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EXTRACTION FROM FLIGHT DATA OF LONGITUDINAL AERODYNAMIC COEFFICIENTS IN MANEUVERING FLIGHT FOR F-8C AIRCRAFT

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Flight-test data have been used to extract the longitudinal aerodynamic parameters of the F-8C aircraft. The aircraft was trimmed in a steady turn at angles of attack of approximately 9° and 13° at Mach numbers of 0.7 and 0.8. The parameters extracted resulted in a good match to the flight data and the values obtained were reasonable. The values were further verified by comparing the period and time to damp to half-amplitude, as calculated by using the extracted parameter values, with the period and time to damp to half-amplitude actually measured from the flight data traces. These results show that for the set of data examined, a mathematical model using linear aerodynamics was adequate to describe the response motions at the test angles of attack.

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SUMMARY

Flight-test data have been used to extract the longitudinal aerodynamic parameters of the F-8C aircraft at moderate to high angles of attack. The data obtained were from perturbations of the aircraft from steady turns, with trim normal accelerations from 1.5g to 3.0g. The angle-of-attack variation from trim was approximately $\pm 4^{\circ}$ and maximum angle of attack reached during the tests was approximately 16°.

Although wind-tunnel data indicate that the lift and pitching moments are somewhat nonlinear with angle of attack, the linear aerodynamics extracted from the flight tests did permit computation of motion time histories which were in close agreement with the measured time histories. The aerodynamic parameters extracted from flight data were in reasonably good agreement with other data available for a 1g condition.

INTRODUCTION

The National Aeronautics and Space Administration is currently involved in research on fly-by-wire control systems for aircraft. A fly-by-wire system replaces a direct-link mechanical control system with an electronic system to activate controls. Signals from the pilot go to a digital control system which commands the positions of the controls. These commands may be modified by a stability augmentation system. However, to design an augmentation system, the aerodynamic parameters of the aircraft must be known. The aircraft presently used by NASA to study digital fly-by-wire systems is an F-8C aircraft. The F-8C used as a test bed for digital fly-by-wire systems had a standard airframe. The aerodynamics available for the subject aircraft were determined primarily from wind-tunnel and analytical results and the mathematical model of the aircraft was considered to be reasonable, especially for trimmed level flight. To substantiate the existing model for the F-8C at moderate to high angles of attack, some flight tests were made with the angle of attack as high as 16°.

A maximum-likelihood extraction procedure was used to examine the flight data. In this procedure, a set of equations of motion is used to calculate aircraft response to specified control inputs. Initial estimates of aerodynamic parameters (from theory or from wind-tunnel tests) are used for the initial motion computations. The maximum-likelihood extraction program then iterates on the aerodynamic parameters to select a set that maximizes a conditional likelihood function. This program has been used to determine the aerodynamic parameters for several aircraft in the 1g trimmed flight condition. (See refs. 1 to 3.) The details of the program are contained in reference 4. The program has not been used previously at the Langley Research Center for flight data taken at moderate to high angles of attack where the trimmed load factor is greater than 1g.

The purpose of this paper is to present the longitudinal aerodynamic parameters of the F-8C aircraft as obtained from flight data flown at Mach numbers 0.7 and 0.8 and in trim with a load factor of 1.5g to 3.0g. The equations used and additional information on the flight data are presented and are followed by some comments on the extraction procedure and a discussion of the results of the study.

SYMBOLS

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

a	acceleration, g units
$\bar{c}$	wing mean geometric chord, m (ft)
g	acceleration due to gravity, g units
h	altitude, m (ft)
I	moment of inertia, kg-m^2 (slug-ft^2)
l_t	distance from aircraft center of gravity to mean aerodynamic chord of horizontal tail, m (ft)
M	Mach number
P	period of oscillatory motion, sec
p	rate of roll, rad/sec or deg/sec

q	rate of pitch, rad/sec or deg/sec
r	rate of yaw, rad/sec or deg/sec
S	wing area, m^2 (ft^2)
$t_{1/2}$	time to damp to half-amplitude, sec
u	component of velocity along X body axis, m/sec (ft/sec)
V	aircraft total velocity, m/sec (ft/sec)
v	component of velocity along Y body axis, m/sec (ft/sec)
W_t	aircraft weight, N (lb)
w	component of velocity aong Z body axis, m/sec (ft/sec)
X_i	ith component of state vector X
α	angle of attack, rad or deg
δ_e	tail-plane deflection, positive for trailing edge down, rad or deg
ζ	damping ratio
θ	pitch angle, rad or deg
ρ	air density, kg/m^3 ($slugs/ft^3$)
φ	bank angle, rad

Coefficients:

C_m	pitching-moment coefficient
C_X	axial-force coefficient, positive along X body axis direction
C_Z	normal-force coefficient, positive along Z body axis direction

$C_{Z_\alpha} = \frac{\partial C_Z}{\partial \alpha}$ $\qquad$ $C_{m_{\delta_e}} = \frac{\partial C_m}{\partial \delta_e}$

$C_{m_\alpha} = \frac{\partial C_m}{\partial \alpha}$ $\qquad$ $C_{Z_q} = \frac{\partial C_Z}{\partial \frac{q\bar{c}}{2V}}$

$C_{X_\alpha} = \frac{\partial C_X}{\partial \alpha}$ $\qquad$ $C_{Z_{\delta_e}} = \frac{\partial C_Z}{\partial \delta_e}$

$C_{m_q} = \frac{\partial C_m}{\partial \frac{q\bar{c}}{2V}}$ $\qquad$ $C_{m_{\dot{\alpha}}} = \frac{\partial C_m}{\partial \frac{\dot{\alpha}\bar{c}}{2V}}$

Subscripts:

c	computed
e	tail plane
f	measured flight
o	indicates coefficient at trim conditions
t	indicates state at trim conditions
X,Y,Z	body-coordinate axes

A dot over a symbol signifies a derivative with respect to time.

Symbols used on computer plots:

ALPHA	$= \alpha - \alpha_t$, rad
AXI	acceleration along X body axis, g units
AZI	acceleration along Z body axis, g units
DE	$= \delta_e - \delta_{e,t}$, rad
G	acceleration due to gravity, 32.2 ft/sec^2 (9.8 m/sec^2)

Q	rate of pitch, rad/sec
THETA	pitch angle, rad
U	velocity along X body axis, m/sec (ft/sec)
V	velocity along Y body axis, m/sec (ft/sec)
W	velocity along Z body axis, m/sec (ft/sec)

EQUATIONS OF MOTION

The equations of motion used in this study are referred to a body-axis system shown in figure 1 and are as follows:

X-force:

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

Z-force:

$$\dot{w} = qu - pv + g \cos \theta \cos \varphi + \frac{1}{2}\rho V^2 S \frac{g}{W_t}\left[C_{Z,o} + C_{Z_\alpha}(\alpha - \alpha_t)\right.$$

$$\left. + C_{Z_{\delta_e}}(\delta_e - \delta_{e,t}) + C_{Z_q}\frac{q\bar{c}}{2V}\right]$$

Pitching moment:

$$\dot{q} = \frac{I_Z - I_X}{I_Y} pr + \frac{I_{XZ}}{I_Y}(r^2 - p^2) + \frac{1}{2}\rho \frac{V^2 S\bar{c}}{I_Y}\left[C_{m,o} + C_{m_\alpha}(\alpha - \alpha_t)\right.$$

$$\left. + C_{m_q}\frac{q\bar{c}}{2V} + C_{m_{\dot{\alpha}}}\frac{\dot{\alpha}\bar{c}}{2V} + C_{m_{\delta_e}}(\delta_e - \delta_{e,t})\right]$$

$$a_X = \frac{1}{g}(\dot{u} + qw - rv + g \sin \theta)$$

$$a_Z = \frac{1}{g}(\dot{w} + pv - qu - g \cos \theta \cos \varphi)$$

$$\dot{\theta} = q \cos \varphi - r \sin \varphi$$

FLIGHT DATA

Description of Airplane

The subject aircraft is a modified prototype F-8C and has been a flight-test vehicle since its manufacture in 1958. This aircraft is a single-seat high-performance airplane with a single jet engine embedded in the fuselage. Pitch control is achieved with a unit horizontal tail. The center of gravity used was at 29.0-percent mean geometric chord. The X body axis was parallel to and 10.16 cm (4 in.) above water line 100. (See fig. 1.) The geometric characteristics of the aircraft are given in table I.

Flight Tests

The data used in this report were obtained from flights flown at the NASA Flight Research Center. The pilot was instructed to fly a coordinated turn at nominal Mach numbers of 0.7 and 0.8 with nominal trim angles of attack of 9° and 13°. As can be seen from figures 2 to 5, the sideslip angle was less than $\pm 2^{\circ}$ at all times and the aircraft returned to the initial trim angle of attack after being perturbed by elevator steps and pulses. The angle-of-attack variation from trim was approximately $\pm 4^{\circ}$. The aircraft stability augmentation systems were off during the tests. The actual test conditions for each individual run are given in table II. The aircraft weights and inertias listed in table II were obtained from tables supplied by NASA Flight Research Center and these weights and inertias were calculated as a function of the percent fuel in the aircraft. The weights and inertias used are average values for the test duration. Since the weight and inertias varied less than 3 percent and 1 percent, respectively, these variations were not accounted for in the parameter estimation.

The data pertinent to this study, which were recorded during the flight tests, included longitudinal acceleration, a_X; normal acceleration, a_n; the difference between total pressure and static pressure as measured on a nose boom extending 1.83 m (6 ft) in front of the airplane; pitch attitude, θ; bank angle, φ; pitch rate, q; yaw rate, r; roll rate, p; indicated angle of attack, α and indicated angle of sideslip, β, as measured by vanes on the nose boom; pressure altitude; control surface positions (aileron δ_a, elevator δ_e, and rudder δ_r); and time, t. The full-scale range of the instruments is given in table III. All the data were stored on an onboard magnetic tape recorder by using a pulse control modulation (PCM) recording system. Additional information on the data acquisition system can be found in reference 4.

Data Preparation

The data were initially recorded, digitized, and calibrated at NASA Flight Research Center. A digital tape with the data in engineering units was sent to Langley Research

Center. The accelerometers were corrected for instrument location. The difference in the measured total pressure and static pressure was assumed to be the dynamic pressure. Density was determined from the standard atmosphere tables for the measured pressure altitude and the airspeed was calculated. The indicated α and β were corrected for the effects of aircraft angular rates by using the $0.29\bar{c}$ center of gravity. The linear velocities along the vehicle body axes were calculated from the airspeed, angle of attack, and angle of sideslip. All the data were put on a tape at a rate of 20 points per second. The tape was then ready for use in the extraction program.

PARAMETER-ESTIMATION PROCEDURE

The parameter-estimation procedure used in this study is an iterative procedure which maximizes the conditional likelihood function L (aerodynamic parameters, weights, and initial conditions):

$$L = \frac{1}{(2\pi)^{1/2}|R|^{1/2}} \exp\left[-\frac{1}{2}\sum_{i=1}^{N}\left(X_{i,f} - X_{i,c}\right)^T R^{-1}\left(X_{i,f} - X_{i,c}\right)\right]$$

where R is the estimate of the error covariance matrix and X is the vector describing the state of the aircraft. The states used in the likelihood function were u, w, q, θ, a_Z, and a_X. The calculated states $X_{i,c}$ were determined by using the equations of motion previously introduced. In these equations the lateral quantities p, r, v, and φ were input directly into the equations from the flight data tape. The weighting matrix which is R^{-1} can be obtained from the complete error covariance matrix, the diagonal terms of the error covariance matrix, or a diagonal matrix with fixed weights on the diagonal, at the discretion of the investigator. The diagonal form was used in this investigation and therefore, the weights represent the estimated lower bound of the noise on the measured states. The use of the likelihood function in parameter identification is discussed in reference 5. Maximizing the likelihood function results in a parameter-updated equation which is given by

$$\Delta P = \left(M^T R^{-1} M\right)^{-1} M^T X_i$$

where M is the matrix of sensitivities of the calculated states with respect to the unknown parameters, T denotes the transpose and -1 denotes the inverse. (See ref. 5.) The matrix is the estimated parameter covariance matrix. The updated equation is determined by forming a set of differential equations with the changes in the unknown parameters as the variables. This set of simultaneous equations is then solved by the least-squares method to give the updated equation. (See ref. 5.)

After the convergence of the likelihood function, for a given flight data record, the extracted aerodynamic derivatives were examined. The derivatives were accepted as well determined if (a) the standard deviations of the computed time histories of the aircraft motion from the measured time histories were less than 3 percent of the full-scale range of the instrument used to measure the quantity examined, (b) the changes in the derivatives were less than one-hundredth of the derivative value for successive iterations, and (c) the estimated standard deviation of each derivative was less than about one-tenth of the extracted value of the derivative.

Past experience has indicated a very high correlation between C_{m_q} and $C_{m_{\dot{\alpha}}}$. To eliminate this correlation, a value for $C_{m_{\dot{\alpha}}}$ was chosen that was similar to values of $C_{m_{\dot{\alpha}}}$ that were determined in reference 6. The value chosen was -0.6 and $C_{m_{\dot{\alpha}}}$ was held fixed at this value for all the runs. Also the geometric relation $C_{Z_{\delta_e}} = C_{m_{\delta_e}} \frac{\bar{c}}{l_t}$ was assumed and so $C_{Z_{\delta_e}}$ was not extracted but was calculated from the extracted value of $C_{m_{\delta_e}}$.

RESULTS AND DISCUSSION

Data for the flight conditions listed in table II were used with the model given to determine iteratively a set of aerodynamic derivatives for each of the flight conditions. The measured and computed time histories for each flight condition are shown in figures 2 to 5. The measured data are represented by dotted lines in the figures. The computed time histories shown are those attained after the differences between the measured and calculated trajectories become constant. The figures show that in all cases the computed time histories were generally in close agreement with the flight records. Table IV gives the standard deviations of the computed states from the measured states. The inverses of the quantities in table IV were the diagonal terms used in the weighting matrix when the fits to the flight data shown in figures 2 to 5 were obtained. The standard deviations of the individual fits can be seen to be less than 3 percent of full scale, which was the uncertainty in the measured data (see table III); and in many cases, the standard deviations were less than 1 percent of the full-scale measured quantity.

The derivatives extracted for each flight condition (the derivatives which resulted in the computed time histories of figs. 2 to 5) are listed in table V along with their estimated standard deviations. The correlation matrices for the aerodynamic parameters extracted are shown in table VI. An examination of table V indicates that the numbers extracted seem to be reasonable. The estimated variances of the parameters were less than 10 percent of the values extracted; thus confidence in the values obtained was indicated.

For comparison, values of some aerodynamic coefficients obtained from reference 6 are also shown in table V. The numbers given are for an altitude of 12.19 km (40 000 ft) and were transformed from stability to body axes and converted to a $0.29\bar{c}$ center-of-gravity location.

The wind-tunnel and analytical investigation used for comparison showed a somewhat nonlinear variation with pitching moment, drag force, and lift force with angle of attack for the angle-of-attack range of this investigation. These nonlinearities were not considered severe enough to make a linear aerodynamic model inappropriate. To show this, the curves of pitching moment, drag, force, and lift force plotted against angle of attack were obtained from reference 6 and straight-line approximations were made to these curves at the trim angles of attack for the flight tests. The derivatives estimated by use of these approximate curves were in reasonable agreement with the extracted derivative values.

The results of this investigation seem to substantiate the curves generated from analytical and wind-tunnel studies for the test angle-of-attack range. The results also show that for the flight data examined, a mathematical model using linear aerodynamics was adequate to describe the response motions. To determine how well the numbers from reference 6 would represent the aircraft in the flight conditions of the subject study, the transformed numbers were put into the mathematical model and used to fit the two runs with the normal accelerations closest to 1g. The fits to the data are shown in figures 6 and 7. As can be seen, the fits to the data are not bad, but are not as good as those for the mathematical model determined by the extraction process.

The value of C_{m_q} from reference 6 was different from the value extracted; however, a corresponding difference in the damping between the calculated response using the parameters from reference 6 and the calculated response of the extracted model was not observed. The flight data and extracted model appeared to have about 10 percent more damping than the calculated response using a model based on the parameters of reference 6. In an attempt to get a better comparison of the two models and to substantiate the extracted model, the period and time to damp to half-amplitude of the short-period mode were determined by examining the damping envelope of the flight data and also calculating the period P and time to damp to half-amplitude $t_{1/2}$ by use of the values from table V and equations from reference 7. The values of t, P, and ζ are given in table VII. As can be seen the stability characteristics of the aircraft, as defined by the period and time to damp to half-amplitude for the longitudinal short period mode, are similar for the extracted parameters and for the parameters from reference 6. In this case a simplified analysis has shown that even though the parameter values in the two models, and especially C_{m_q}, are different the response motions they define are similar.

Additional verification of the aerodynamic parameters of table V was obtained by using them to calculate motions, and comparing these calculated motions with motions from additional runs flown at the same time as the test runs but not used during the extraction process. These comparisons are shown in figures 8 and 9. It can be seen that the fits to the additional data are essentially as good as the fits to the data used when parameters were extracted.

CONCLUDING REMARKS

Flight-test data flown at high to moderate angles of attack and using the F-8C airplane has been used to determine the longitudinal aerodynamic parameters of the aircraft at four flight conditions. The tests were conducted with the aircraft trimmed in a steady turn with angles of attack of approximately 9° and 13° and at Mach numbers of 0.7 and 0.8.

The extracted parameters resulted in a good fit to the flight data and the parameter values obtained were in fair agreement with values obtained from wind-tunnel and analytical studies. The extracted aerodynamic model was further verified by comparing the period and time to damp to half-amplitude as calculated by using the extracted parameter values with the period and time to damp to half-amplitude measured from the flight-data traces. The extracted mathematical model was also used to fit data not used during the extraction process and the resulting fit was essentially as good as the fit obtained when the parameters were extracted.

The wind-tunnel and analytical investigations indicated that the nonlinear variation of pitching moment and lift force with angle of attack for the angle-of-attack range of the investigation was not severe enough to make a linear aerodynamic model inappropriate. The results seem to substantiate this conclusion and show that for the flight data examined, a mathematical model using linear aerodynamics was adequate to describe the response motions at the test angles of attack.

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June 10, 1975

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TABLE I.- GEOMETRIC CHARACTERISTICS OF THE F-8C AIRCRAFT

Fuselage:	
Length, m (ft)	16.52 (54.17)
Wing:	
Area, m^2 (ft^2)	34.88 (375)
Aspect ratio	3.4
Span, m (ft)	10.88 (35.67)
Mean geometric chord, m (ft)	3.59 (11.78)
Vertical tail:	
Area, m^2 (ft^2)	10.14 (109)
Aspect ratio	1.5
Span, m (ft)	3.89 (12.75)
Rudder:	
Area, m^2 (ft^2)	1.17 (12.56)
Horizontal tail:	
Area, m^2 (ft^2)	8.68 (93.4)
Aspect ratio	3.5
Span, m (ft)	5.52 (18.1)
Tail length, center of gravity to quarter-chord point of mean geometric chord, m (ft)	5.06 (16.6)

TABLE II.- FLIGHT CONDITIONS

Nominal altitude, m	Nominal altitude, ft	Nominal Mach number	Nominal α_t, deg	Trim elevator, $\delta_{e,t}$, deg	Trim bank angle, φ_t, deg
10 370	34 000	0.71	9.2	-8.13	50
10 370	34 000	.66	13.0	-9.74	60
10 370	34 000	.81	8.2	-7.45	63
10 370	34 000	.78	12.0	-10.3	70

Weight		Inertias							
		I_X		I_Y		I_Z		I_{XZ}	
N	lb	kg-m^2	slug-ft^2	kg-m^2	slug-ft^2	kg-m^2	slug-ft^2	kg-m^2	slug-ft^2
*94 000	21 125	12 500	9200	118 000	86 800	124 000	91 600	4030	2970

*Any errors in assuming nominal conditions were no greater than 3-percent system uncertainty on the measurements themselves.

TABLE III.- INSTRUMENT RANGES

[u and w are calculated from $u = V \cos \alpha$ and $w = V \sin \alpha$. Individual sensors are basically more accurate than 3 percent of full scale; however, because of unknown errors, the effects of incompatibilities between measured states and processing errors the system accuracy was assumed to be 3 percent of the full-scale range of the instrument for the data as used during the extraction procedure.]

State	Range
a_X	± 0.5g
a_Z	-4g to +6g
V	30.91 to 515.15 m/sec 101.34 to 1689.0 ft/sec
θ	$\pm 30^{\circ}$
φ	± 90 rad
q	$\pm 20^{\circ}$/sec
r	$\pm 10^{\circ}$/sec
p	$\pm 40^{\circ}$/sec
α	-5° to $+30^{\circ}$
β	$\pm 20^{\circ}$
h	0 to 21 000 m 0 to 63 000 ft
δ_e	$+7^{\circ}$ to -28°

TABLE IV.- STANDARD DEVIATIONS OF CALCULATED STATES FROM MEASURED STATES AT CONVERGENCE

α_t, deg	M	State							
		u		w		q, deg/sec	θ, deg	a_X, g	a_Z, g
		m/sec	ft/sec	m/sec	ft/sec				
9.2	0.71	0.824	2.7	0.300	0.98	0.28	0.17	0.014	0.028
13	.66	1.98	6.49	.427	1.4	.21	.11	.007	.033
8.2	.81	.619	2.03	.372	1.22	.22	.15	.012	.039
12	.78	.573	1.88	.741	2.43	.62	2.27	.012	.084

TABLE V.- PARAMETER VALUES EXTRACTED (STANDARD DEVIATIONS IN PARENTHESES)

Parameter	Extracted values for –				Values* from reference 5 for –			
	$\alpha_t = 9.2^\circ$ M = 0.71	$\alpha_t = 13^\circ$ M = 0.66	$\alpha_t = 8.2^\circ$ M = 0.81	$\alpha_t = 12^\circ$ M = 0.78	$\alpha_t = 9^\circ$ M = 0.7	$\alpha_t = 13^\circ$ M = 0.7	$\alpha_t = 8^\circ$ M = 0.8	$\alpha_t = 12^\circ$ M = 0.8
$C_{X,o}$	0.067 $(\pm 1.2 \times 10^{-4})$	0.098 $(\pm 1.5 \times 10^{-4})$	0.07 $(\pm 1.1 \times 10^{-4})$	0.12 $(\pm 1.7 \times 10^{-4})$				
C_{X_α}	0.40 (±0.016)	0.284 (±0.0073)	0.40 (±0.025)	0.20 (±0.02)	0.32	0.19	0.30	0.17
$C_{Z,o}$	-0.53 (±0.0016)	-0.68 $(\pm 6.3 \times 10^{-4})$	-0.52 $(\pm 5.6 \times 10^{-4})$	-0.77 $(\pm 5.6 \times 10^{-4})$				
C_{Z_α}	-3.36 (±0.036)	-3.20 (±0.035)	-3.50 (±0.047)	-3.20 (±0.075)	-3.70	-3.40	-3.75	-3.30
C_{Z_q}	-5.0 (-)(Fixed)	-5.0 (-)(Fixed)	-5.0 (-)(Fixed)	-5.0 (-)(Fixed)				
$C_{Z_{\delta_e}}$	-0.65 (-)(Fixed)	-0.60 (-)(Fixed)	-0.64 (-)(Fixed)	-0.60 (-)(Fixed)				
$C_{m,o}$	-0.13 $(\pm 1.7 \times 10^{-4})$	-0.14 $(\pm 8.8 \times 10^{-4})$	-0.11 $(\pm 5.6 \times 10^{-5})$	-0.14 $(\pm 5.6 \times 10^{-5})$				
C_{m_α}	-0.61 (±0.002)	-0.64 (± 0.0015)	-0.57 (±0.0018)	-0.65 (±0.0019)	-0.63	-0.70	-0.64	-0.69
$C_{m_{\dot{\alpha}}}$	-0.6 (-)(Fixed)	-0.6 (-)(Fixed)	-0.6 (-)(Fixed)	-0.6 (-)(Fixed)	-0.52	-0.52	-0.69	-0.69
C_{m_q}	-8.2 (±0.16)	-6.74 (±0.13)	-8.3 (±0.17)	-6.42 (±0.19)	-4.5	-4.5	-4.7	-4.7
$C_{m_{\delta_e}}$	-0.92 (±0.0063)	-0.85 (±0.0051)	-0.906 (±0.006)	-0.84 (±0.006)	-0.88	-0.88	-0.92	-0.92

*Reference values are for an F-8C at 12.19 km (40 000 ft) and with the weight about 5 percent greater and the inertias about 15 percent greater than those of the subject aircraft. They have been transferred from a $0.287\bar{c}$ center of gravity to a $0.29\bar{c}$ center of gravity and from stability axes to body axes.

TABLE VI.- CORRELATION MATRICES

	C_{X_α}	C_{Z_α}	C_{m_α}	C_{m_q}	$C_{m_{\delta_e}}$
Run with $\alpha = 9.2^\circ$, $M = 0.71$					
C_{X_α}	1	-0.058	-0.028	-0.064	-0.080
C_{Z_α}	-.058	1	.18	-.47	-.30
C_{m_α}	-.028	.18	1	-.0022	.26
C_{m_q}	.064	-.47	-.0022	1	.77
$C_{m_{\delta_e}}$	.080	-.30	.26	.77	1
Run with $\alpha = 12^\circ$, $M = 0.78$					
C_{X_α}	1	-0.15	-0.13	0.17	-0.03
C_{Z_α}	-.15	1	.38	-.76	-.08
C_{m_α}	-.13	.38	1	-.27	.17
C_{m_q}	.17	-.76	-.27	1	.53
$C_{m_{\delta_e}}$	-.03	-.08	.17	.53	1
Run with $\alpha = 13^\circ$, $M = 0.66$					
C_{X_α}	1	-0.08	-0.07	0.08	0.11
C_{Z_α}	-.08	1	.29	-.57	-.25
C_{m_α}	-.07	.29	1	-.12	.16
C_{m_q}	.08	-.57	-.12	1	.74
$C_{m_{\delta_e}}$	.11	-.25	.16	.74	1
Run with $\alpha = 8.2^\circ$, $M = 0.81$					
C_{X_α}	1	-0.12	-0.07	0.075	-0.01
C_{Z_α}	-.12	1	.29	-.58	-.20
C_{m_α}	-.07	.29	1	-.2	.1
C_{m_q}	.075	-.58	-.2	1	.73
$C_{m_{\delta_e}}$	-.01	-.20	.1	.73	1

TABLE VII.- PERIODS AND TIME TO DAMP TO HALF-AMPLITUDE

FOR THE SHORT-PERIOD MODES OF THE RUNS MADE

M	α, deg	$t_{1/2,f}$, sec	$t_{1/2,c}$, sec	P_f, sec	P_c, sec	ζ_c	ζ_f
0.71	9.2	1.1	0.96	2.20	2.38	0.260	0.220
.66	13	1.35	1.20	2.30	2.31	.208	.192
.81	8.2	1.20	1.12	2.70	2.62	.250	.240
.78	12	1.40	1.36	2.90	2.75	.218	.217
*.7	9		1.3		2.4	.200	
	13		1.4		2.3	.180	
*.8	8		1.2		2.3	.207	
	12		1.3		2.4	.200	

*From reference 6.

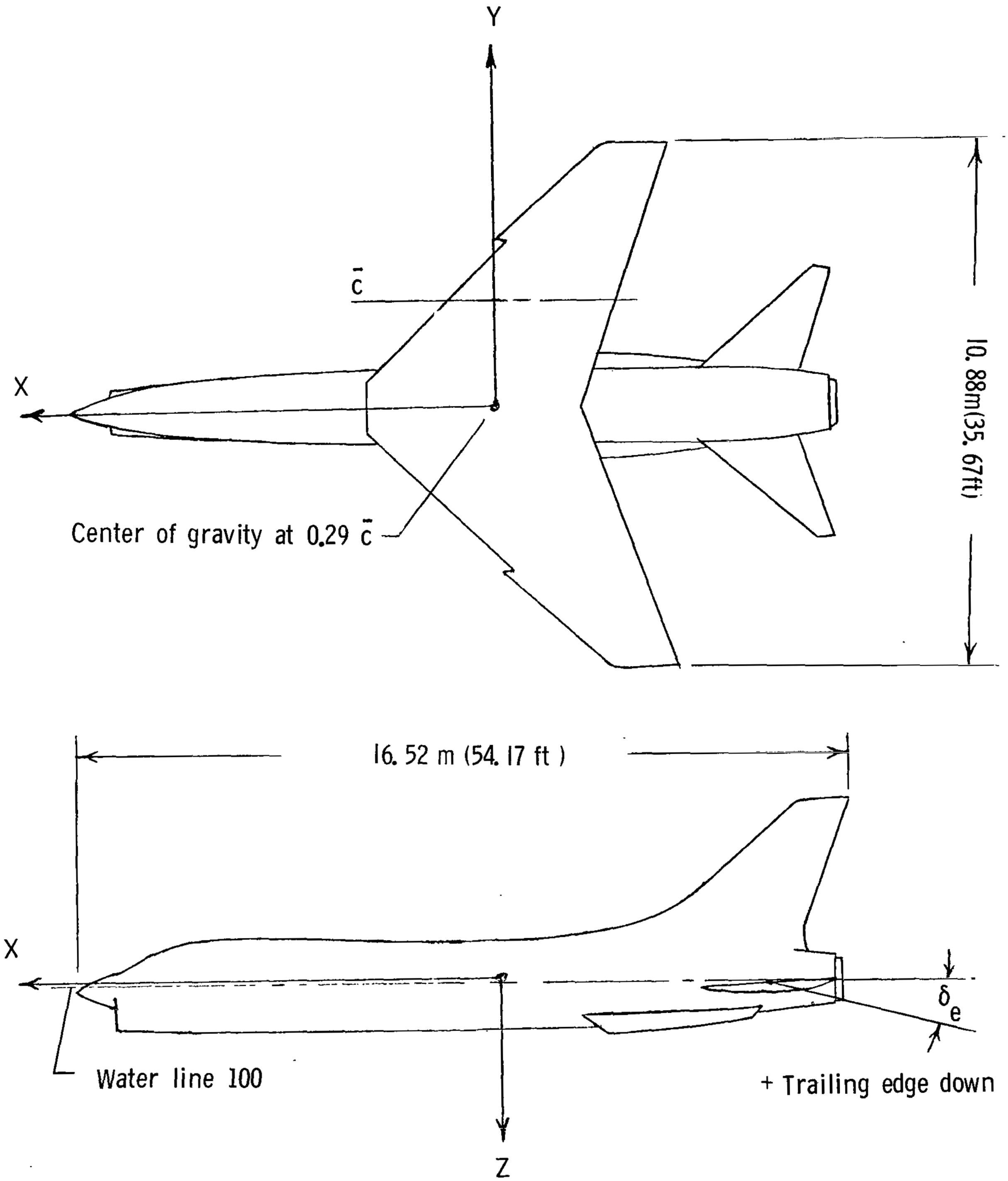

Figure 1.- Sketch of aircraft.

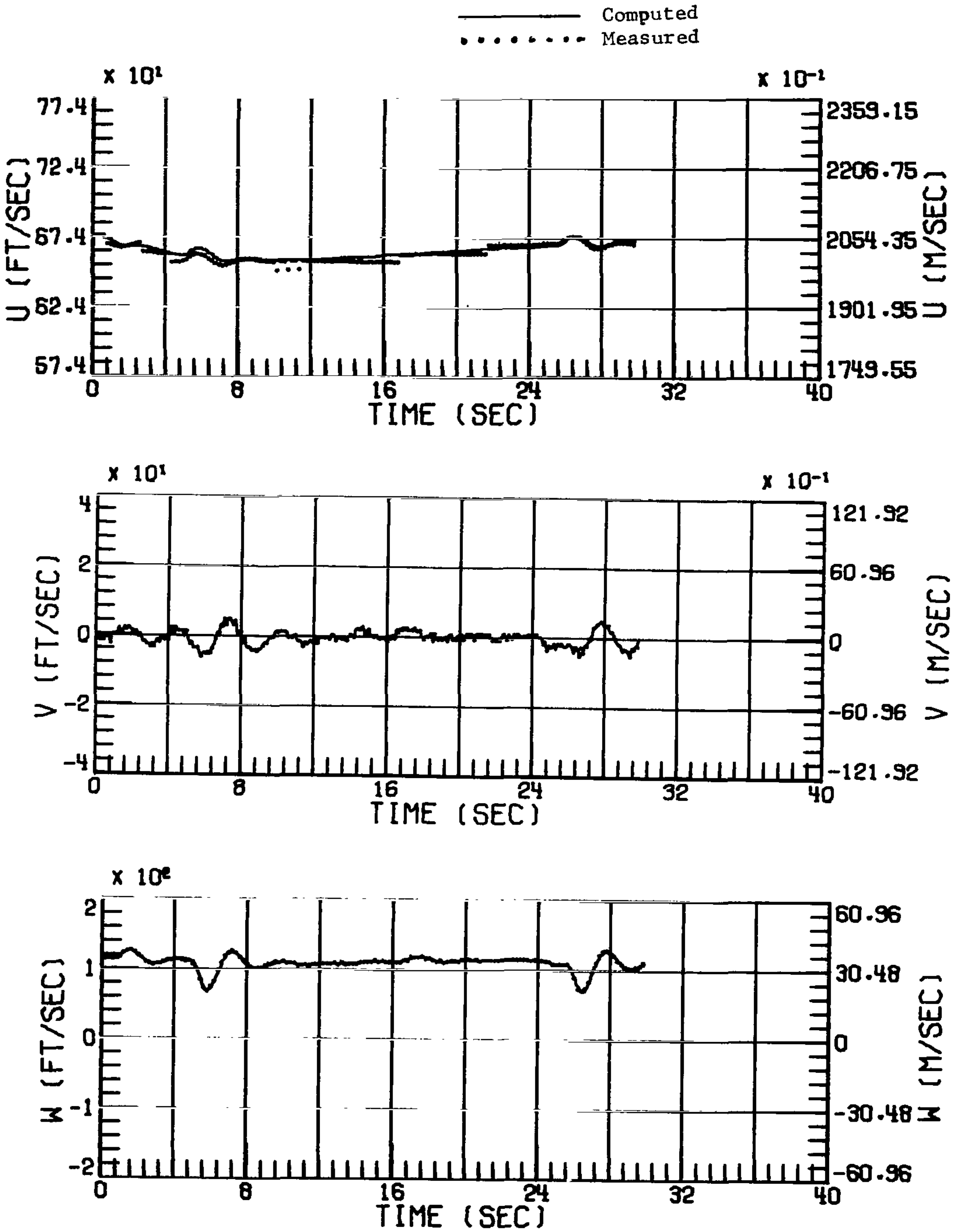

Figure 2.- Comparison of measured data with time histories computed by using parameters given in table V for flight data taken at $M = 0.71$ and $\alpha_t = 9.2^{\circ}$.

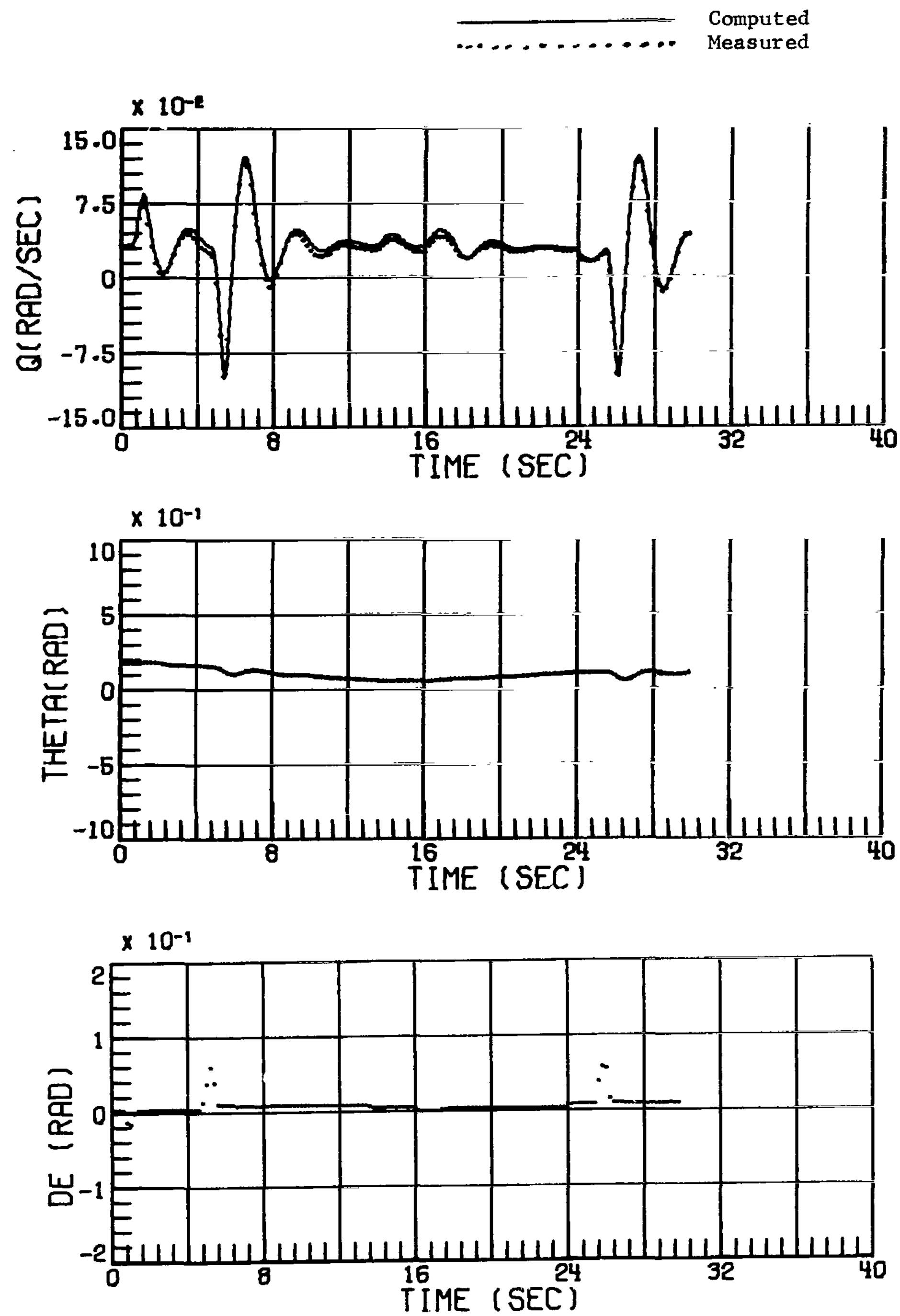

Figure 2.- Continued.

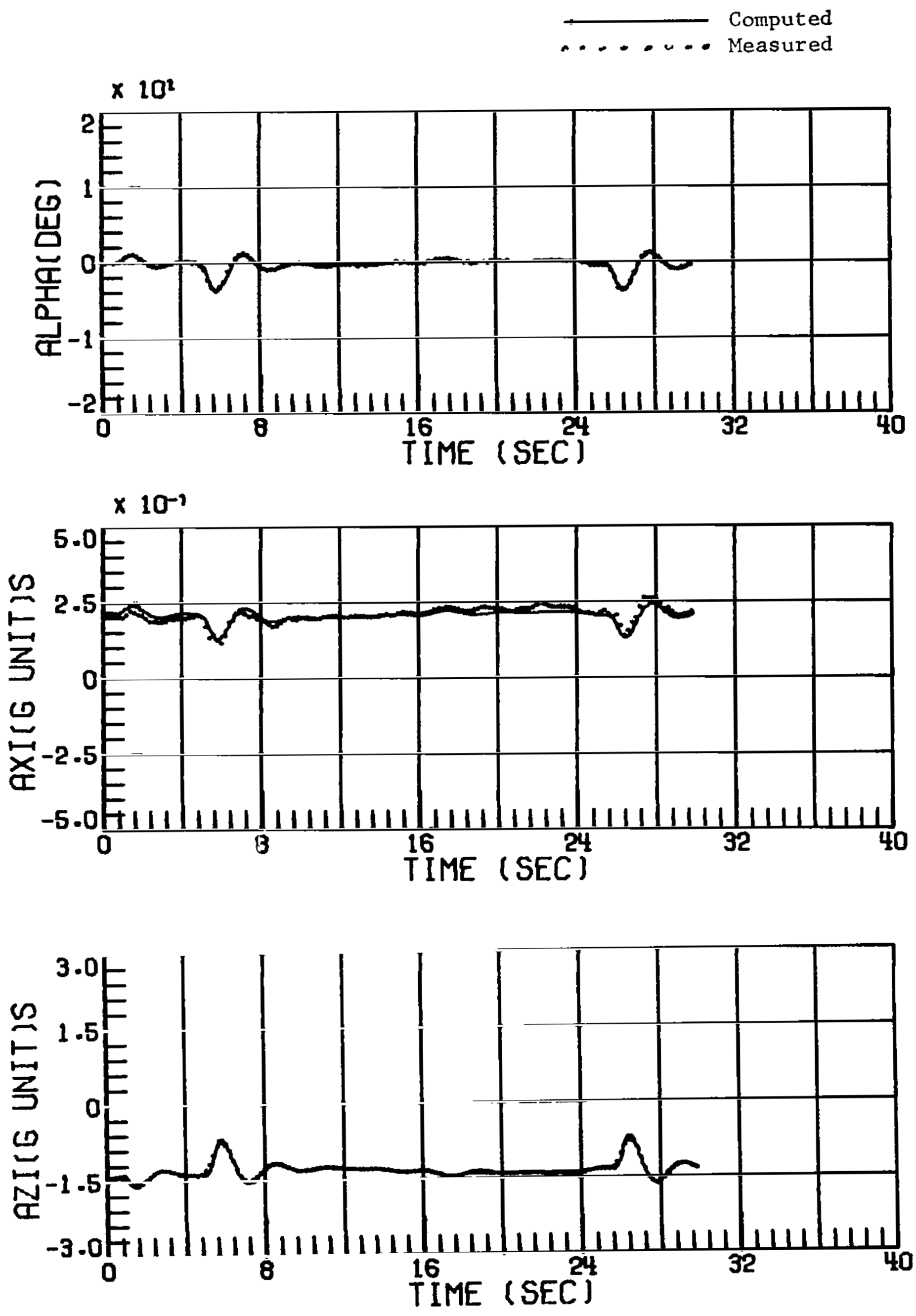

Figure 2.- Concluded.

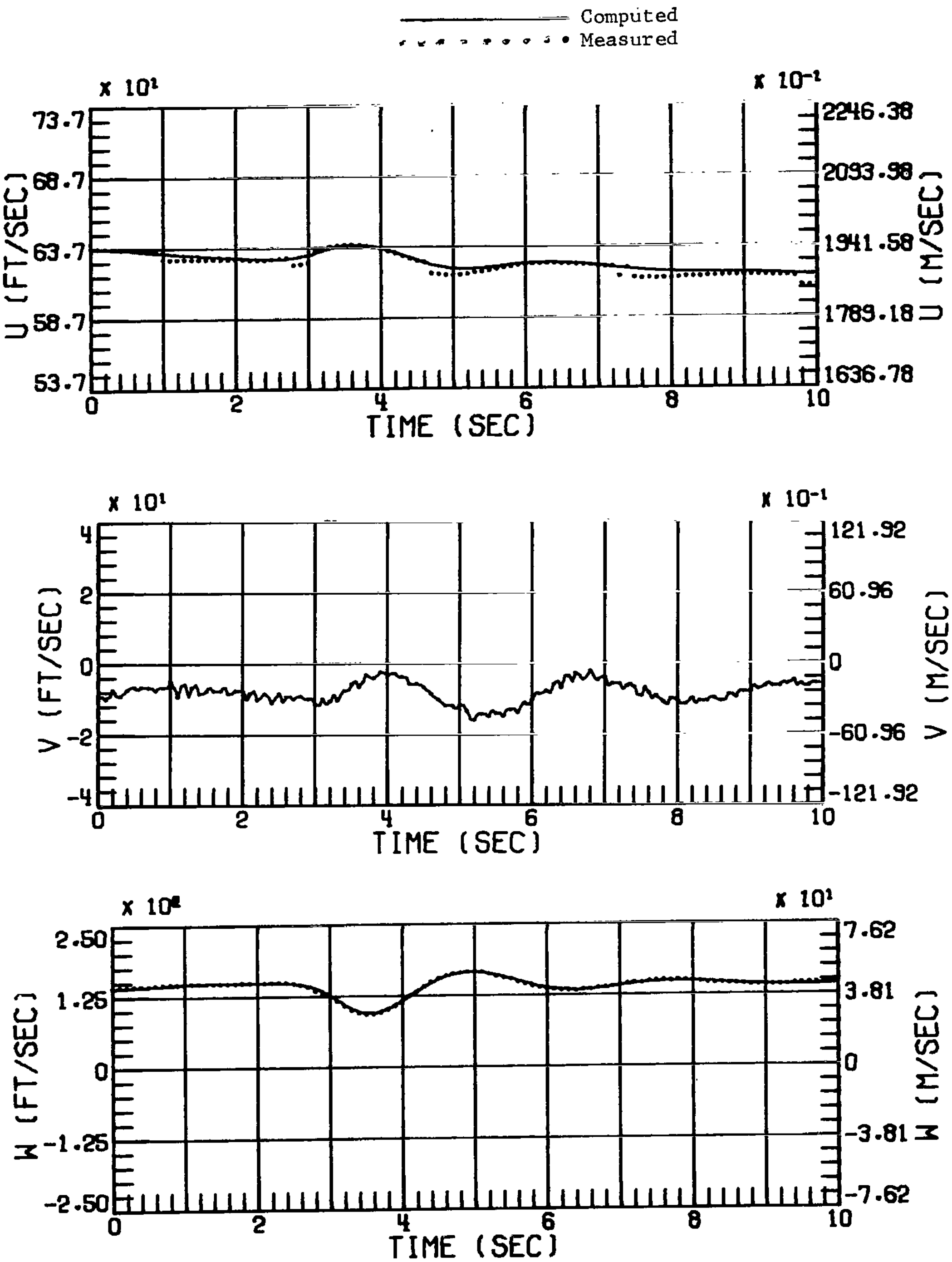

Figure 3.- Comparison of measured data with time histories computed by using parameters given in table V for flight data taken at $M = 0.66$ and $\alpha_t = 13^{\circ}$.

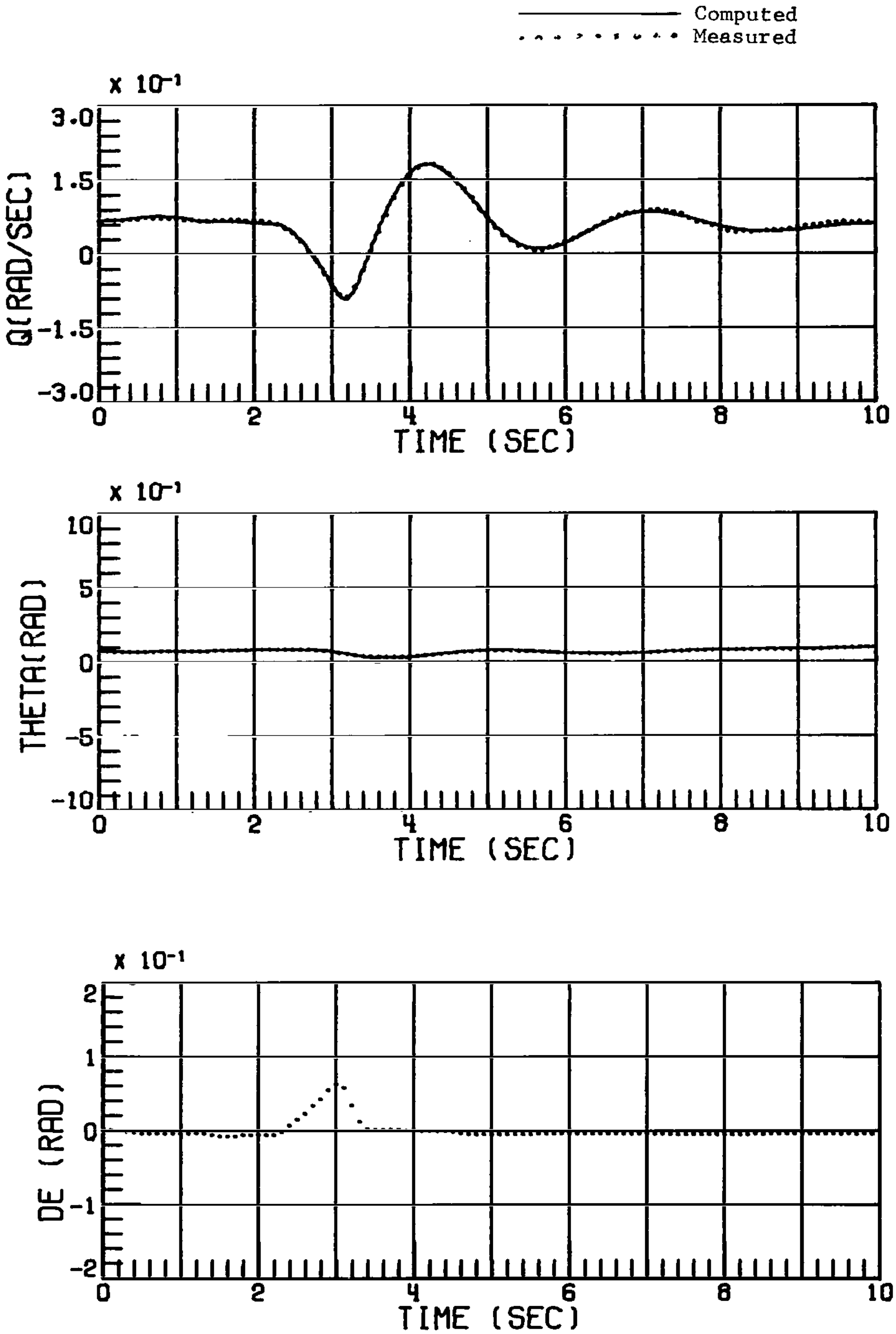

Figure 3.- Continued.

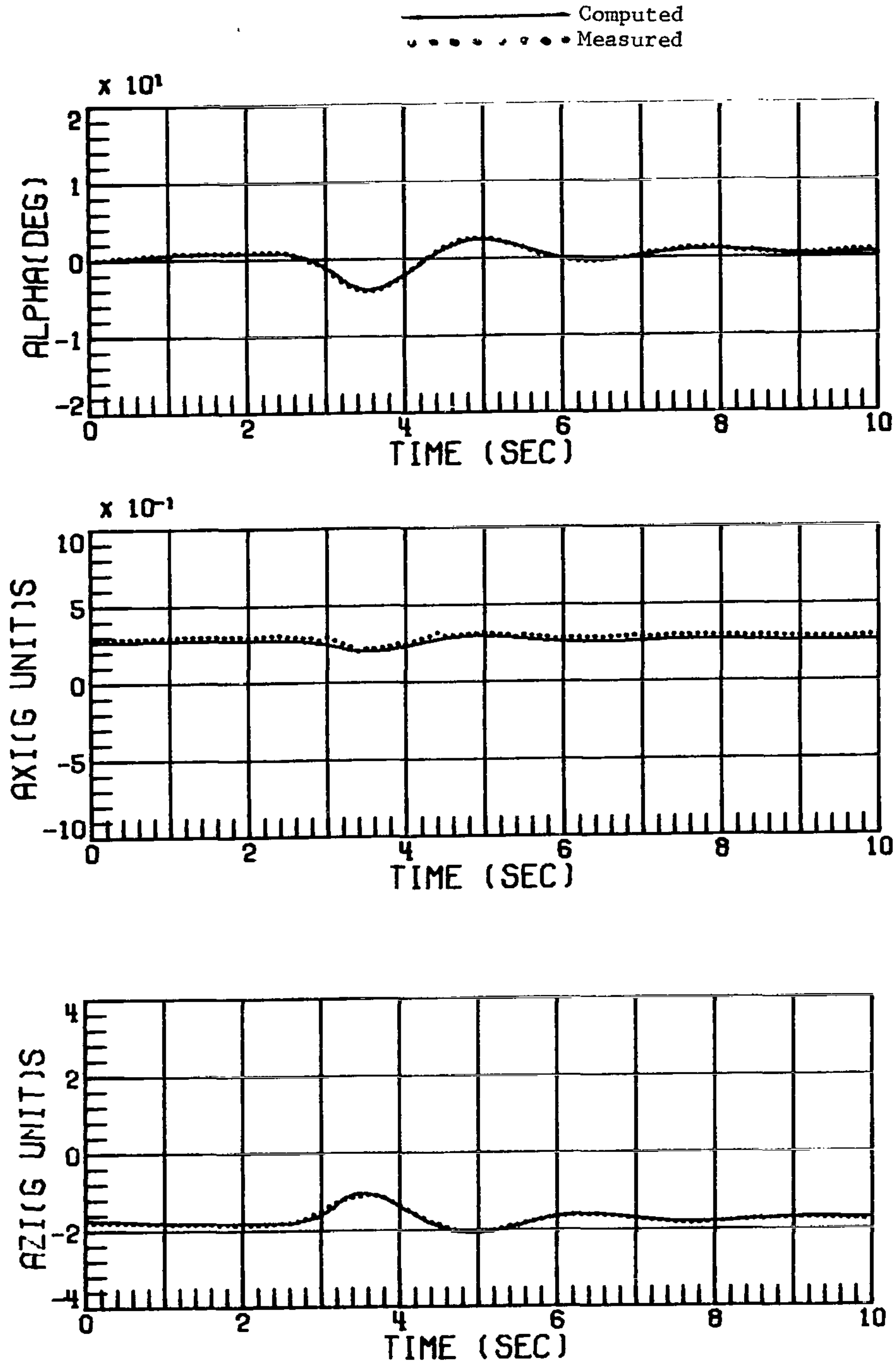

Figure 3.- Concluded.

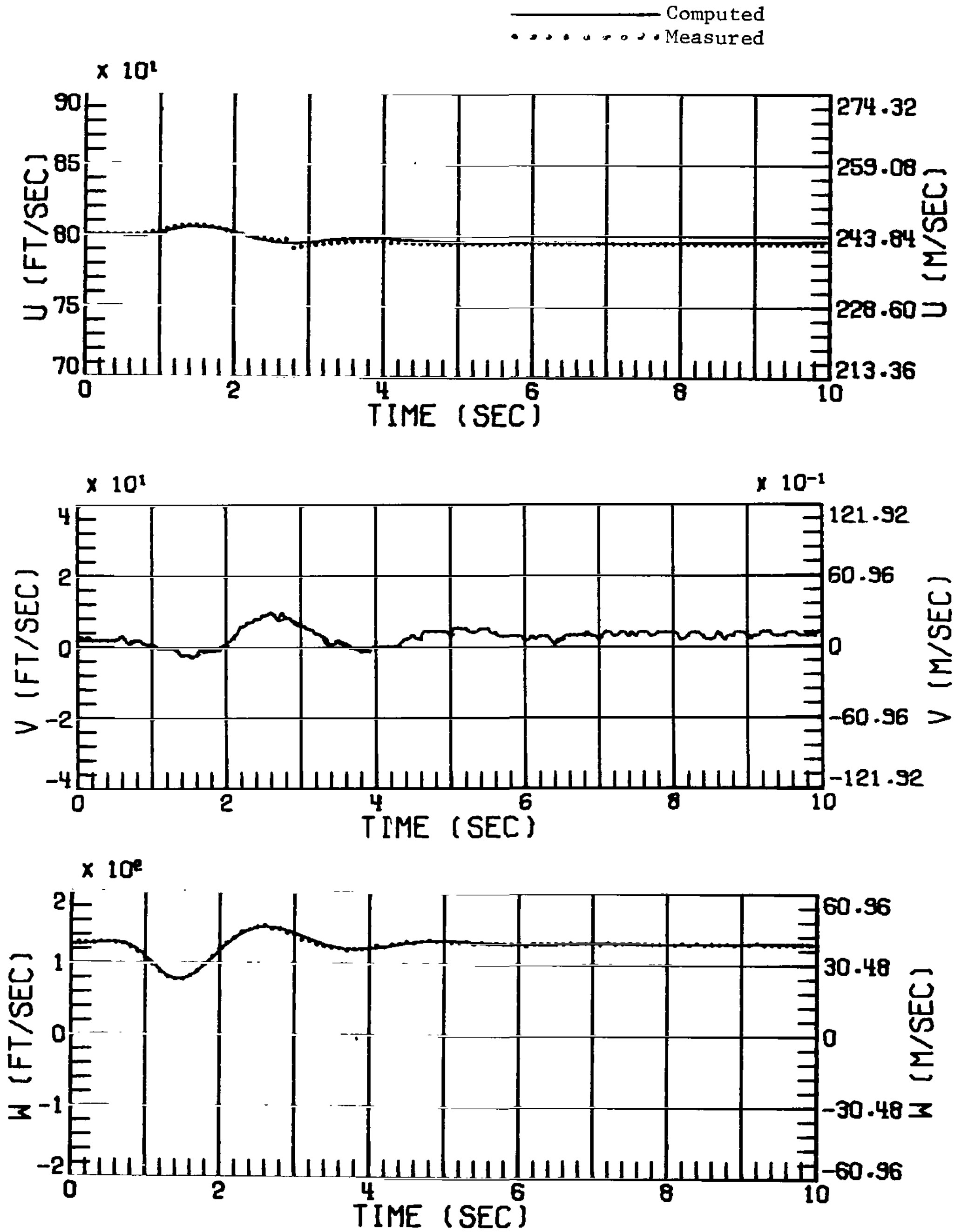

Figure 4.- Comparison of measured data with time histories computed by using parameters given in table V for flight data taken at $M = 0.81$ and $\alpha_t = 8.2^{o}$.

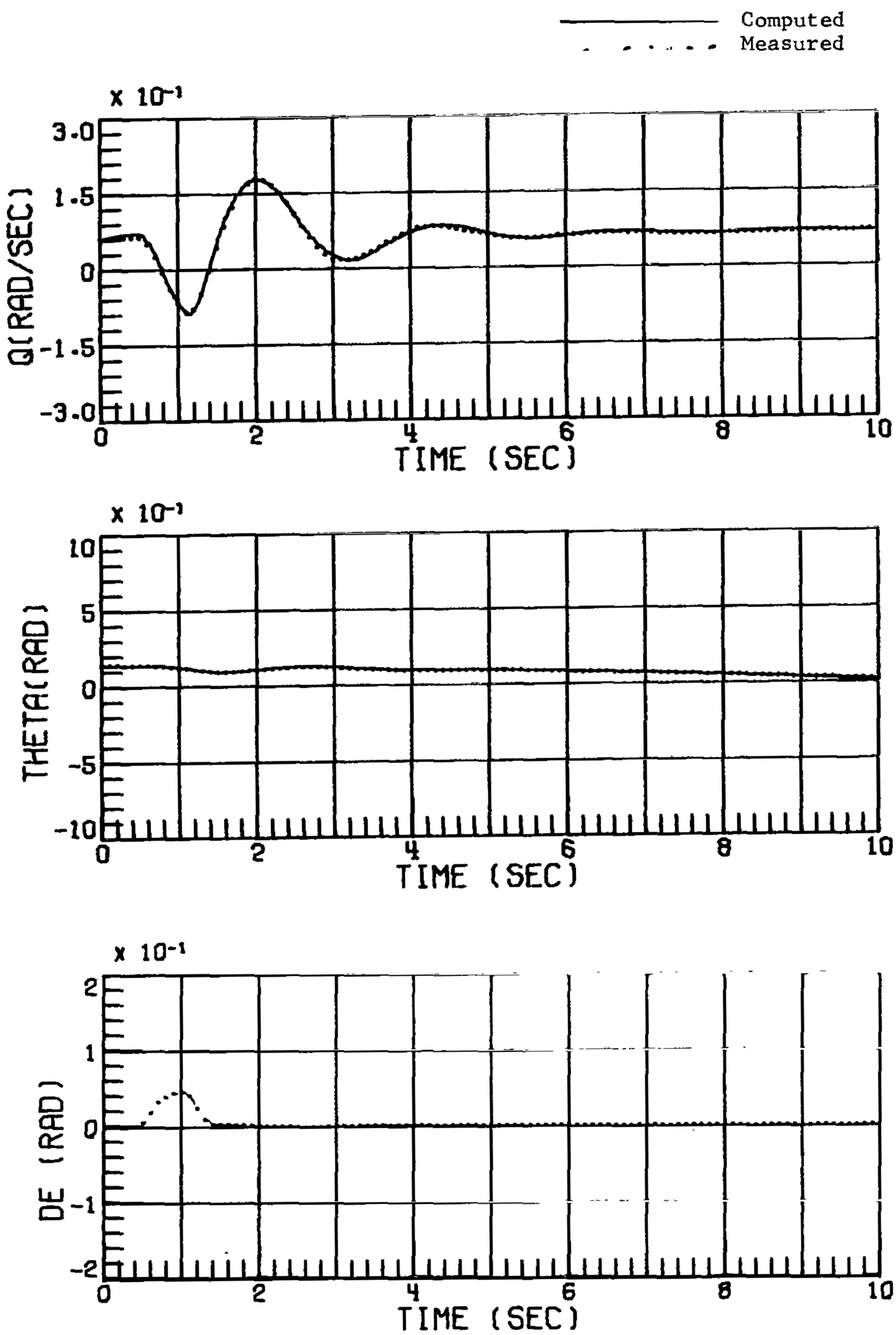

Figure 4.- Continued.

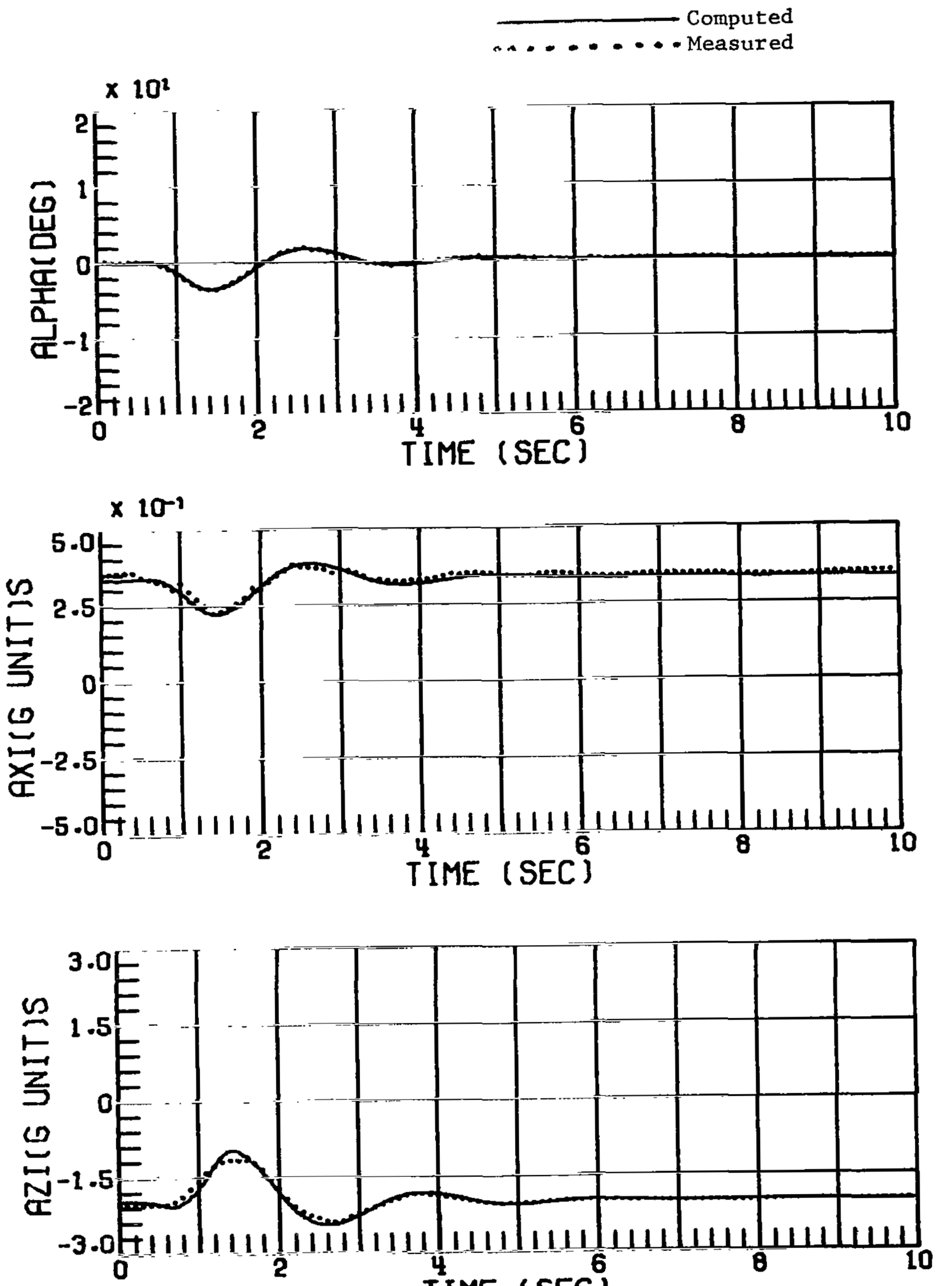

Figure 4.- Concluded.

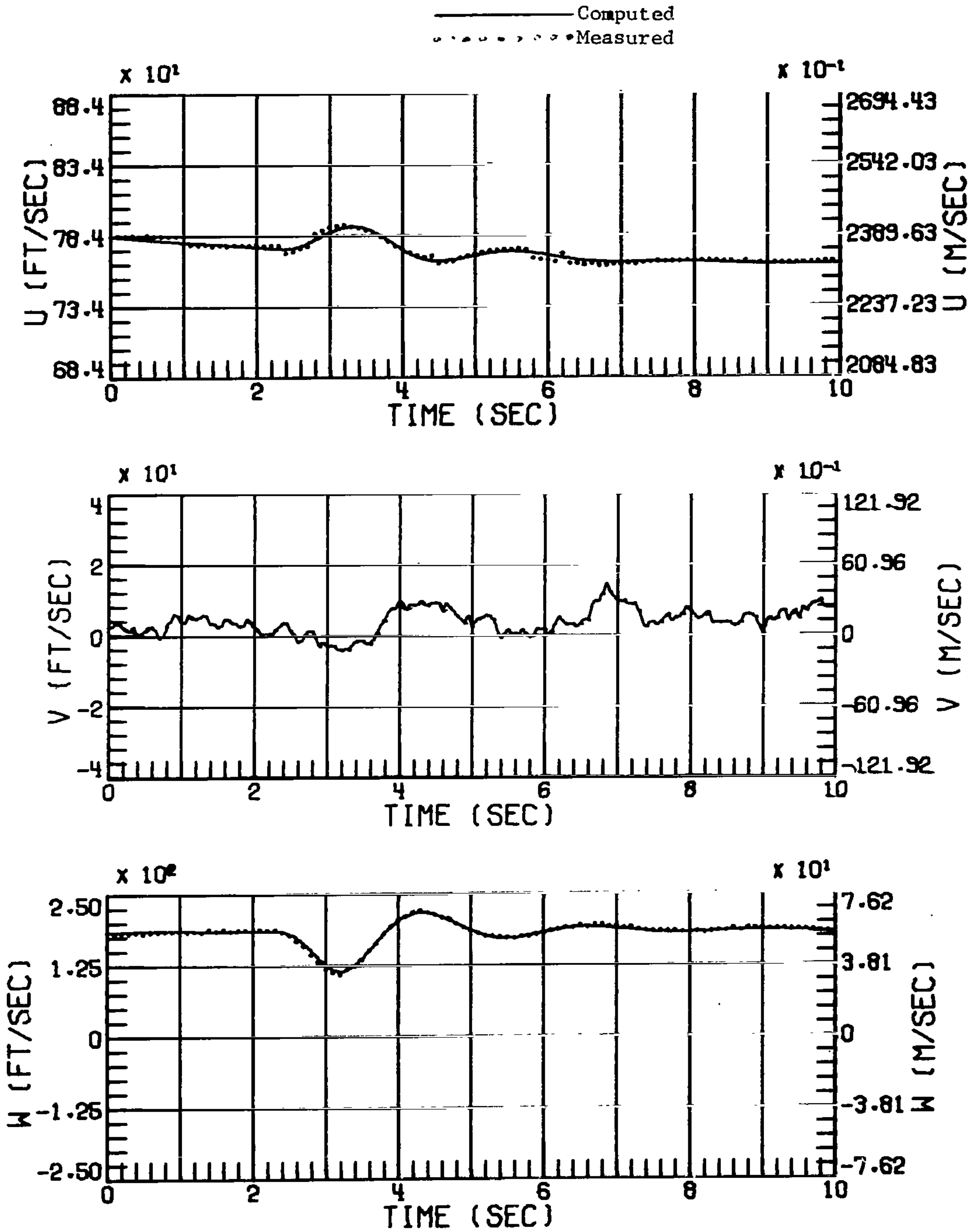

Figure 5.- Comparison of measured data with time histories computed by using parameters given in table V for flight data taken at $M = 0.78$ and $\alpha_t = 12^{\circ}$.

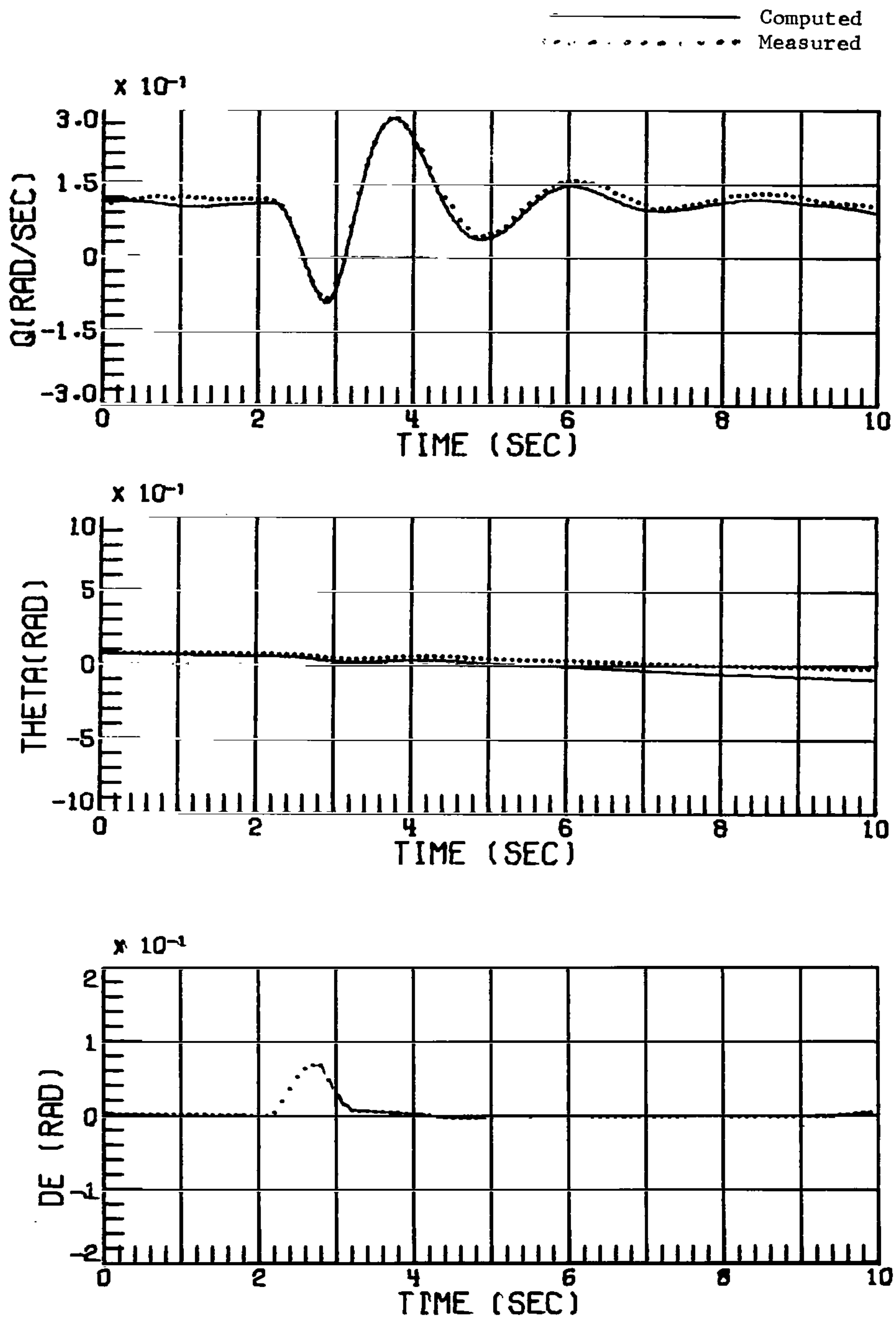

Figure 5.- Continued.

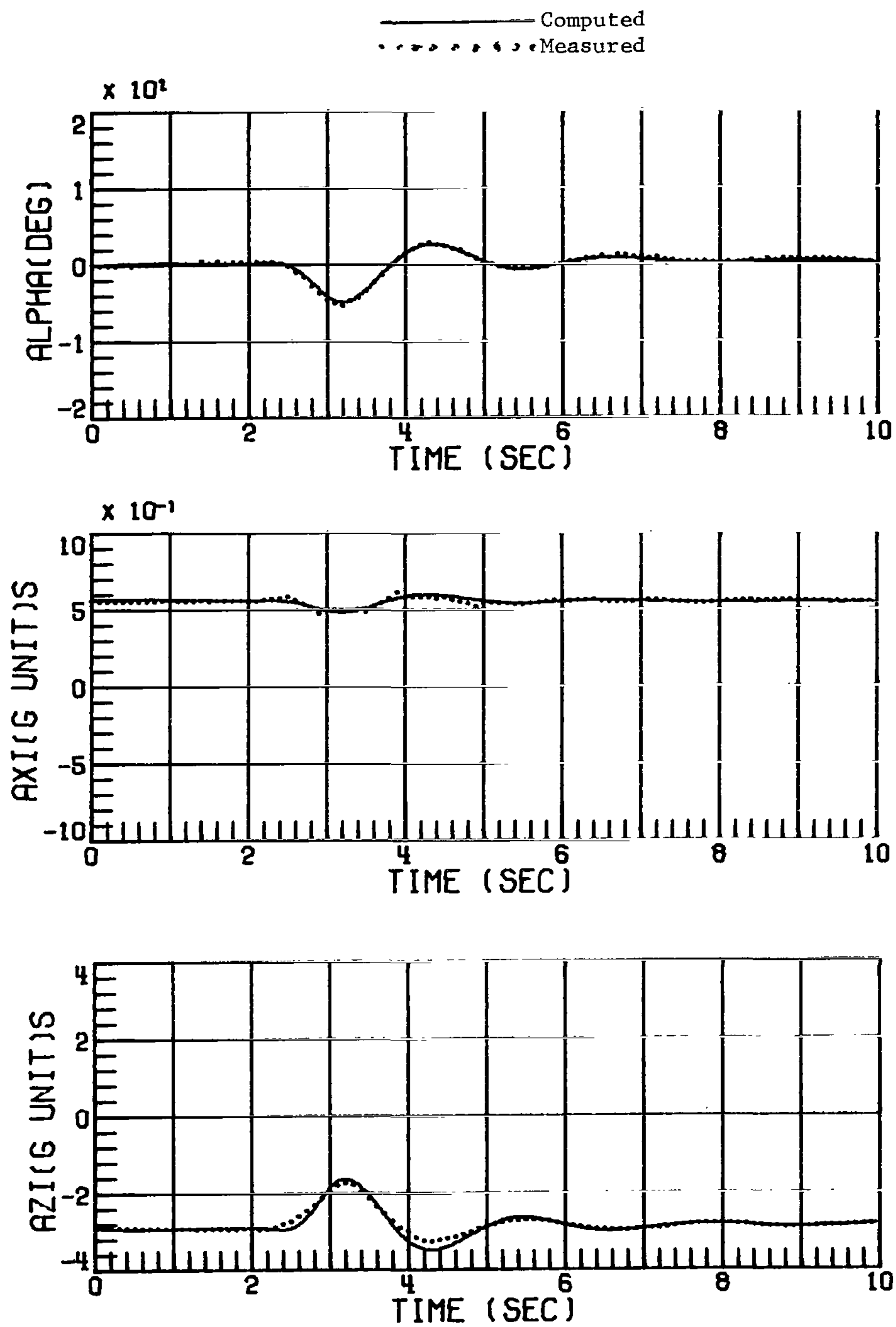

Figure 5.- Concluded.

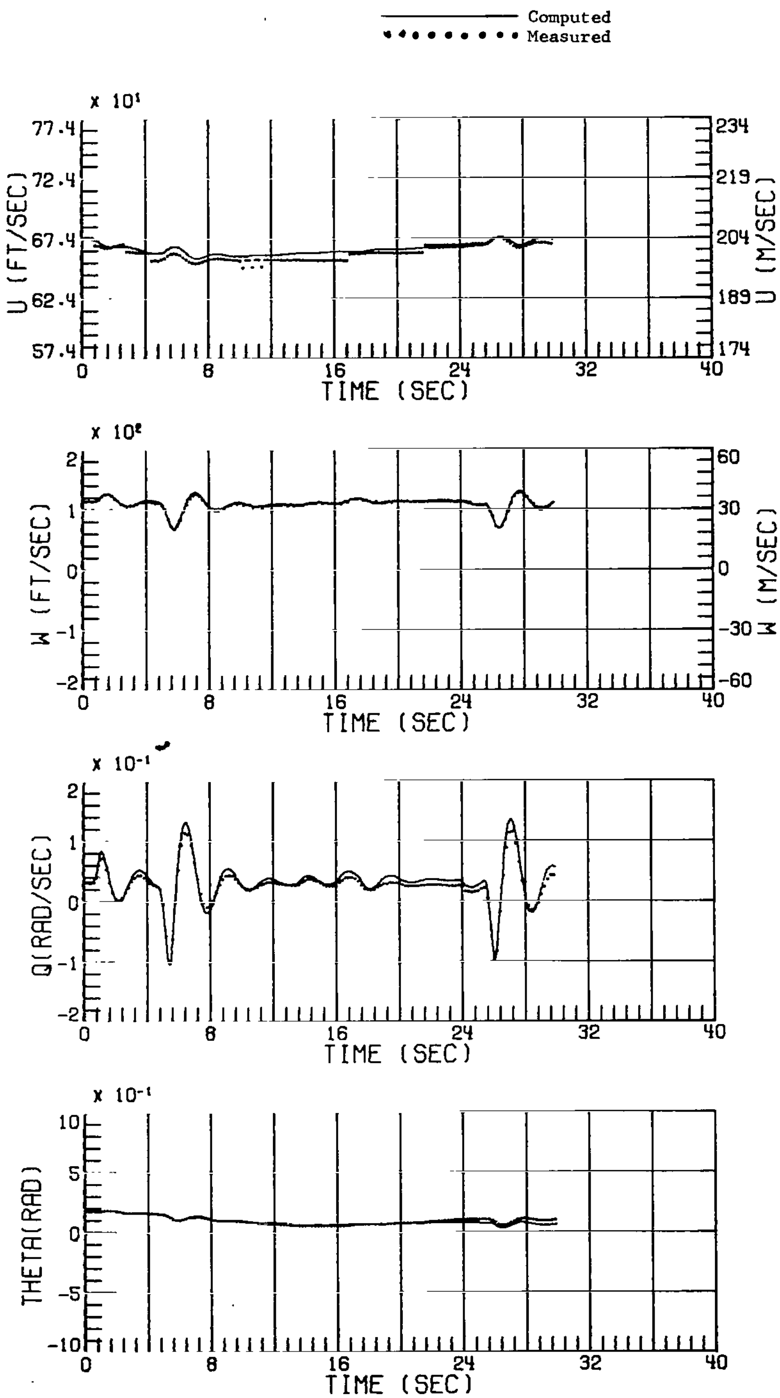

Figure 6.- Comparison of measured data of figure 2 with time histories computed by using parameters of reference 5 as given in table V.

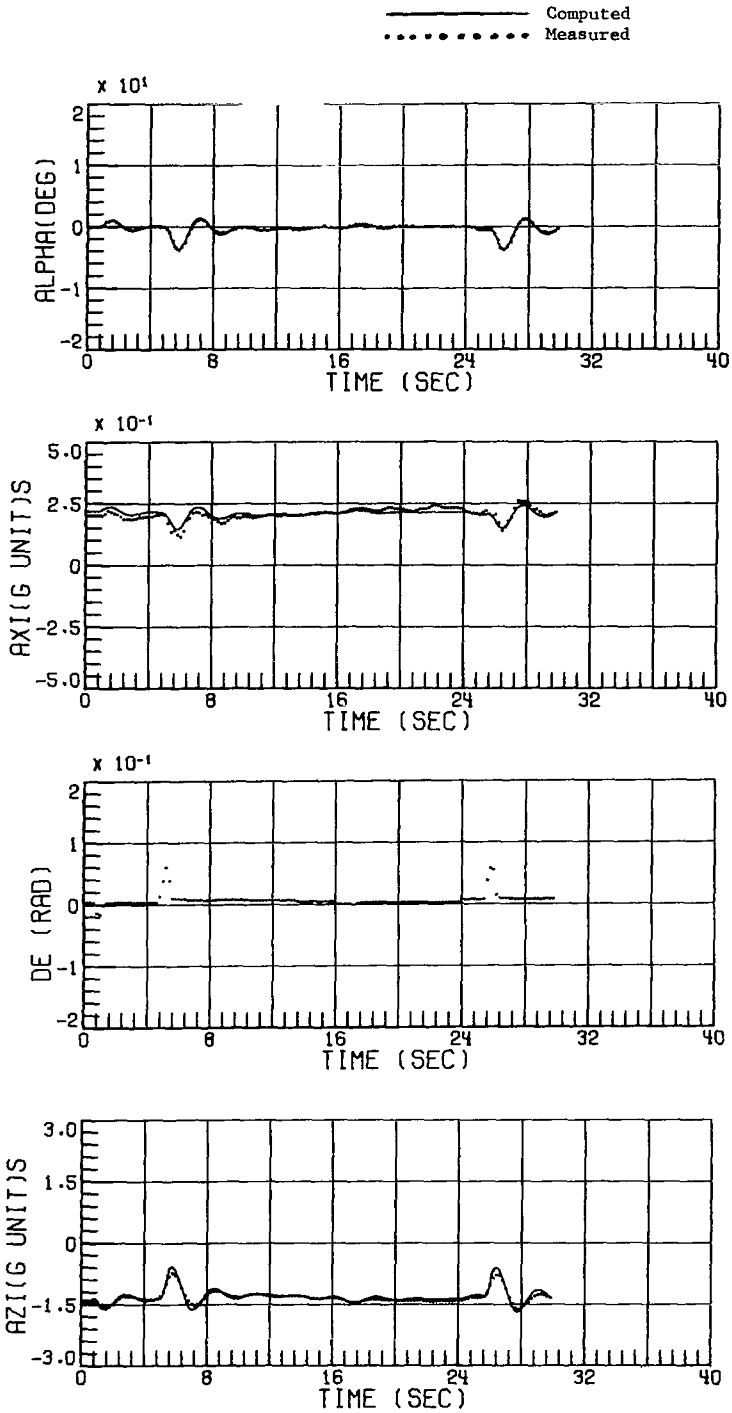

Figure 6.- Concluded.

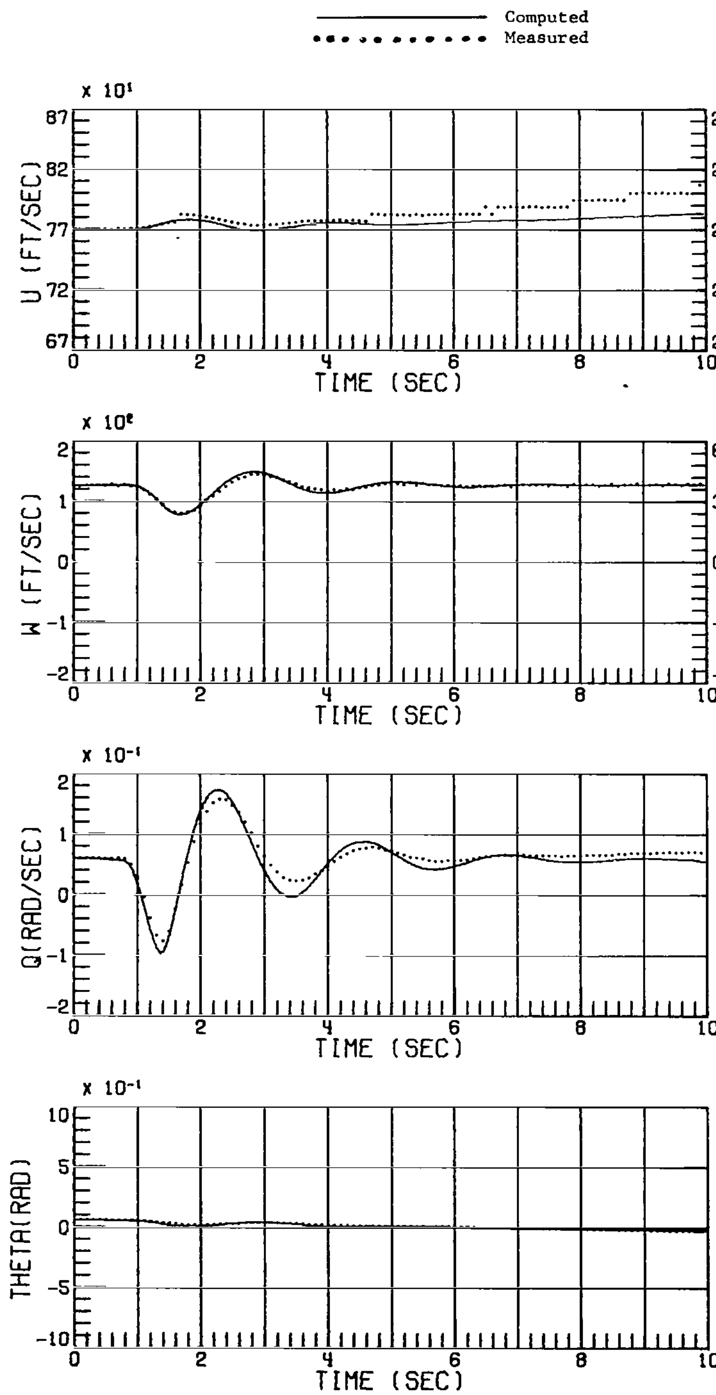

Figure 7.- Comparison of measured data of figure 4 with time histories computed by using parameters of reference 5 as given in table V.

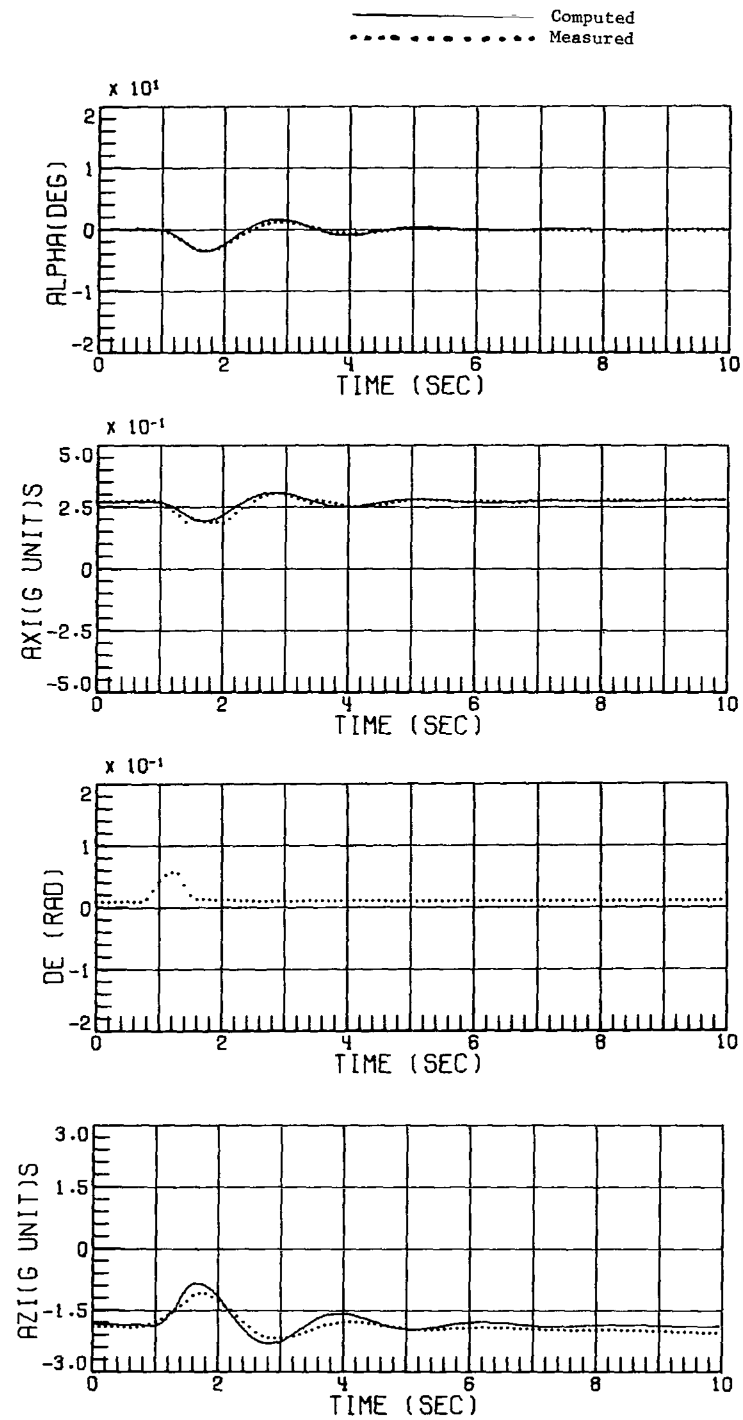

Figure 7.- Concluded.

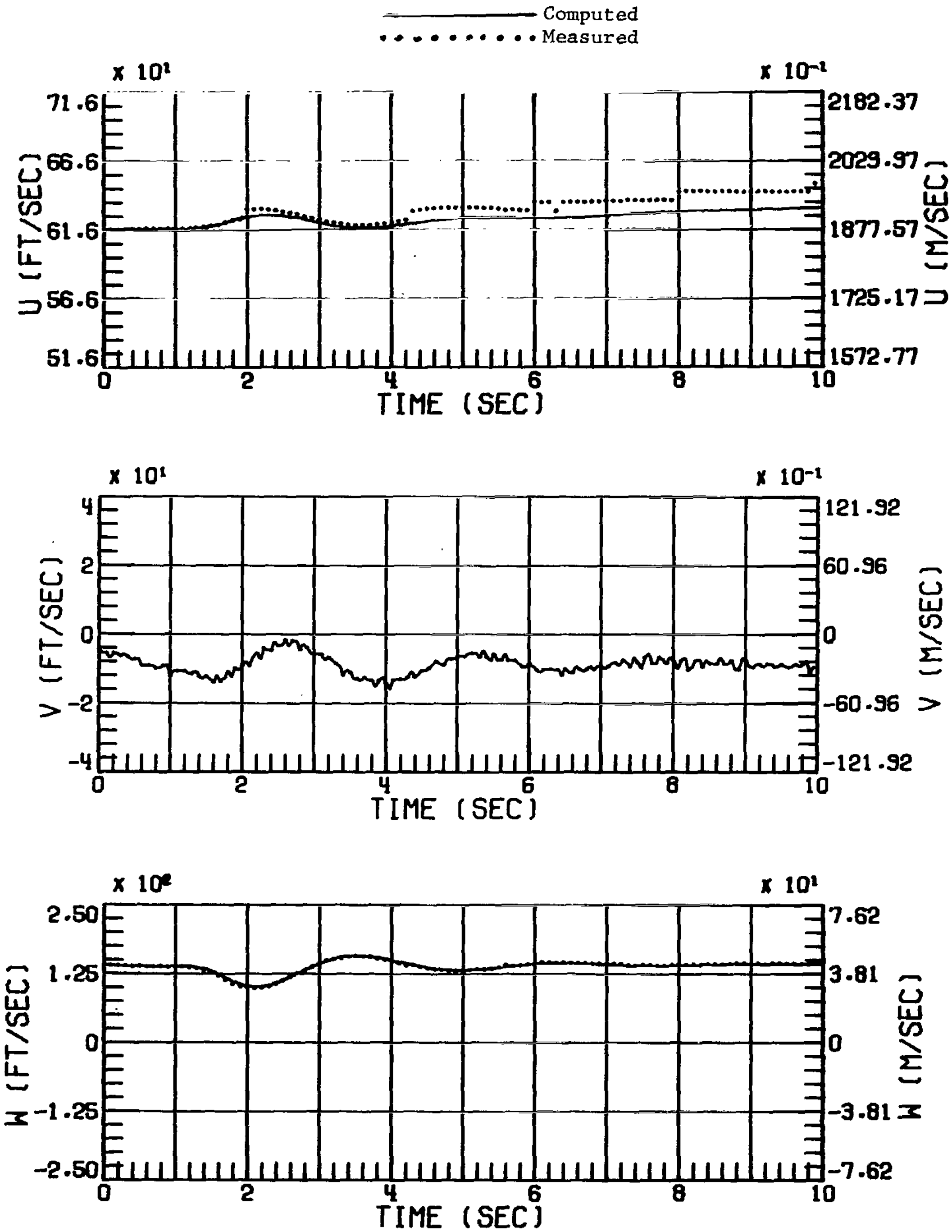

Figure 8.- Comparison of measured data not used during parameter extraction with time histories computed by using parameters of table V determined for flight conditions M = 0.66 and $\alpha_t = 13^{\circ}$.

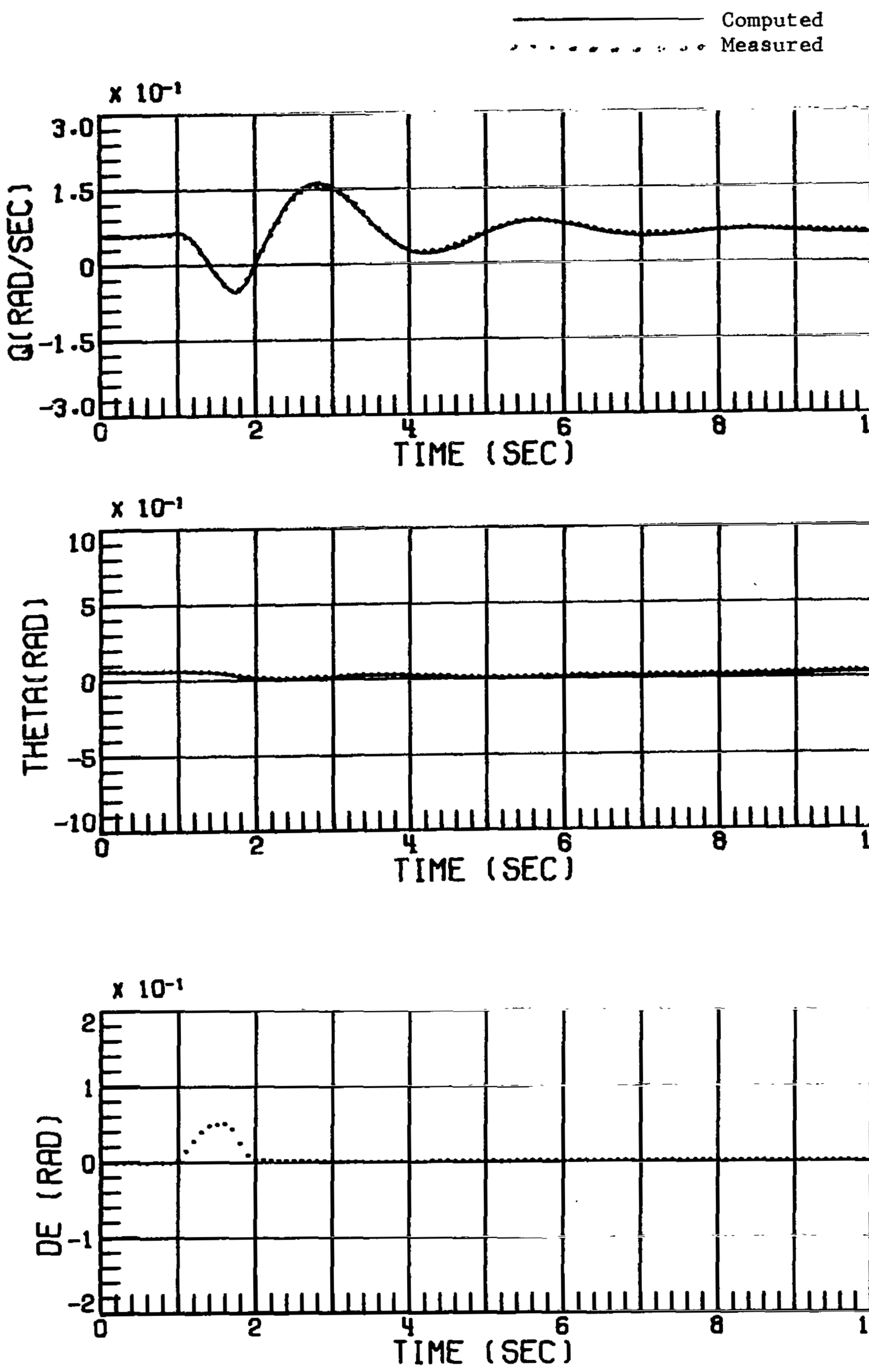

Figure 8.- Continued.

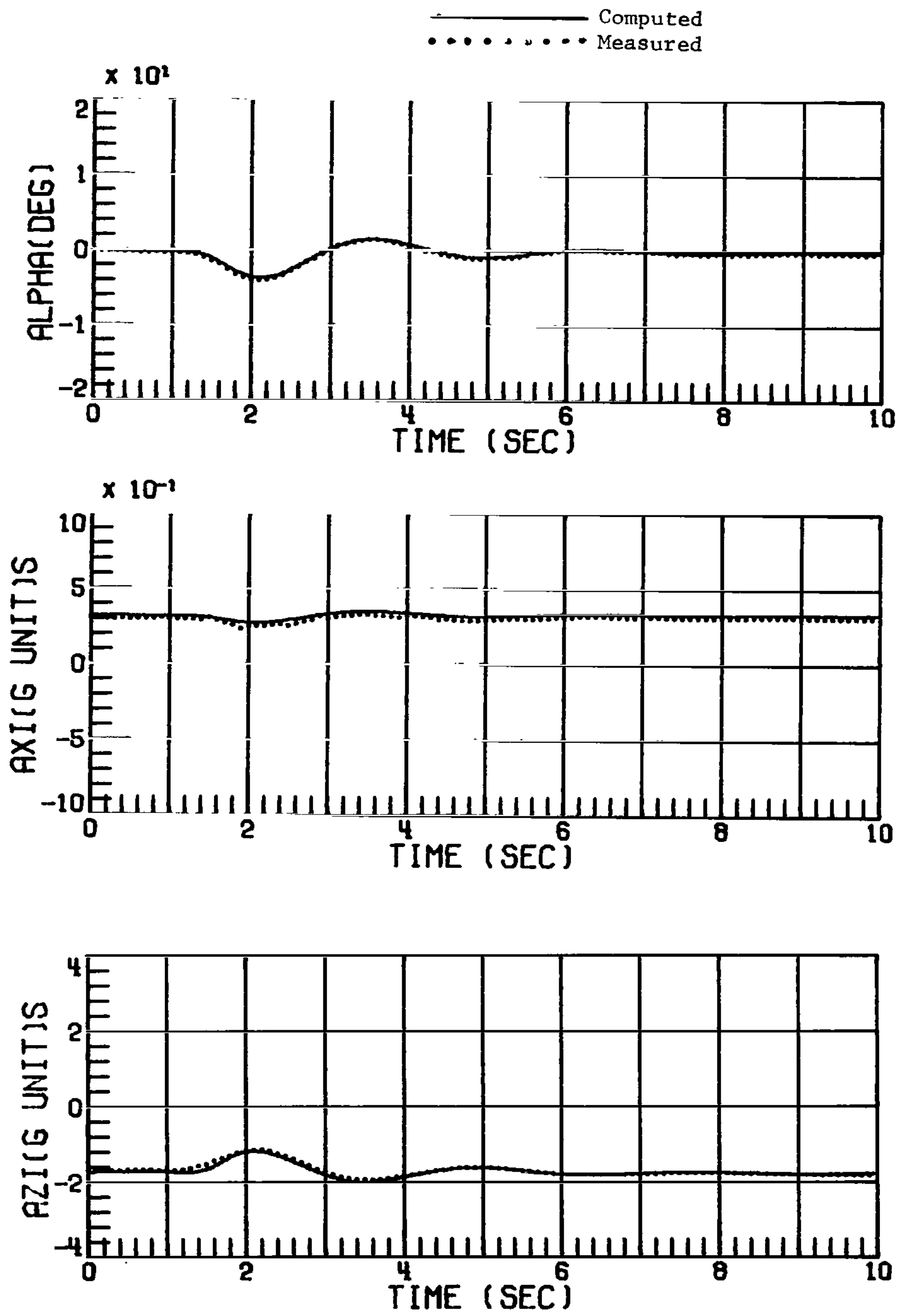

Figure 8.- Concluded.

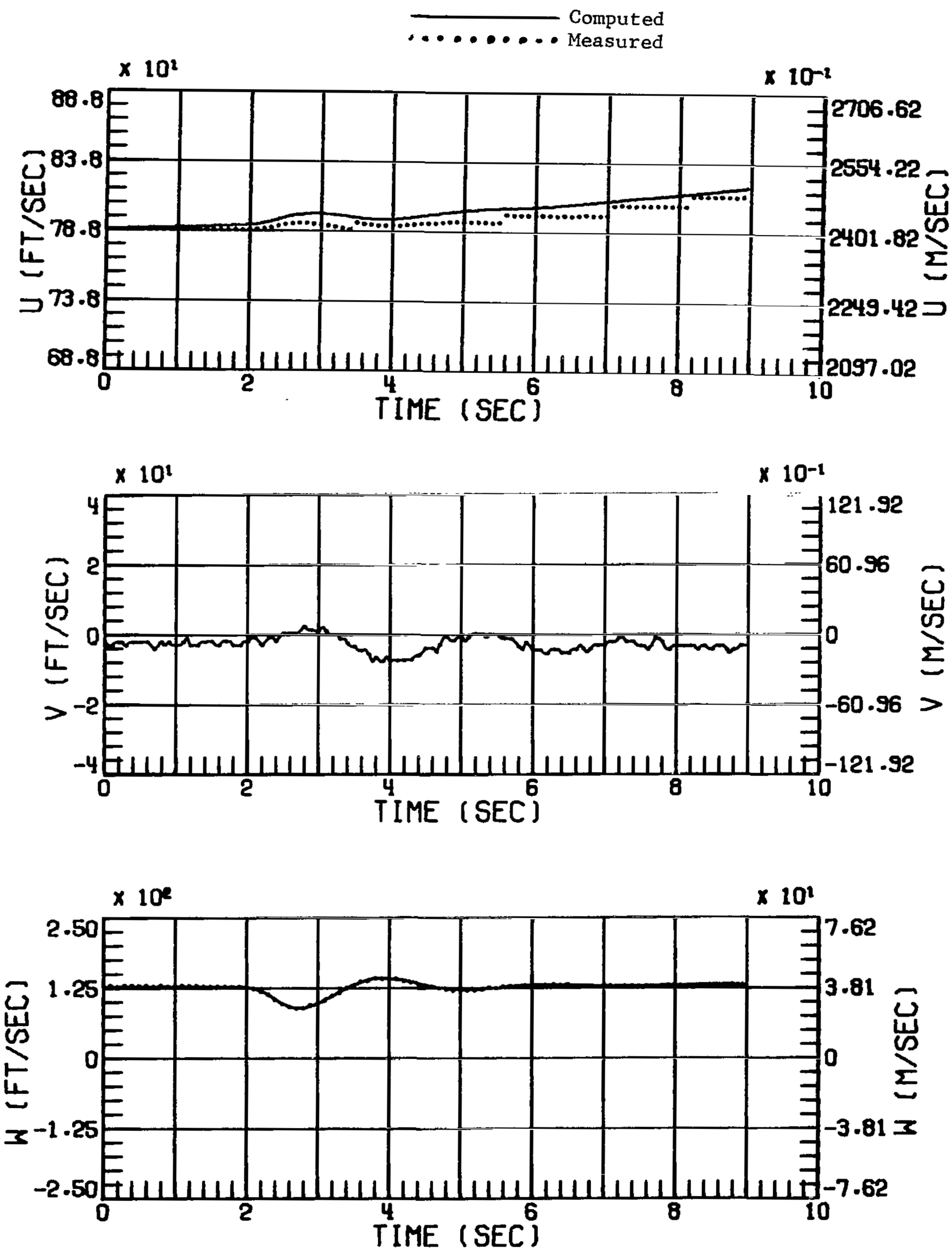

Figure 9.- Comparison of measured data not used during parameter extraction with time histories computed by using parameters of table V determined for flight conditions M = 0.81 and $\alpha_t = 8.2^{\circ}$.

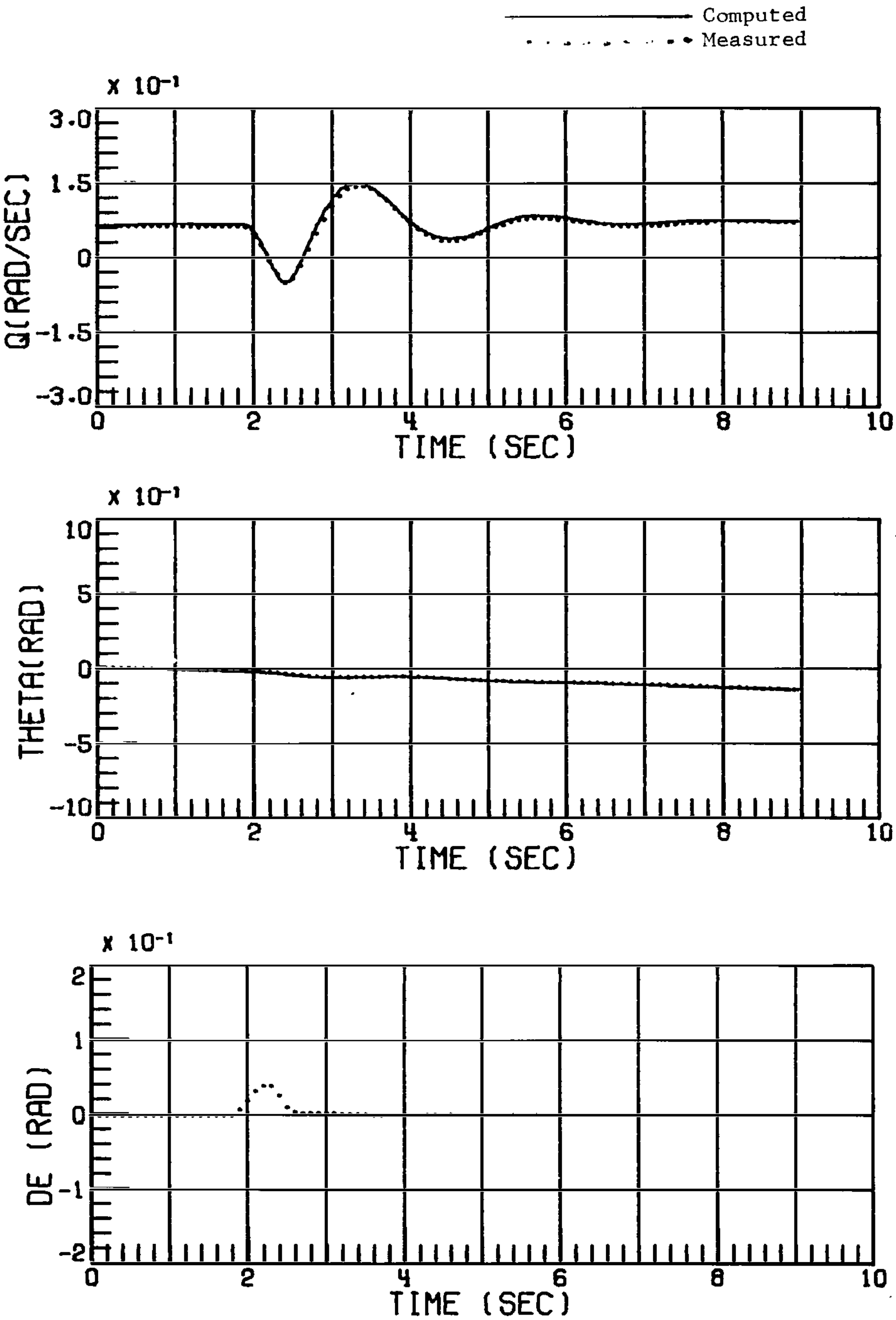

Figure 9.- Continued.

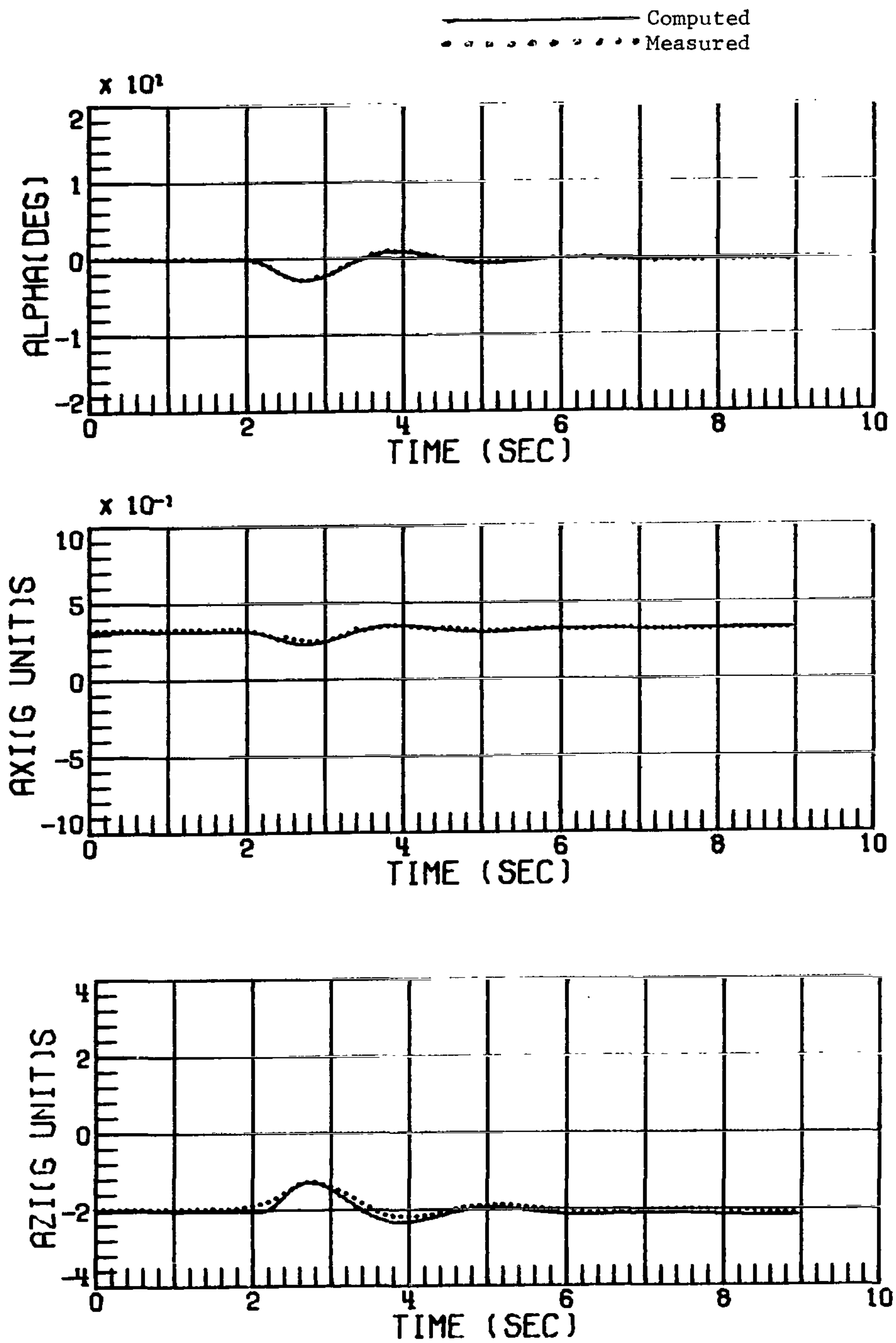

Figure 9.- Concluded.

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