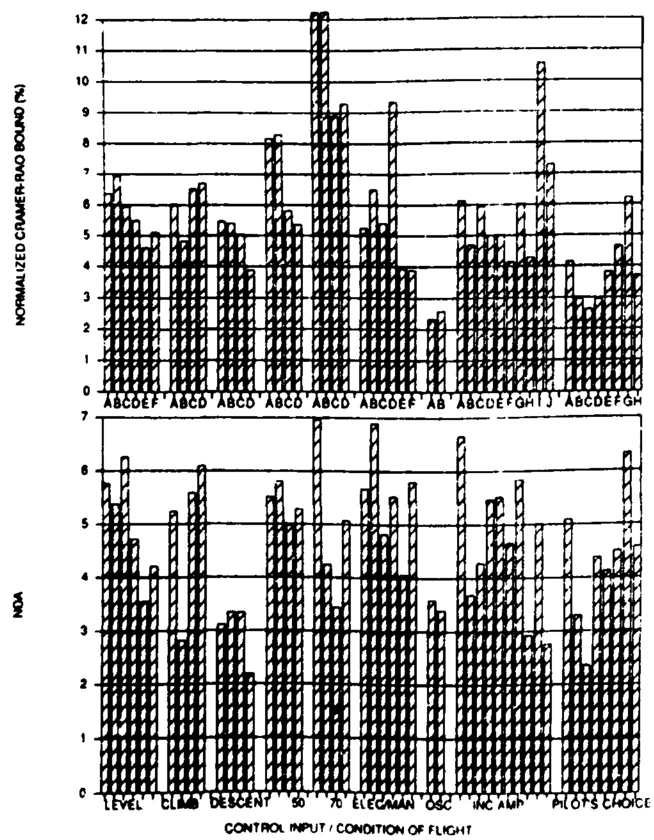
Stability Derivative - Nv



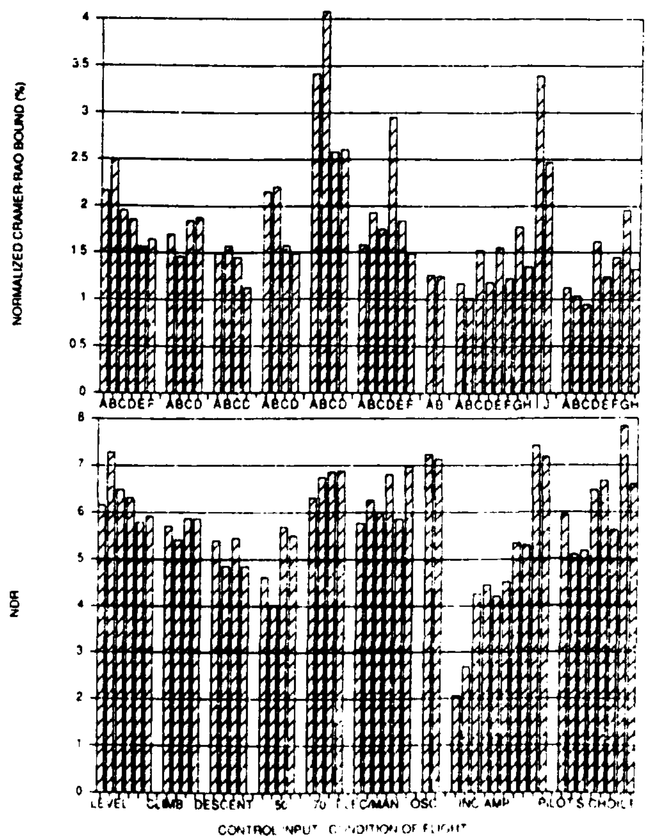
Stability Derivative - Nr



Control Derivative - Nda



Control Derivative - Ndr



THE PLOT OF FOUR TYPICAL N-DERIVATIVES

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4 The Application of Linear Maximum Likelihood Estimation of Aerodynamic Derivatives for the Bell-205 and Bell-206

AUTHOR(S)/AUTEUR(S)

J.H. deLeeuw, K. Hui

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SUMMARY/SOMMAIRE

Parameter identification from flight test data of fixed-wing aircraft is currently a common procedure for application to aircraft development work, validation of simulation, flight simulator verification, flight control systems synthesis, aircraft handling qualities, flight envelope expansion and airplane certification. Similar work on the identification of the more complex helicopter system is currently still in the research stage. This report describes a number of flight test experiments involving the application of parameter estimation techniques to helicopters in order to determine the stability and control derivatives and to obtain information to identify improvements in the structure of the helicopter model. (Canada, translation.)

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