N73-13025

NASA TECHNICAL NOTE

NASA TN D-7114

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EXTRACTION OF LONGITUDINAL AERODYNAMIC COEFFICIENTS FROM FORWARD-FLIGHT CONDITIONS OF A TILT WING V/STOL AIRPLANE

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • DECEMBER 1972

1. Report No. NASA TN D-7114	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle EXTRACTION OF LONGITUDINAL AERODYNAMIC COEFFICIENTS FROM FORWARD-FLIGHT CONDITIONS OF A TILT-WING V/STOL AIRPLANE		5. Report Date December 1972
		6. Performing Organization Code
7. Author(s) James L. Williams		8. Performing Organization Report No. L-8301
		10. Work Unit No. 136-62-02-02
9. Performing Organization Name and Address NASA Langley Research Center Hampton, Va. 23365		11. Contract or Grant No.
		13. Type of Report and Period Covered Technical Note
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

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17. Key Words (Suggested by Author(s)) Maximum likelihood; Parameter estimation; V/STOL airplane; Tilt-wing aircraft		18. Distribution Statement Unclassified – Unlimited	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 24	22. Price* $3.00

*For sale by the National Technical Information Service, Springfield, Virginia 22151

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SUMMARY

A parameter-estimation algorithm was used to extract the longitudinal aerodynamic derivatives from flight data for the XC-142A airplane in a cruise condition. The flight data were the response to a tail-plane doublet input. Results of this study showed that a set of derivatives were determined which yielded a calculated aircraft response in close agreement with the measured response. This calculated response was in much closer agreement with the flight data than the response obtained by using derivatives which were calculated from empirical methods. There were large differences between some of the important derivatives extracted from flight data and those calculated from empirical methods. The reasons for these differences were not identified.

INTRODUCTION

There is currently much interest in the possible application of vertical or short take-off and landing (V/STOL) aircraft for short-range transportation. A necessary requirement for making meaningful studies of the V/STOL flight characteristics is an accurate determination of the vehicle's aerodynamic characteristics. These characteristics can be obtained from theory, empirical data, wind-tunnel tests, and flight tests. Of these methods, extraction of aerodynamic characteristics from flight tests should be the most accurate since these tests are made with full-scale aircraft in the exact flight environment.

The purpose of the present study was to obtain the longitudinal aerodynamic parameters from flight tests of the XC-142A tilt-wing V/STOL airplanes in a cruise condition. The specific method used in extracting the aerodynamic derivatives was a maximum likelihood estimation technique described in reference 1.

SYMBOLS

Values are given in both SI and U.S. Customary Units. The measurements and calculations were made in U.S. Customary Units.

a_X, a_Z linear accelerations, m/sec^2 (ft/sec^2)

C_m pitching-moment coefficient about Y body axis

$C_{m,o}$ trim moment coefficient about Y body axis

$C_{m_q} = \dfrac{\partial C_m}{\partial\left(\dfrac{q\bar{c}}{2V_R}\right)}$, nondimensional

$C_{m_\alpha} = \dfrac{\partial C_m}{\partial \alpha}$ per radian

$C_{m_{\delta_h}} = \dfrac{\partial C_m}{\partial \delta_h}$ per radian

$C_{T,o}$ trim thrust coefficient

$\left(C_{T_\beta}\right)_m$ change in thrust coefficient with respect to main-propeller blade angle

$\left(C_{T_\beta}\right)_t$ change in thrust coefficient with respect to tail-rotor blade angle

C_X force coefficient along X body axis

$C_{X,o}$ trim force coefficient along X body axis

$C_{X_q} = \dfrac{\partial C_X}{\partial\left(\dfrac{q\bar{c}}{2V_R}\right)}$, nondimensional

$C_{X_\alpha} = \dfrac{\partial C_X}{\partial \alpha}$ per radian

C_Z force coefficient along Z body axis

$C_{Z,o}$ trim force coefficient along Z body axis

$C_{Z_\alpha} = \dfrac{\partial C_Z}{\partial \alpha}$ per radian

$C_{Z_{\delta_h}} = \frac{\partial C_Z}{\partial \delta_h}$ per radian

$\bar{c}$	mean aerodynamic chord, m (ft)
D	diameter of main propeller (total of four), m (ft)
D_t	diameter of tail rotor, m (ft)
g	gravitational acceleration, 9.81 m/sec^2 (32.2 ft/sec^2)
I_X, I_Y, I_Z	moment of inertia about the roll, pitch, and yaw axis, kg-m^2 (slug-ft^2)
I_{XZ}	product of inertia, kg-m^2 (slug-ft^2)
i	number of data points where $i = 1, 2, \ldots, N$
i_w	wing incidence angle, deg
J	performance index function
l_t	distance from center of gravity to hub of tail rotor, m (ft)
l_t'	horizontal-tail length
m	mass of test aircraft, kg (lbm)
N	total number of data points
n	percent maximum speed of main propeller
n_t	percent maximum speed of tail rotor
q	angular pitch rate, rad/sec

$R_q = \frac{1}{W_q}$

$R_u = \frac{1}{W_u}$

$R_w = \frac{1}{W_w}$

S	wing area, m^2 (ft^2)
T	transpose of a matrix
u	velocity component along X body axis, m/sec (ft/sec)
V_m	measured resultant velocity, m/sec (ft/sec)
V_R	resultant velocity, m/sec (ft/sec)
W_q, W_u, W_w	weight value of q, u, and w state variables
w	velocity component along body axis, m/sec (ft/sec)
X	vector describing state of aircraft
z_{cg}	vertical distance from center of gravity to line of thrust applications, m (ft)
α	angle of attack $\left(\tan^{-1} \frac{w}{u}\right)$, deg or rad
α_o	trim angle of attack, deg or rad
β_L	blade angle, left outboard main propeller, deg or rad
β_L^o	initial blade angle, left outboard main propeller, deg or rad
β_R	blade angle, right outboard main propeller, deg or rad
β_R^o	initial blade angle, right outboard main propeller, deg or rad
β_t	blade angle of tail rotor, deg or rad
δ_h	horizontal-tail deflection, positive when trailing edge is down, deg or rad
θ	pitch attitude angle, deg or rad
ρ	mass density of air, kg/m^3 $(slugs/ft^3)$

Subscripts:

c calculated value

i inertial

m measured value; also main propeller

o trim conditions

Dot over a symbol denotes rate of change of parameter with respect to time.

DESCRIPTION OF AIRPLANE, FLIGHT TESTS, AND DATA PROCESSING

Description of Airplane

The airplane for which the aerodynamic parameters were determined is the XC-142A V/STOL experimental airplane. This airplane is a four-propeller, tilt-wing vehicle with a tail rotor to supplement the all-movable horizontal tail in the low-speed flight regime. The wing can be tilted from an angle of 0° to 90° relative to the fuselage center line. Figure 1 is a photograph of the airplane. Pertinent mass and geometric characteristics of the airplane are given in table I.

Flight instrumentation pertinent to this study included the following items:

(1) Pitch-rate gyro
(2) Angle-of-attack indicator
(3) Altimeter
(4) Total velocity indicator
(5) Control-surface-position indicator
(6) Magnetic tape recorders

Flight Tests

The data which are used in the present study were obtained from flights conducted at the Langley Research Center as part of an evaluation program of the XC-142A aircraft. In the tests of this study, the wing tilt was 3°, the flap was deflected about 38°, and the tail rotor was locked.

The flight velocity was 61.73 m/sec (120 knots) and the air density was 1.1802 kg/m^3 (0.00229 slug/ft^3). The control input applied to the aircraft was approximately a pitch doublet, as shown in figure 2. The aircraft response is shown as figure 3. All tests were made with the stability augmentation system in continuous operation.

Data Processing

The flight data that are used in this study were obtained from in-flight measurements of the aircraft response to a tail-plane input. The outputs of the various measuring instruments were electrical voltages. These outputs were conducted to a 90-channel commutator and the voltages were sampled and recorded at a rate of 10 times per second. The tape records were digitized, multiplied by the proper calibration constants, and recorded in engineering units.

Since the data used in this study were commutated, there existed at least a 0.001-second time difference between data points of the measured quantities. Linear interpolation between data points of each measured quantity was used to obtain data at common times for each quantity. The basic data were also corrected for instrument bias and displacement of measuring instruments from the aircraft center of gravity.

EXTRACTION OF AERODYNAMIC PARAMETERS

The procedure used in extracting aerodynamic parameters from the flight data is basically the maximum likelihood technique of reference 1. A flow chart depicting the steps involved in the extraction program is given as figure 4. Initial values of the state variables were obtained from the flight records for the time period just prior to a control input. Initial estimates of the aerodynamic derivatives were obtained by use of reference 2 and are listed in table II.

Equations of Motion and Auxiliary Equations

The equations of motion used to model the aircraft were as follows:

$$\dot{u} = -qw - g \sin \theta + \frac{1}{2} \frac{\rho}{m} V_R^2 S \left[C_{X,o} + C_{X_\alpha}\left(\alpha - \alpha_o\right)\right]$$

$$+ \frac{\rho n^2 D^4}{m} \left\{4C_{T,o} + 2\left(C_{T_\beta}\right)_m \left[\left(\beta_L - \beta_L^o\right) + \left(\beta_R - \beta_R^o\right)\right]\right\} \cos i_W \tag{1}$$

$$\dot{w} = qu + g \cos \theta + \frac{1}{2} \frac{\rho}{m} V_R^2 S \left[C_{Z,o} + C_{Z_\alpha}(\alpha - \alpha_o) + C_{Z_{\delta_h}}(\delta_h - \delta_{h,o})\right]$$

$$- \frac{\rho}{m} n_t^2 D_t^4 \left[\left(C_{T_\beta}\right)_t \beta_t\right] - \frac{\rho n^2 D^4}{m} \left\{4C_{T,o} + 2\left(C_{T_\beta}\right)_m \left[\left(\beta_L - \beta_L^o\right) + \left(\beta_R - \beta_R^o\right)\right]\right\} \sin i_w \qquad (2)$$

$$\dot{q} = \frac{1}{2} \rho V_R^2 \frac{S\bar{c}}{I_Y} \left[C_{m,o} + C_{m_\alpha}(\alpha - \alpha_o) + C_{m_q} \frac{q\bar{c}}{2V_R} + C_{m_{\delta_h}}(\delta_h - \delta_{h,o})\right]$$

$$- \frac{\rho n_t^2 D_t^4 l_t}{I_Y} \left(C_{T_\beta}\right)_t \beta_t + \frac{\rho n^2 D^4}{I_Y} \left\{4C_{T,o} + 2\left(C_{T_\beta}\right)_m \left[\left(\beta_L - \beta_L^o\right) + \left(\beta_R - \beta_R^o\right)\right]\right\} z_{cg} \qquad (3)$$

The auxiliary equations used were

$$\dot{\theta} = q$$

$$V_R = \left(u^2 + v^2 + w^2\right)^{1/2}$$

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

The propeller blade angles were measured in flight and these values were used in the equations of motion.

Derivative Extraction Procedure

The parameter extraction procedure used in this study is an iterative technique which utilizes the maximum likelihood method to estimate the stability and control parameters. This method uses the likelihood function which, when maximized, provides the following information:

(1) The parameter changes which are used to update the parameter

(2) The covariance matrices whose elements are proportional to the estimated standard deviations and the pairwise correlation coefficients for the parameters and the states

(3) The performance index function J which is an indicator of the fit between measured and calculated motions

Details of the method are given in reference 1.

The iterative technique produces a set of estimated derivatives which, when used in the equation of motion, provide the best fit to the time variation of the aircraft motion measured in flight. The criterion for the best fit is the performance index J which is defined as

$$J = \det\left[\frac{1}{N}\sum_{i=1}^{N}\left(X_{i,m} - X_{i,c}\right)\left(X_{i,m} - X_{i,c}\right)^{T}\right]$$

where det means determinant and X is the vector describing the state of the aircraft. Generally, the performance index J becomes smaller with successive iterations. The iteration procedure is stopped when the value of J does not change appreciably for several successive iterations. The components of the vector X are the state variables u, w, and q. The linear accelerations a_X, a_Z, and the angle θ were not included in the X vector and hence were not included in the performance index function J. The quantities u_m and w_m were not measured directly but were obtained from the measured total velocity and angle of attack through the use of equations

$$u_m = V_m \cos \alpha_m$$

$$w_m = V_m \sin \alpha_m$$

Maximization of the likelihood function yields the covariance matrix for the measurement noise based on the current nominal solution. (See ref. 1.) The inverse of this matrix is the weighting matrix used in the parameter change equations. In this investigation, the diagonal form of the weighting matrix was used and the diagonal elements can be expressed as the squares of the difference between the measured and calculated data. For example, the weight for the state variable u is expressed as

$$\frac{1}{R_u^2} = \frac{1}{N}\sum_{i=1}^{N}\left(u_m - u_c\right)_i^2$$

Similar equations are obtained for $1/R_w^2$ and $1/R_q^2$.

During the parameter estimation routine, the geometric relationship

$$C_{Z_{\delta_h}} = \frac{\bar{c}}{l_t'} C_{m_{\delta_h}}$$

was maintained in the extraction procedure to estimate $C_{Z_{\delta_h}}$. This constraint equation was used to maintain a proper relationship and to avoid the high degree of correlation between parameters.

RESULTS AND DISCUSSION

The derivatives of the basic aircraft were computed by the methods of reference 2 and were used as initial estimates in the equations of motion. The calculated responses using these derivatives are shown in figure 5, where they are compared with the measured flight data. This figure shows that some appreciable differences exist between calculated and measured states over sections of the 8-second duration of the flight record.

The derivative extraction was allowed to iterate until the cost function reached a minimum. At that time, the computed aircraft responses were in much better agreement with flight data. (See fig. 6.) The extracted derivatives corresponding to the computed responses of figure 6 are given in table III. Also shown are the residuals (incremental changes in extracted derivatives, measured from the previous iteration), and the estimated standard deviations at convergence. The small magnitude of the residuals is a further indication that the cost function had reached close to a minimum.

The estimated standard deviation is a measure of the uncertainty in the extracted parameter. Column 5 of table III indicates that $C_{X,o}$, C_{X_α}, and C_{X_q} were not very well defined, since their standard deviations are about 15 percent of the magnitude of the extracted parameters. The inaccuracy in these derivatives resulted from the fact that only small changes in forward velocity u of the aircraft occurred after the control input, and these derivatives occur in the equation for forward velocity. (See eq. (1).) The remaining parameters appear to be reasonably well defined, based on their standard deviations.

Correlation Coefficient

The degree of dependency between parameters is indicated by the magnitude of the correlation coefficient. A pairwise correlation coefficient of ±1 implies a linear relationship (linear dependency) between parameters. In the extraction routine, this statement means that one parameter can be replaced by a linear function of the other. This condition leads to a uniqueness problem in the determination of the derivatives, since any linear combination of values would produce the same estimated aircraft response.

A matrix of correlation coefficients is computed as part of the parameter extraction program and is presented in table IV for the extracted parameters listed in table III.

An examination of table IV shows that no exact linear pairwise relationships existed between the extracted parameters (correlation coefficient of ±1). There are several pairs of parameters, however, that have high correlation values, that is, coefficients greater than 0.8. The more important of these derivatives (for stability) are the coefficients C_{m_α} and C_{Z_α}. Because of the magnitude of the correlation coefficient, C_{m_α} and C_{Z_α} were examined to determine whether these derivatives could be related linearly. The method used in this study was to fix C_{m_α} at several values and then to obtain the corresponding converged value of C_{Z_α} and the magnitude of the performance index for each combination. If C_{m_α} and C_{Z_α} are truly linearly related, a plot of C_{m_α} against C_{Z_α} should be a straight line and the performance index J should have about the same magnitude for all cases. Calculations were made for three fixed values of C_{m_α}, and the results of these computations showed that C_{m_α} was not linearly dependent on C_{Z_α}. Therefore, the extracted derivatives presented in table III formed a consistent set.

Comparison of Initial Estimated and Extracted Derivatives

One of the uses of derivatives extracted from flight is to assess the accuracy of analytical and empirical methods of estimating aerodynamic derivatives. If the extracted derivatives for the XC-142A are presumed to be correct, since they permitted accurate estimates of the aircraft response, it is of interest to compare these derivatives with the derivatives estimated by using the methods of reference 2. Such a comparison is shown in table III. (Compare columns 2 and 3.) As noted previously, the longitudinal parameters $C_{X,o}$, C_{X_α}, and C_{X_q} are not well defined and, therefore, will not be discussed. Of the remaining derivatives, large differences are noted between computed and extracted values of C_{m_α} and the effective damping parameter C_{m_q} (the computed value shown for C_{m_q} also contains an estimated contribution of $C_{m_{\dot{\alpha}}}$). The estimated value of C_{m_α} is considerably smaller than the extracted value. The smaller value of C_{m_α} (that is, smaller than the estimated value) required to match flight time histories was also noted in references 3 and 4. The reasons for these differences have not been established.

CONCLUDING REMARKS

A parameter-estimation algorithm was used to extract the longitudinal aerodynamic derivatives from flight data for the XC-142A airplane in a cruise condition. The flight data were the response to a tail-plane doublet input. Results of this study showed that a set of derivatives were determined which yielded a calculated aircraft response in close agreement with the measured response. This calculated response was in much closer agreement with the flight data than the response obtained by using derivatives which were calculated from empirical methods. There were large differences between some of the important derivatives extracted from flight data and those calculated from empirical methods. The reasons for these differences were not identified.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., November 20, 1972.

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3. Suit, W. T.: Aerodynamic Parameters of the Navion Airplane Extracted From Flight Data. NASA TN D-6643, 1972.
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TABLE I.- GEOMETRIC AND MASS CHARACTERISTICS OF AIRCRAFT

Aircraft mass, kg (lbm)	16 352 (36 050)
Fuselage:	
Length, m (ft)	15.24 (50.0)
Wing:	
Area, m^2 (ft^2)	49.65 (534.4)
Aspect ratio	8.53
Span, m (ft)	20.60 (67.6)
Mean aerodynamic chord, m (ft)	2.46 (8.07)
Vertical tail:	
Area, m^2 (ft^2)	12.08 (130.0)
Aspect ratio	1.87
Span, m (ft)	4.75 (15.6)
Tail length, center of gravity to 0.25 mean aerodynamic chord, m (ft)	6.52 (21.39)
Horizontal tail:	
Area, m^2 (ft^2)	15.19 (163.5)
Aspect ratio	5.68
Span, m (ft)	9.48 (31.1)
Tail length, center of gravity to 0.25 mean aerodynamic chord, m (ft)	7.56 (24.8)
Propellers:	
Main propeller:	
Number of blades	4
Diameter, m (ft)	4.75 (15.6)
Tail rotor:	
Number of blades	3
Diameter, m (ft)	2.50 (8.2)
Moment arm, wing pivot to rotor center, m (ft)	9.78 (32.1)
Moments of inertia:	
I_X, kg-m^2 (slug-ft^2)	203 373 (150 000)
I_Y, kg-m^2 (slug-ft^2)	173 545 (128 000)
I_Z, kg-m^2 (slug-ft^2)	366 071 (270 000)
I_{XZ}, kg-m^2 (slug-ft^2)	10 846 (8000)

TABLE II.- STARTING VALUES FOR SELECTED PARAMETERS

C_{X_q}	0.000
C_{m_q}	-31.0
C_{m_α}	-1.50
$C_{m_{\delta_h}}$	-3.12
C_{Z_α}	-4.30
$C_{X,o}$	0.0178
C_{X_α}	-0.29
$C_{Z,o}$	-1.327
$C_{Z_{\delta_h}}$	-1.10
$C_{m,o}$	0.0429
$C_{T,o}$	0.008
$\left(C_{T_\beta}\right)_m$	0.774

TABLE III. - COMPARISON OF THE STARTING AND EXTRACTED PARAMETERS AT CONVERGENCE BASED ON 78 DATA POINTS

Derivatives	Starting values (from table II)	Extracted values	$\Delta\alpha$ residual	Estimate of standard deviation
$C_{X,o}$	0.0178	0.0096	-2.0×10^{-8}	0.0013
C_{X_α}	*-.29	.4608	2.77×10^{-6}	.0734
C_{X_q}	.000	16.50	-1.98×10^{-5}	2.84
$C_{Z,o}$	-1.327	-1.27	-5.57×10^{-8}	.009
C_{Z_α}	*-4.30	-5.600	-6.50×10^{-7}	.353
$C_{m,o}$	.0429	.033	5.82×10^{-8}	.001
C_{m_α}	*-1.50	-.489	-6.16×10^{-7}	.036
$C_{m_q} + C_{m_{\dot{\alpha}}}$	*-31.00	-54.848	1.35×10^{-4}	1.765
$C_{m_{\delta_h}}$	*-3.12	-4.396	1.55×10^{-5}	.088
$C_{Z_{\delta_h}}$	-1.10	-1.432	-----------	-----

*Values calculated from reference 2.

TABLE IV.- MATRIX OF CORRELATION COEFFICIENTS FOR EXTRACTED PARAMETERS AT CONVERGENCE (78 DATA POINTS)

	$C_{X,o}$	C_{X_α}	C_{X_q}	$C_{Z,o}$	C_{Z_α}	$C_{m,o}$	C_{m_α}	C_{m_q}	$C_{m_{\delta_h}}$
$C_{X,o}$	1	-0.3985	0.3596	-0.6627	0.6722	0.6493	-0.6360	-0.2161	0.2817
C_{X_α}	-.3985	1	-.5569	-.0574	-.1841	.1456	.1038	.1486	-.0018
C_{X_q}	.3596	-.5569	1	-.4204	.5556	.3941	-.5023	-.2261	.1370
$C_{Z,o}$	-.6627	-.0574	-.4204	1	-.9422	-.8825	.7699	.4095	-.2380
C_{Z_α}	.6722	-.1841	.5556	-.9422	1	.8357	-.8132	-.4349	.2443
$C_{m,o}$	.6493	.1456	.3941	-.8825	.8357	1	-.8520	-.3727	.1721
C_{m_α}	-.6360	.1038	-.5023	.7699	-.8132	-.8520	1	.0221	-.4906
C_{m_q}	-.2161	.1486	-.2261	.4095	-.4349	-.3727	.0221	1	.6772
$C_{m_{\delta_h}}$	.2817	-.0018	.1370	-.2380	.2443	.1721	-.4906	.6772	1

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Figure 1.- Photograph of vehicle used in this investigation.

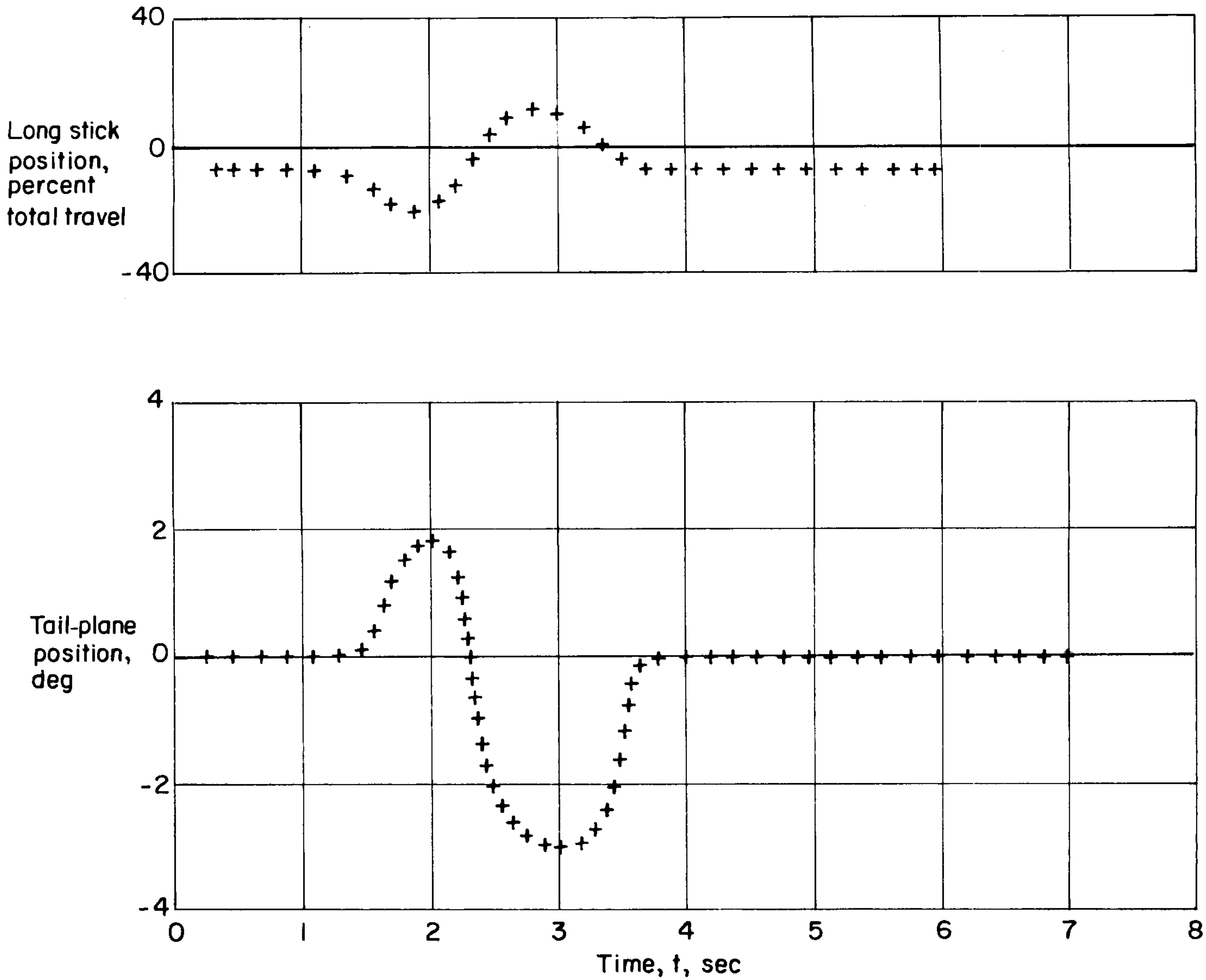


Figure 2.- Typical pitch control input for longitudinal motion.

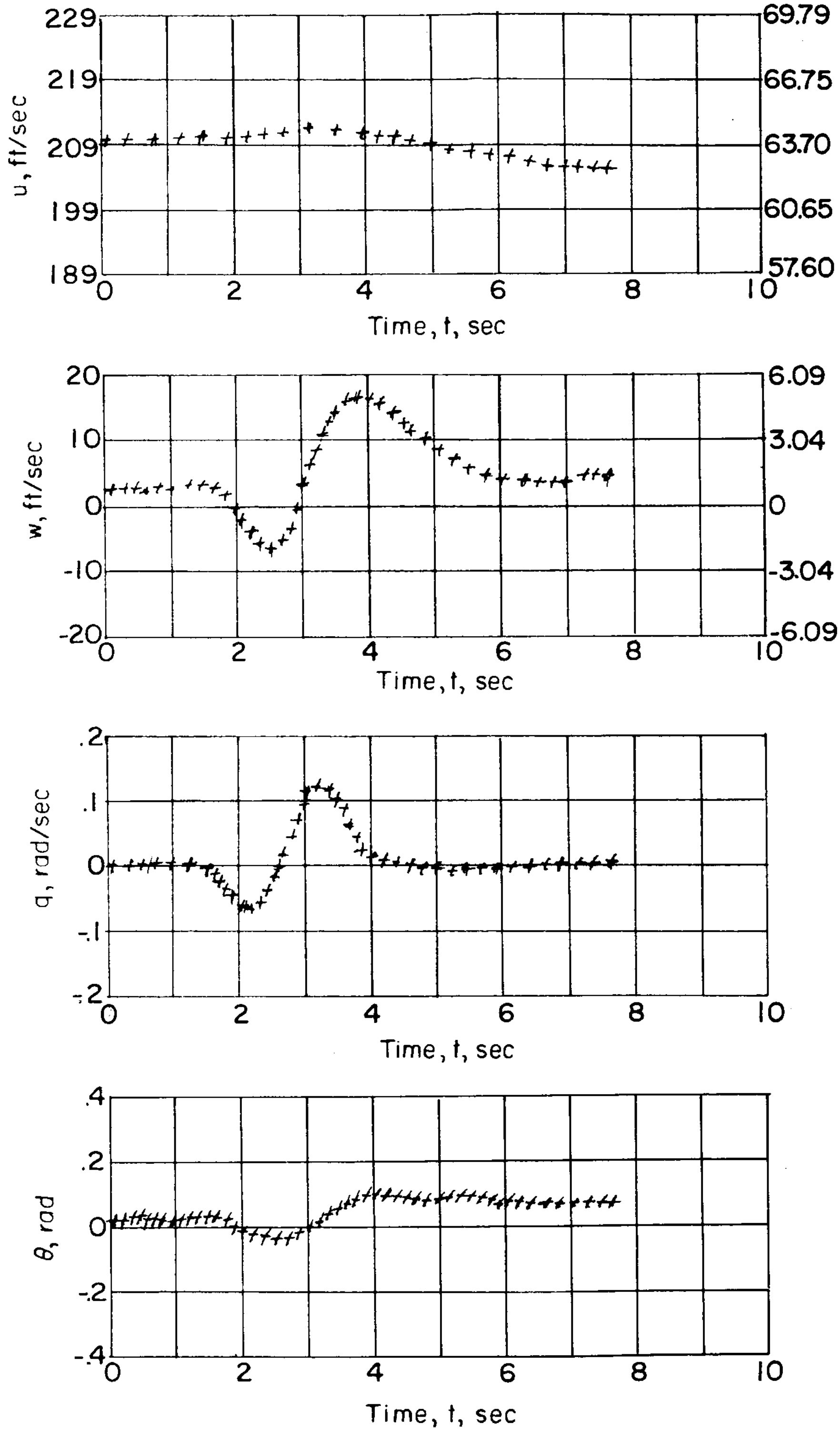


Figure 3.- Aircraft response to pitch control.

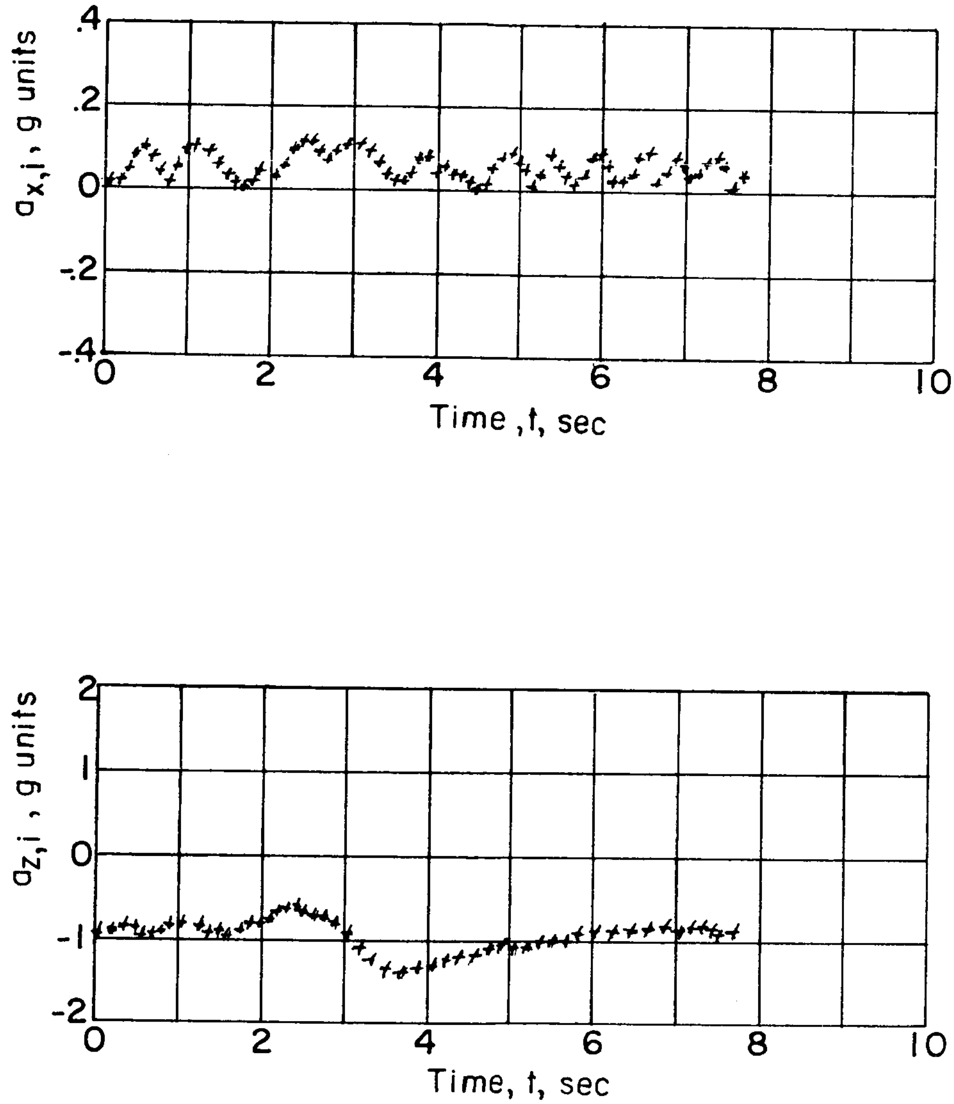


Figure 3.- Concluded.

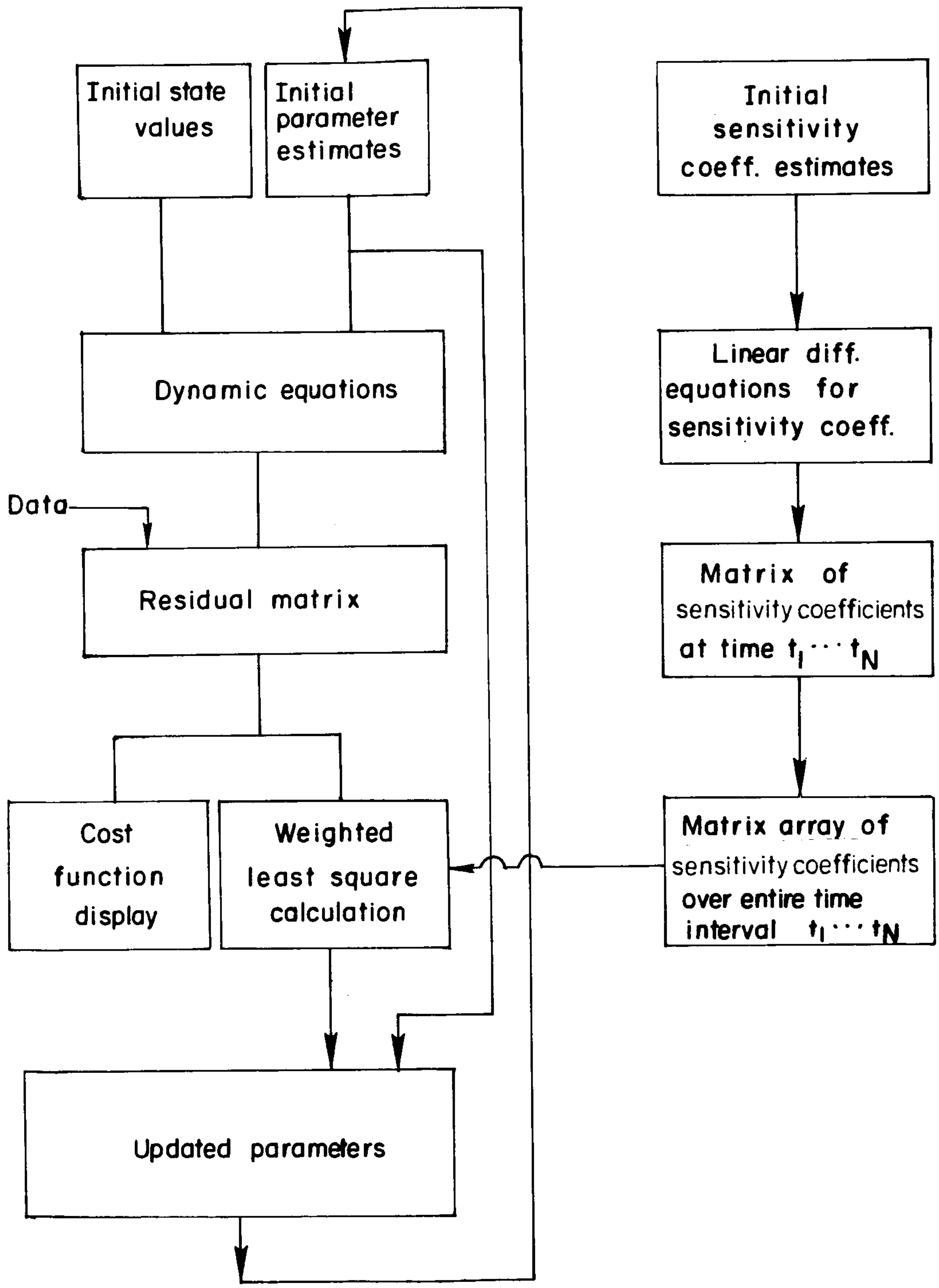


Figure 4.- Flow chart for parameter-extraction technique.

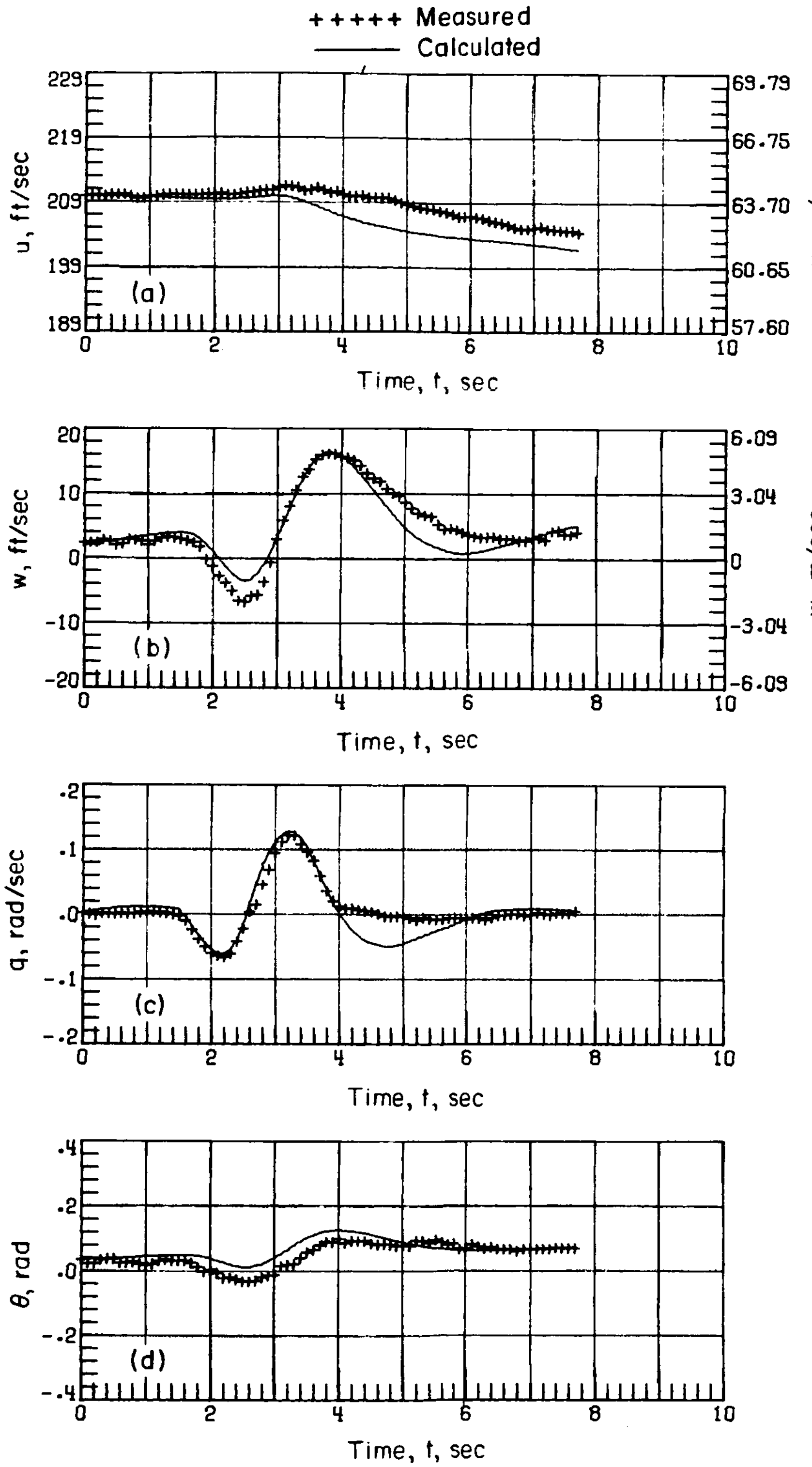
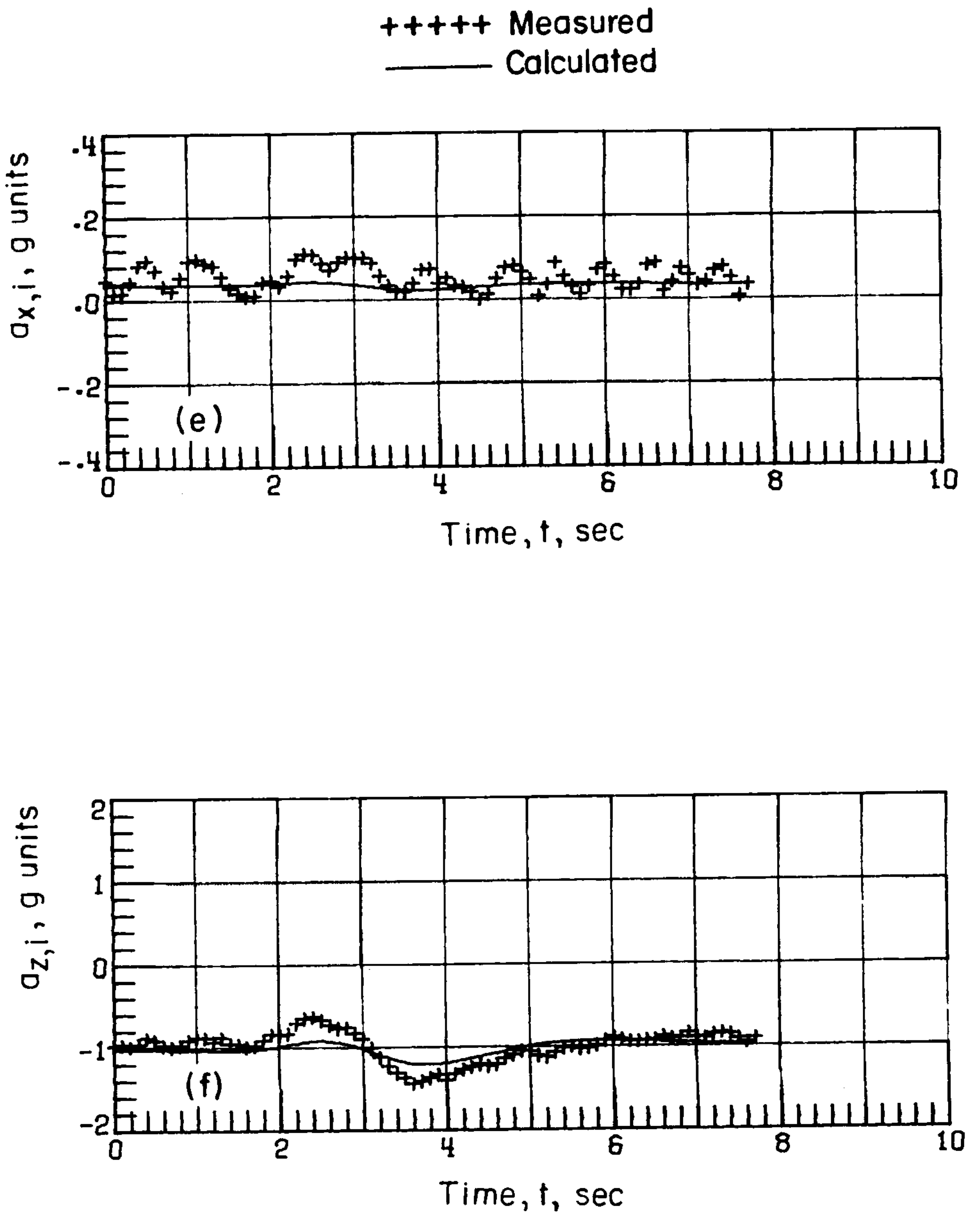


Figure 5.- Comparison of calculated and measured response. Calculated response based on initial estimated parameters.



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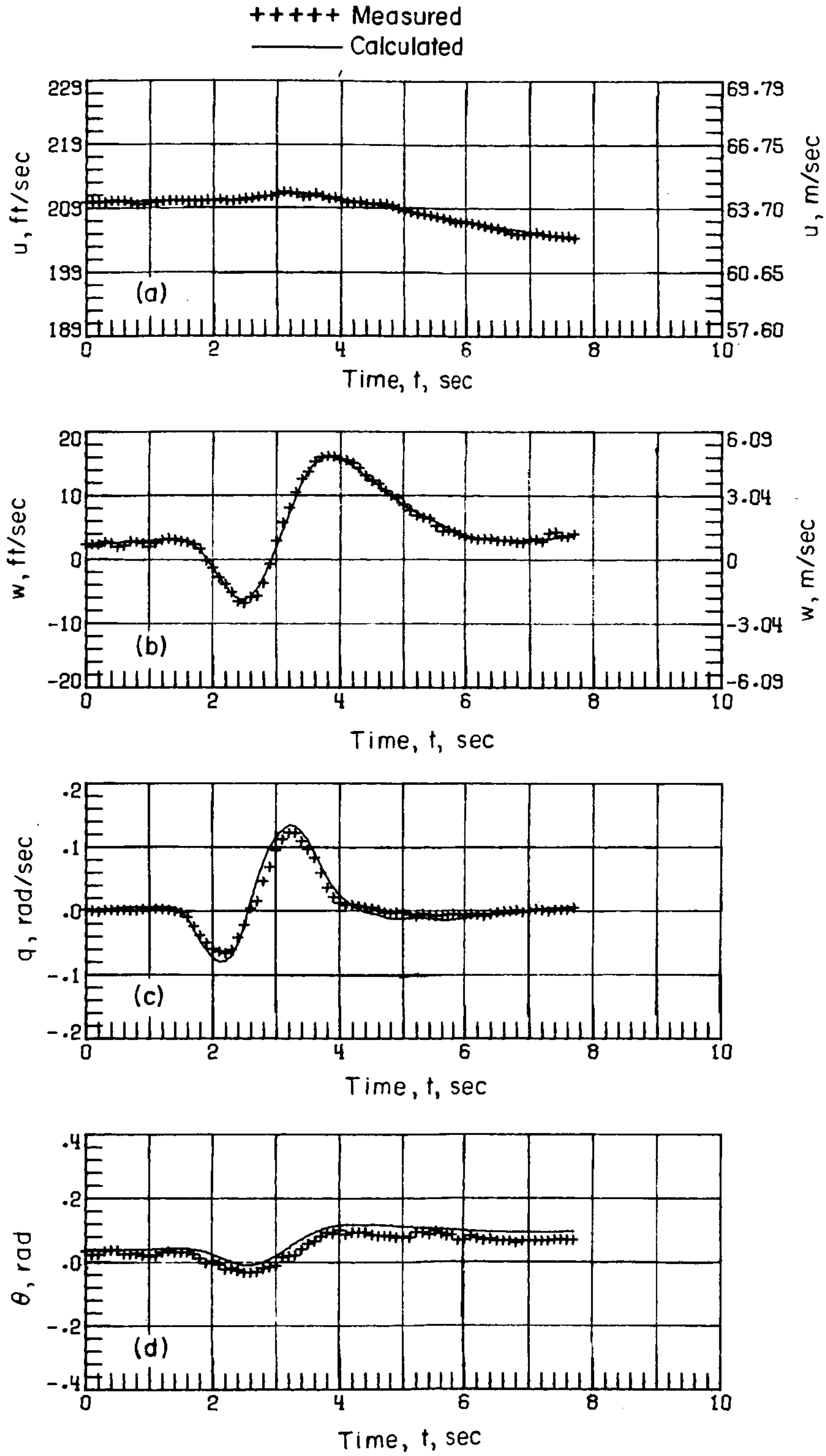

Figure 6.- Comparison of calculated and measured response at convergence.

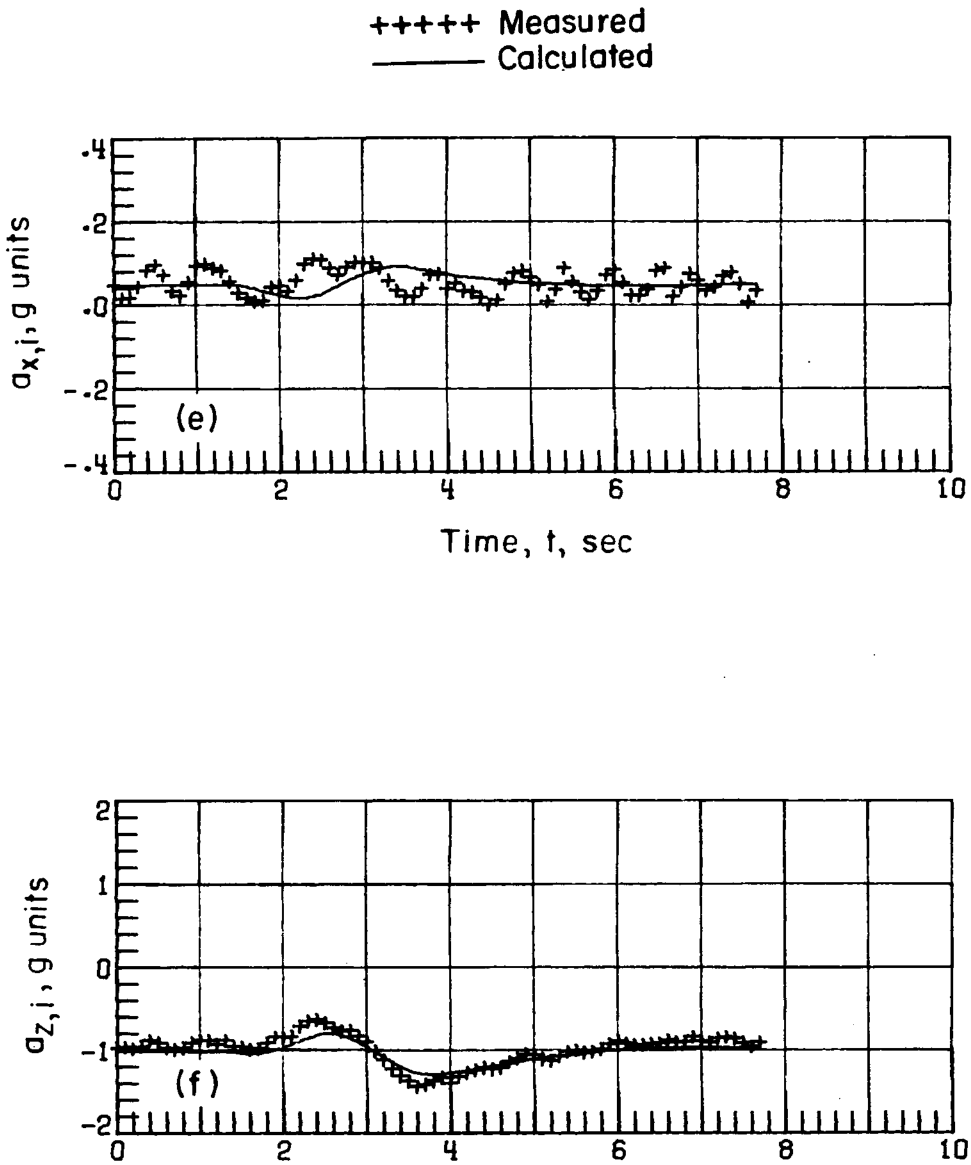

Figure 6.- Concluded.

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