National
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Effects of Flap Position on Longitudinal Parameters of HFB-320

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This Technical Memorandum is documentation of the work carried out by the authors as a part of the Technical Collaboration in the field of Parameter Estimation between the National Aeronautical Laboratory, Bangalore, India and the Institute for Flight Mechanics, DFVLR, Federal Republic of Germany under the CSIR/DFVLR special arrangements.

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Introduction

investigate the effects of small changes in flap position on the longitudinal derivatives of an aircraft, a project was initiated at the Institute for Flight Mechanics, DFVLR, Braunschweig, Federal Republic of Germany. The Systems Engineering Division of NAL, Bangalore,India undertook analysis of the flight data supplied by the Institute for Flight Mechanics, under the umbrella of technical collaboration arrangement between these two institutions.

Flight test data is from an instrumented research aircraft HFB-320 with complete longitudinal data set covering flap settings from approximately zero to 12 degrees in six steps. Data set provides adequate information to study the effects of flap settings on the longitudinal stability derivatives. This report is a preliminary documentation of the parameter estimation results obtained using the maximum likelihood method.

2.- Details of the Flight Test

A specific flight test programme has been carried out on the HFB-320, which is a twin jet high tail aircraft, originally manufactured by MIs MBB but modified and completely instrumented for In-Flight simulation by the Institute for Flight Mechanics, DFVLR, Braunschweig,-FRG [1]. A photograph of the test aircraft is shown in Fig. 1.

For the specific purpose of parameter estimation, the flight tests are carried out in a clean configuration with initial mass of 8170 Kg. Six different settings of flap position namely 0.54 2.64, 4.75, 7.22, 9.03, and 11.34 degrees are used to carry out the experiments. All the tests are initiated with'aircraft trimmed at an altitude of-about 5000m and at an indicated airspeed of 105 mlsecond approximately. Trim between the two flight experiments is carried out using only the elevator and the throttle controls. The spoilers and the slats are in retracted position. The specific flight procedure followed enables to attribute any effects that may be observed on aerodynamic derivatives to those due to the changes in flap positions.

At each test condition, the aircraft motion is excited only in the longitudinal plane through an elevator control input. Excitation input signal derived from the on-board flight computer consists of a multistep 3-2-1-1 signal followed by 'a'larger duration, pulse'[2].. The multistep input which has fairly wideband frequency range, excites the short period mode of the aircraft. The additional pulse excites the 'low frequency phugoid mode of the aircraft. This combination of.input signal provides adequate excitation of the longitudinal mode to enable accurate determinationof the longitudinal parameters [1].

Two flight tests at each of the six flap positions are carried out determine the run to rung variability. Thus a total of twelve flight tests are recorded for analysis. Each record consists of 60 to 90 seconds duration. The sampling rate of the data acquisition system is 10 sampless per second.

A list of variables recorded relevant to the current investigations is given in Table 1. Majority*of the sensors are located near` *the* centre. of gravity of the test aircraft. The true airspeed, angle of attack and angle of sideslip are measured at a nose boom. The geometrical data necessary for parameter estimation is provided in Table 2,. A summary of the twelve flight runs with *fuel* consumed' noted prior to certain runs is presented *in* Table 3. Fig. shows the plot of fuel consumed as 'a function of flight duration The total' fuel consumed prior to *each* flight test is obtained by,interpolation. To improve the accuracy of parameter estimates, the variations in the mass can be appropriately accounted for in the nonlinear estimation procedure, by *using* the details of the fuel consumed

3. p Compatibility Chec

From the available records of the various variables the data consistency has been checked by bootstrapping the information by relevent kinematic equations The complete set of state,, equations and the observation equations *used* for this. **purpose are given o**n the next page. The estimated state variabl s are ,*e.* All-the control variables a. a_y, a_z p are assumed to be biased The observation variabl s are V 4>,@ and q.

State equations:

u a_x - Aa.) (r -. Ar)v - (- Aq)w - g Sin6 , u(0)=u0

(a$_y$ - Aa$_Y$) + (p - Ap)w - (r - Ar)u + g COBB Sin4

w = $(a_z - \Delta a_z) + (q - \Delta q)u - (p - \Delta p)v + g \cos\theta \cos\phi$

= (p - A$_p$) + ($_q$ - Aq) Sink Tan6 + (r - Ar) Coscp Tang (1)

$- \Delta r) \sin\phi$; $\theta(0)=\theta_0$

Observation equations

Vm = V$_n$ + AV

a$_m$ = Tan^{-1} (w,/un)

a Sin^{-1} (vn/Vn) (2)

$\phi_m = \phi$

$\theta_m = K_\theta \theta$

$\bar{q} = K_{\bar{q}} \frac{1}{2} \rho V^2 + \Delta\bar{q}$

with additional equations for true airspeed and measurements at the nose boom:

(3)

$$V = (u^2 + v^2 + w^2)^{\frac{1}{2}}$$

$$V_n = (u_n^2 + v_n^2 + w_n^2)^{\frac{1}{2}}$$

where $u_n = u -$ (r -'Ar)-Y$_n$ _ Aq) Zn (4)

$v_n = v -$ (p AP) zn - Ar) x$_n$

$w_n = w -$ (- Aq) x$_n$ + (p - 4P) Yn

Results of this analysis for six runs obtained using the maximum likelihood parameter estimation method (31 are shown in Fig 3 (i) to (vi). The estimated numerical values of the various measurement errors, scale factors and the initial conditions are summarized in Table 4. Fig. 3 indicates a generally good match between the measured and reconstructed responses for variables a.and 6. Some variability is observed in variables (3 and 0. This is attributed to the reason that the input variables such as $_3$ p and r were of low magnitudes, since the flight experiments were restricted to excitation in the longitudinal plane alone. Furthermore, the variables $_i$ and 4 being lateral quantities, they are not of significant importance in the present analysis.

Based on Table 4 and Fig. 3 and on the results of data consistency for other six runs not presented here, the following general conclusions can be drawn:

i) Measurement of true airspeed is consistently high by an average of 9 to 10 m/sec.

ii). Measurement errors in a_x and a_z are negligible.

iii) Scale factor error in a is small.

It has been observed from Table 4 that the estimates of scale factor K_q and bias pq in the dynamic pressure measurement show larger deviations. For this reason it is recommended to use the reconstructed time histories for derivative estimation.

In addition, for some of the runs a time lag has been observed between the measured and the estimated airspeed V. A first order lag model can be used to approximate the time delay. Possible sources of the time lag in airspeed have, been discussed in Ref. 14], However, for this data set of twelve runs it was pot possible to estimate the time constant consistently. Also wheneverit was estimated, the associated standard deviation was high. Since;, these results did not show consistent trends, the slight time delay observed in some of the records of airspeed is not considered for further investigations using this flight data. However, based- on the above pointed out first observation, correction of 9 m/s (A V) has been carried out to the true airspeed, before using the data for aerodynamic. derivative estimation.

ti **actin 21** Dimensional periyatives. Using LineaC Model

The problem considered in the present investigations is to determine the effects of small, changes in the flap deflections on the aerodynamic coefficients which define the lift, drag. and the pitching moment f aircraft. This specific problem definition requires modeling of the aircraft' motion in the **longitudinal** plane.

The flight tests, discussed in Section 2, are carried out under reasonably steady atmospheric conditions and at an angle of attack cC 6.5 degrees approximately. At such flight conditions, linear aerodynamic models can be postulated fo force and moment coefficients. The mode coupling effects ar usually small and are assumed to be neglected. The following linear model in terms of the dimensional derivatives definin the normalized forces and moment is considered (4-71;

State equations:

$$\dot{u} = X_0 + X_u\, u + X_w\, w - qw - g\, \mathrm{Sin}\theta + \frac{F_e}{m}\, \mathrm{Cos}\sigma_T + b_{x_{\dot{u}}}$$

$$\dot{w} = Z_0 + Z_u\, u + Z_w\, w + qu + g\, \mathrm{Cos}\theta - \frac{F_e}{m}\, \mathrm{Sin}\sigma_T + b_{x_{\dot{w}}} \qquad (5)$$

$$\dot{\theta} = q + b_{x_{\dot{\theta}}}$$

$$\dot{q} = M_0 + M_u\, u + M_w\, w + M_q\, q + M_{\delta_e}\delta_e + \frac{F_e}{I_y}\left(\ell_{tx}\, \mathrm{Sin}\sigma_T + \ell_{tz}\, \mathrm{Cos}\sigma_T\right) + b_{x_{\dot{q}}}$$

Observation equations:

$$u_m = u + b_{y_u}$$

$$w_m = w + b_{y_w}$$

$$\theta_m = \theta + b_{y_\theta}$$

$$q_m = q + b_{y_q} \qquad (6)$$

$$\dot{q}_m = M_0 + M_u\, u + M_w\, w + M_q\, q + M_{\delta_e}\delta_e + \frac{F_e}{I_y}\left(\ell_{tx}\mathrm{Sin}\sigma_T + \ell_{tz}\, \mathrm{Cos}\sigma_T\right)$$

$$a_{x_m} = X_0 + X_u\, u + X_w\, w + \frac{F_e}{m}\, \mathrm{Cos}\sigma_T$$

$$a_{z_m} = Z_0 + Z_u\, u + Z_w\, w - \frac{F_e}{m}\, \mathrm{Sin}\sigma_T$$

where u, w, 8, q are the four state variables, $_{de}$ is the control input. The above equations include the effects due to variations in the thrust F_e. Jet *engines* are located behind and above the centre of gravity. The offset distances are given by Q_{tx} -2.67m and £,$_z$= -0.5 m, Furthermore, the thrust axis is inclined **upward** at **an** angle of or, = 3 degrees. For the purpose of estimation, thrust **is** considered as an input variable. Subscript m refers to the measurement variables obtained from the flight **test.**

It may be noted from Table l' that the thrust F_e is not a directly measured variable. However, prior to estimation it i computed using, the thrust calibration curves and the actual measurements *of* engine pressure'ratio, velocity and the static pressure.

Variables u, w are the velocity components along the X- and Z-directions. These are not measured directly during the test and hence need to *be* derived prior to estimation, using the measurements of V, oc All the three quantities are measured at the nose boom with offset distances x„ = 10.992m, y" O.Om and zh= 0.556m (Table 2). The corresponding quantities referred to the centre of gravity are computed prior to estimation.

In addition to the unknown aerodynamic derivatives, Eqs. **5 & 6 contain** bias terms b., and b_y. These terms are necessary to appropriately account for the unknown initial conditions and measurement zero shifts 13 41. This yields the complete parameter vector to be estimated as:

$$= \{ X_0 \quad X_u \quad X_w \quad Z_0 \quad Z_u \quad Z_w \quad M_0 \quad M_u \quad M_w \quad M_q \quad M_{\delta_e} \quad b_{x_{\dot{u}}} \quad b_{x_{\dot{w}}} \quad b_{x_{\theta}} \quad b_{x_{q}} \quad b_{y_u} \quad b_{y_w} \quad b_{y_\theta} \quad b_{y_q} \}^T \tag{7}$$

Using the above linear model, all the twelve sets of flight records are analysed. Typically 60 to 80 seconds long records are used for the estimation. Maximum -.likelihood estimation method has been used here I3). The estimates of dimensional derivatives along with respective standard deviations in percentages are summarized in Table.5. The time history plots, showing the comparison between the measured and estimated (model) responses, for all the twelve runs are provided in Figs. 4 (i) through (xii).

The estimated dimensional derivatives are converted into the non-dimensional form and are presented inTable 6 and Fig. 6. The above conversion is required t compare the results with those obtained using a nonlinear, model formulated in terms of non-dimensional derivatives to be discussed in the next section. A constant nominal value of dynamic pressure, equal to that at the begining of the test, is assumed in such conversions. In the present case a value of $\bar{q}$ = 4200 pa is used.for all the runs. Appropriate mass for each run, after accounting for the fuel consumed (Fig. 2), is used. The other values of :aircraft data like I_Y, S, and $\bar{c}$ are defined in Table 2. Furtherdiscussion of these results is presented in Section 6.

5. Estimation 2 N,gn-Dimensional Derivatives Using Nonlinear Model

The, hitherto most widely: used approach of estimating the dimensional stability and control **derivatives** from linearized system equations has been presented in the previous section. The dimensional derivatives depend the dynamic pressure q. **For convenience, the dynamic pressure has been assumed to be constant over** th run However en greater airspeed changes take place during the maneuver can result in significant variations in the dynamic pressure q which i proportionalto square of the velocity.. These variations in are found to affect significantly the estimates of some of, **the derivatives**,, particularly those which are, functions of. **velocity [4,8].** In such cases it becomes necessary to consi-der' the dynamic pressure as an additional variable i the estimation procedure This necessitates-reformulation of the system equations derived in Section 4 in`terms of non-dimen-sional quantities slowing nonlinear model i consi-dered here [4];

State equations:

$$\dot{u} = \frac{\bar{q}S}{m}\left(C_{x_0} + C_{x_u}\frac{u}{V_0} + C_{x_w}\frac{w}{V_0}\right) - qw - g\,\mathrm{Sin}\theta + \frac{F_e}{m}\mathrm{Cos}\sigma_T \qquad ; u(0)=u_0$$

$$\dot{w} = \frac{\bar{q}S}{m}\left(C_{z_0} + C_{z_u}\frac{u}{V_0} + C_{z_w}\frac{w}{V_0}\right) + qu + g\,\mathrm{Cos}\theta - \frac{F_e}{m}\mathrm{Sin}\sigma_T \qquad ; w(0)=w_0$$

$$\dot{\theta} = q \qquad ; \theta(0)=\theta_0 \tag{8}$$

$$\dot{q} = \frac{\bar{q}S\bar{c}}{I_y}\left(C_{m_0} + C_{m_u}\frac{u}{V_0} + C_{m_w}\frac{w}{V_0} + C_{m_q}\frac{\bar{c}}{2V_0}q + C_{m_{\delta_e}}\delta_e\right) + \frac{F_e}{I_y}\left(\ell_{tx}\,\mathrm{Sin}\sigma_T + \ell_{tz}\,\mathrm{Cos}\sigma_T\right) \qquad ; q(0)=q_0$$

Observation equations:

$$V_m = (u_n^2 + w_n^2)^{\frac{1}{2}}$$

$$\alpha_m = \tan^{-1}\left(\frac{w_n}{u_n}\right)$$

$$\theta_m = \theta$$

$$q_m = q \qquad (9)$$

$$\dot{q}_m = \frac{\bar{q}Sc}{I_Y} \; C_{m_0} \quad u \quad \frac{u}{0} \quad 0 \quad C_{M_q} \sim q \quad C_{m_{\delta e}} \; d$$

$$+ \frac{F_e}{I_Y}\left(\ell \ldots \sin\sigma_T + l_z \cos\sigma_T\right),$$

$$a_{x_m} = \frac{\bar{q}S}{m}\left(C_{x_0} + C_{x_u}\frac{u}{V_0} + C_{x_w}\frac{w}{V_0}\right) + \frac{F_e}{m}\cos\sigma_T$$

$$a_{z_m} = \bar{q} \quad C_{z_u} \quad a \quad + C_{z_w}\frac{w}{V_0}\Big) - \frac{F_e}{m}\sin\sigma_T$$

with additional equations for measurements at nose boom

$$u_n = u + q\,z_n$$

$$w_n = w - q\,x_n$$

and for reference air speed $V_0 = (u_0^2 + w_0^2)^{\frac{1}{2}}$

The above equations include theeffects due thrust variations. The various offset distances and other quantities have already been defined (refer Table 2). Appropriate mass t the beginning of each run, after accounting for the fuel consumption, is used in the'estimation procedure.

It is important to note that the formulation of system equations using non-dimensional derivatives and using dynamic pressure q as a variable, necessarily leads to a nonlinear system representation The aerodynamic parameters be estimated are:

$$\Theta = \{ C_{x_0} \; C_{x_u} \; C_{x_w} \; C_{z_0} \; C_{z_u} \; C_{z_w} \; C_{m_0} \; C_{m_u} \; C_{m_w} \; C_{m_q} \; C_{m_{\delta_e}} \}^T \quad (11)$$

In addition to the'above unknown coefficients, it is necessary to estimate the unknown initial states u_0, θ_0, q_0. Although the measurementt zero shifts are not explicitly included in the *Eqs, (5.8 & 5.9)* for the sake of clarity, it is essential to. consider them appropriately for nonlinear systems as discussed in Ref 13,,81.

Using the abo e nonlinear model, Eqs. (8-& 9), all the twelve flight test recordsare analysed. The numerical values of the directly estimated non-dimensional longitudinal derivatives are provided in Table 7. Response matching of time histories obtained from parameter estimation are shown in Figs. 5 (i) through 5 (xii).'The non-dimensional coefficients are also plotted as function of flap deflection in Fig. 6.

The lift and drag coefficients, C and C_D ,for all the twelve runs are obtained from the estimates of the longitudinal coefficients, $_V$ and C_Z (Table 6 & 7) using the following transformations.:

$$C_L = C_X \text{ S na} \quad \text{Cosa}$$
$$C_D = - C_X \text{ Cosa} \quad \text{Sina} \quad (12)$$

The aerodynamic characteristics, obtained. from the estimated coefficients, appearing in the Taylor series expansion are shown in a conventional manner as plots of C_L Vs a^c C vs C_D and CL vs C.., in Figs. 7 8 and 9 respectively. Alternatively, It is also possible o estimate the lift, drag and pitching moment coefficients directly, by, reformulating the nonlinear model in terms of variables n the wind axes (4].

Result..a and piscussions

n this section the estimation results based on the 1 near, and nonlinear modes are evaluated compared both qualitatively and quantitatively. Effects of flap position on the longitudinal aerodynamic coefficients are also highlighted.

In general, observed from Fig. 4 (1) through (xii) that the linear model with dimensional derivatives yield fairly acceptable fit f r a,majority of the. variables. Some discrepencies in Fig. 4 (ii, v and ix) for pitch rate q' and Fig.; A (,viii,ix,xi) for longitudinal acceleration a are observed, However, considering the complexity, of flight test procedure,, and the aggregated errors accruing in flight instrumentation, estimation of gross aircraft parameters like fuel consumption, mass inertia, such a linear model can be considered acceptable in n overall sense.

However, a one to-one comparison of Fig 4 (D through (xii) with Fig. 5 (1) through (xii) clearly indicates that the nonlinear model, in terms of non-dimensional derivatives, yields significantly improved overall agreement of all the variables. Particularly, the comparison of time histories for pitch rate in Fig. (ii, v and ix) with corresponding plots in Fig 5 (iii v and as well for longitudinal'accelera-tion in Fig. _4'(iv,v,viii,ix,xi) with those in Fig. 5 (iv,-v,viii,ix,xi) vividly demonstrates significant qualitative improvement provided by the nonlinear model.

Quantitatively also the above indicated improvements are corroborated by the fact that the determinant of the measure-ment error covariance matrix is lower the nonlinear model by factor of 10**3 to 10**5 than that obtained for the linear model. Although the absolute value of the determinant can not be directly interpreted, it=roughly provides informa-tion about the goodness of fit It is observed from Fig, 6 .that the'run to run scatter in the estimates are low for both linear and nonlinear models.

All the moment derivatives except C. show reasonably good agreement between the two types of modeling.. The diffe-rences in values of C between the two models i attributed to the fact that, the linear model the nonlinear inertial and gravitational terms are computed' using the uncorrected measured values and treated as additional pseudo control inputs This can introduce small bias errors. The nonlinear model automatically overcomes this difficulty, Values of $-C_{M_{q}}$

estimated by the nonlinear model are found to be more realistic.

The speed derivatives are substantially different in the two types of modeling. This is mainly due-to the reason that in the nonlinear model the non--dimensional coefficients are multiplied by a term proportional to the cube of V,, while in the linear case, the dimensional coefficients by term proportional to V I 4].

6.1 Flap Sensitivity Coefficients:

The effects of the flap deflection on the aircraft longitudinal aerodynamic derivatives can be expressed as flap sensitivity coefficients, which are defined as:

$$\frac{dc_{xu}}{a6f} - xuaf,$$

$$\frac{dc_{mo}}{d6} = c_{mo\delta} \tag{13}$$

etc. for all'the eleven longitudinal derivatives estimated in section' 4 and 5. Implicit in this definition is that the derivative has a dominant first order relation to flap deflection. To determine such a relation, a linear regression line fit of the form y=A+Bx was fitted to all the estimates f derivatives as a function of flap position. In order t determine whether a monotonic relation did exist or not, normalized percentage variations of the derivative per degree of flap deflection (i.e. 100*B/A) as well as for the experi--

mental range of 10 degrees have been estimated and provided in Table 8.

From Table 8 and Fig. 6 (i)-(vi) the following general: conclusions can be drawn:

The derivatives C_{MO} and $C_{x\mu}$ are strong functions of flap **setting,** showing 150 to 200'% variation for a 10 degrees flap change.
Thus, $C_{,0}$ and C_xu~ are flap sensitivity coefficients which adequately describe the effects of flap.

2. The derivatives Cu, , $Cx0$, Czu and .;C ,are, weak functions of flap setting, indicating about 20 to 40 % variations for 10 degrees flap change.
The flap sensitivity coefficients $Cxw,$, Cxo_{df}, C_{zud_f} and Cm_{us_f} can be defined but their validity needs to be further confirmed.

3 The derivatives Cx_w,- C_zO, C_{mW}, C,"$_S$and C_m4 do not appear to be functions of flap position, showing a variation of less than 10% for 10 degrees of flap deflection.

The general -trend of the effect of flap- po ition on aerodynamic coefficients is consistently predicted by both the models except for the derivative C_XU,. However based on the criteria of trajectory match,low fit error and run to run consistency, the nonlinear model is considered to be a superior choice compared to the linear model, Hence, the nonlinear model has been used in the final analysis.

6.2 Lift, Drag and Moment analysis:

It is possible generate the lift and drag data for the gross aircraft corresponding to. trimmed and level flight for each of the, twelve flight runs. Two, different approaches, one based on static balance and the other using Taylorseries expansion sum from estimated stability derivatives can be used to generate such basic data.

From the static balance considerations:

$$
\begin{aligned}
1/2\ \rho\ V^2\ S\ C_{L_{trim}} &= L = W \\
1/2\ \rho\ V^2\ S\ C_{D_{trim}} &= D \\
T - D - W\sin\gamma &= 0
\end{aligned}
\tag{14}
$$

From the Taylor stability derivatives:

$$
\begin{aligned}
C_D &= C_{D_o} + C_{D_V}\frac{V}{V_o} + C_{D_\alpha}\alpha \\
C_L &= C_{L_o} + C_{L_V}\frac{V}{V_o} + C_{L_\alpha}\alpha
\end{aligned}
\tag{15}
$$

In Table 9,, the effects of flap setting on the lift and drag coefficientsare presented indicating the. flap setting, angle of attack, velocity, estimated aircraft mass, dynamic pressure and the estimated C_L and Coby staticbalance method and using maximum; lilkelihood estimates of the stability derivatives. The values of lift and drag, coefficients for gross aircraft from both methods match well.

In order to evaluate the flap effects the flight experiments discussed in Section 2 were carried out,at a constant altitude and constant airspeed. This results in a small variation of only 2 degrees (from 5 to ,7 degrees) in angle of attack, mainly' due to the reduction in mass because of fuel consumed. - in order to generate 'gift and drag data over a larger range of angle of attack, a new flight test data base for the same aircraft was used [4l The flight test data was for a fixed flap setting, presumably of -15 degrees and covers a range of 2 5 to 8 degrees of angle attack.

The analysis technique the this new flight datawas similar to the one described in this document. Table 10 gives the details of the angle of attack, velocity, aircraft mass, dynamic p ssure and the estimated lift and drag coefficents using the static balance and via the maximum likelihood estimates of parameters and subsequent Taylor series summation. Again the match between both the method $ excellent.

Lift coefficients as a function of angl attack and flap deflection for both the data sets (Table 9 and 10)are shown in Fig. 7. The figure also provides the manufacturers loci of constant flap at 0 degrees From this figure, following observations can be ma The $C_{L\alpha}$ obtained through . parameter estimation from flight experiments at a fixed flap position of degrees runs parrallel to the manufacturers locus. The locus obtained corresponding to -15 deg of flap logically should be proportionally much lower. This discrepency could perhaps b attributed to lack of precise information about the initia aircraft mass for the second data set.

In presenting this information and in the estimation procedure approximate typical initial aircraft mass has been assumed, to generate qualitative information.

Secondly enlarged inset shown. in Fig. 7 indicates that 10 degrees flap deflection provides a locus which matches with the flap range, but has a very small-mismatch (11.34-0.54=10.8 nstead of 10 deg.) i the numerical values.

A second analysis the form f drag polar plots. C_L V C~ is provided in Fig. The results from the second flight test data base (Table 10) provides a ,segment of drag polar at -15 deg flap. The results from the first data base corresponding to flap sensitivity study (Table 9) provide an orthogonal locus which is consistent with overall drag polar.

The third, aerodynamic characteristic shown in Fig. 9, viz. C_L vs C_{m_0}, also clearly, shows the effects of small flap deflections. The results agree well with the general trend shown in the inset.

Thus all the three static characteristics conform to the typical behaviour associated with flap deflection. nonlinear model, directly in terms of non-dimensional coefficients with dynamic pressure' as an additional variable in the estimation procedure, yields realistic values of the lift and drag coefficients. T above results notonly help to validate the nonlinear estimation from,the viewpoint of flight mechanics,- but also serve to demonstrate' the utility of more 'accurate-' models' in practice.

The results presented indicate that the linear model in terms of dimensional derivatives is adequate to predict the modes and mode shapes of the aircraft motion. However, when the estimated dimensional derivatives are further transformed and used to compute the primary aerodynamic coefficients like C_L, C_D, C_m they lead to higher values for the above basic parameters.

On the other hand, the nonlinear model, in terms of non-dimensional coefficients and with dynamic pressure as an additional variable, provides significant improvements, both qualitatively and quantitatively. The typical aerodynamic characteristics, such as C_L **VSO(,,** C_L vs C. and C_L **VS** C_{M_0} estimated are in a reasonably good agreement with the basic data. The trend of variations of the above primary parameters, as the flap position is varied from 0.54 to 11.34 degrees, is also consistent. Further, the nonlinear model predicts the dynamics of aircraft more accurately as observed from the excellent trajectory match.

It has been demonstrated that certain flap sensitivity derivatives, defined as variations of aerodynamic coefficient with respect to parameter under investigation, in this case the flap position, can be identified from planned flight test experiments.

8. Acknow7edg n-=

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Table 1 List of Variables Recorded During Flight Test

Symbol	Variable	Units
T	Time	sec
p	Static pressure	
V	True air speed	m/s
α	Angle of attack	rad
β	Angle of sideslip	rad
p	Roll rate	rad/s
q	Pitch rate	rad/s,
r	Yaw rate	tad/s
θ	Pitch angle	rad
ϕ	Roll' angle	rad
a_x	Longitudinal acceleration	m 2
a_y	Lateral acceleration	m/s2
a_z	Normal acceleration'	m/s2'
$\dot{p}$	Roll acceleration	rad/s2
$\dot{q}$	Pitch acceleration	rad/s2
$\dot{r}$	w acceleration	rad/s2
δ_e	Elevator position	rad
δ_a	Aileron position	rad
δ_r	Rudder position	rad:
δ_f	Flap position	rad
$\bar{q}$	Dynamic pressure	p
EPR	Engine Pressure; Ratio	

Table 2 Aircraft Data,:

Initial mass		**8170 kg**
Moment of Inertia		91094 kg m
Wing area		3
Chord length		2. m
Nose boom offset		10 **992 m**
distances from		0 0
	z_n =	**0 5 6 to**
Accelerometer		**0 3 5 m**
offset		**0.0 m**
distances from		**-0. 1 m**
Location of jet	**tx**	**-2.6** m
engines w.r.t. c.g		**-0.50 m**
Tilt ang	**engines a'.$_T$**	**3 deg**

Table 3 : Details of Flight tests

Run No.	Flap position (deg)	Fuel Consumed (kg)
1	0.54	
2	0.54	
3	2.64	
4	2.64	
5	**4.76**	728
6	**4.76**	
7	7.22	
8	7.22	
9	9.02	905
10	9.02	
11	11.34	980
12	11.34	

The flight tests are carried out at-an airspeed of 105 m/sec and altitude of 5000 m approximately.

Table 4 Estimation of Measurement Errors, Scale Factors, and Initial Conditions by Data Compatibility Checking

	Run 1	Run 2	Run	Run 4	Run 5	Run 6
K_9	0.9920	0.9620	0.9880	0.9760	0.9590	0.9940
Kq	0.8640	0.9900	1.0040	1,0480	1.1040	1.0120
4a x	0.0030	0.0100	0.1060	0.0210	-0.0540	0.1620
d	-0.1779	-0.0107	0.1370	-0.2740	0.6890	-0.1620
d,	0.0090	0.0430	0.0110	0.0460	0.0090	-0.0030
QP	0.0015	0.0011	0.0014	0,0012	0.0005	0.0009
Aq	0.0006	0.0003	0.0007	0.0002	0.0004	0.0007
a	-0.0045	-0,0039,	0.0022	-0.0036	0.0056	-0.0014
AV	8.5010	9.7200	8.1390	12.278	8.2620	10.430
Aq	546.30	990710 -	-134.20	58.710	-576.70	31.910
u_0	103.30	100.90	103.40	98.211	101.40	102.71
V_o	0,1490	0.5440	-1.7860	0.1420	-1.1580	-0.0240
	12.774	12.409	12.876	11.099	12.471	10.969
	0.0370	0.0210	0.0580	0.0020	0.0220	-0.0290
	0.1101	0.0970	0.1110	0.1020	0.1070	0.0701

TABLE 5 DIMENSIONAL DERIVATIVES ESTIMATED USING LINEAR MODEL

Flap (deg)	0.54	0.54	2.64	2.64	4.75	4.15	7.22					
	-0.0095 1.5	-0.0054 6.2	-0.0092 1.4	0.0029 6.2	-0.0111 6.1	-0.0077 2.5	-0.0122 2.3	-0.0061 2.6	-0.0114 1.9	0.0132 0.8	-0.0026 3.6	-0.0106 1.2
X_w	0.1170 0.8	0.1049 1.6	0.1177 0.5	0.0730 2.3	0.1121 1.1	0.1134 1.1	0.1141 0.7	0.1063 1.2	0.1016 1.2	0.1151 0.8	0.0462 2.7	0.0962 0.8
X_o	-0.2500 1.4	-0.2603 1.5	-0.3460 0.5	-0.4484 0.6	-0.3556 1.3	-0.4293 1.0	-0.4515 0.6	0.4109 0.8	-0.5051 0.8	-0.445 0.6	0.6415 4.4	-0.5215 0.4
Z_u	-0.0927 0.9	-0.1191 1.3	-0.1175 0.7	-0.1193 0.8	-0.1537 2.1	-0.1302 0.7	0.1297 1.5	-0.1271 0.6	-0.1427 0.9	-0.1154 0.6	-0.1398 0.5	-0.1425 0.6
w	-0.7779	-0.8112 0.7	-0.8096 0.4	-0.8371 0.5	-0.6257 1.0	-0.7933 0.7	-0.82 4 -0.6	-0.8137 0.6	-0.8352 0.8	-D.8723 0.8	-0.8324 0.6	-0.8940 0.5
Z_o	-9.4456 0.3	-9.5037 0.3	-9.8301 0.1	-9.1829 0.2	-9.4028 0.3	-9.5742 0.3	9 6905 0 2	-9.8362 0.2	-10.120 0.3	-9.8743 0.2	-;9.3460' 0-.2	9.8332: 0.2
M_u	0.0055 2.0	0.0085 4.1	0.0054 1.0	0.0055 1.6	0.0061 3.7	0.006	0.0046 2.1	0.0050 2.0	0.0079 4.8	0.0049 1.4	0.0051 1.6	0.0044 1.8
M_w	-0.0303 1.8	-0.0165 5.3	-0.0351 0.7	-0.0'1323 1."	-0.0295 2.3	-0.0272 2.3	-0.0335 1.1	-0.0303 11.5	-0.0195 4.2	-0.0338 1.2	0.0328 1.1	-0.0318 1.2
M_q	-2.3281 2.7	-3.6858 4.9	-1.5991 1.5	-2.0590 2.i	-1.9226 3.8	-2.4985 3.8	-1.8135 2.1	-2.1075 2.6	-3.7231 5.3	-2.2604 1.8	-2.0896 3.9	1.8674" 2.3
M_o	-0.0122 15.6	-0.0467 9.7:	0.0155 4.8	0.0261, 3.3	0.0265 7.0	0.0622 1.7	0.0752 1.1	0.0591 1.6:	0.1003 1.2	0.0825 1.0	0.1194 0.8	0.0924 0.7
M_{δ_e}	-6.2653 1.8	-7.0451 3.6	-6.2291 0.8	-6.4529 1.2	-5.0489' 2.6	-6.1937 2.6	-6.1902 1.2	-6.3297 1.5	-6.0349 4.2	-6.8618 1.2	-6.6000 1.1	-6.3243 1.2

* % Standard deviations

TABLE 6 NON-DIMENSIONAL DERIVATIVES CONVERTED FROK DIMENSIONAL DERIVATIVES ESTIMATED USING LINEAR MODEL (REFER-TABLE 5)

Flap (dog)	0.54	0',54'	2.64	2.64	4 5	4.75	7.22	7.22	9.03	9.03	11.34	11'.'34
C- xu	-0.0593	**-0.0336**	-0.0569	-0.0178	-0.0675	-0.0463	-0.0730	0.0364	**-0.0674-**	0.0775	-0.0152	-0.06 19
C --xw	0.7311	0.6532	**0.7285**	**4483**	**0.6824**	0.6822	0.6834	0.6331	**0.6010**	**0.6782-**	0-2716	**0.56 19**
C xo	-0.0152	-0.0157	-0.0208	-0.0267	-0.0210	-0.0251	-0.0263	-0.0238	-0.0290	-0.0255	-0.0365	_0_0296
C zu	-0.5793	**-0.7416**	-0.7272	-0.7328	-0.9357	0.7834	-0.8671	**-0.7569**	**-0.8440**	-0.6806	-0.8201	**0.8324**
C sw	-4.8612	-5.0507	-5.0111	-5.1418	-3.8092	-4.7732	**-4 9615**	**4 8462**	**- 4 9403**	**-5.1398**	4.8830	**_-5.2225**
Z	-0.5731	-0.5745	**-0.5907**	**0.5476**	-0 **5556**	**0,5593**	**5635**	**p.5688**	**-O 5811**	**-0 5649**	-0.5323	-0.55 77
C	-0.1674 -	0.2588	**0.1644**	**0.1674**	**0.1857:**	**0.1918**	0.1400'	**11.1522**	**0.2374**	0 1492	0.1553	**p.1339**
C mw;	-0.9224	-0.5023	10685	-0.9833	-0.8980	-0..8280	-1.0198	**-0.9224**	**-0.5936**	-1.0289	-0.9985	-o .9680
C mq	-58.330	-92.346	-40.065	-51.587	-48.165	-62.600	-45.437	-52.803	**-93.281**	-56.634	-52.354	_4 6.787
mo	-0.0036	-0.0120	0.0046	0.0078	0.0078	0.0184	0.0222	0,0175	**0.0246**	**0.0244**	**0.0353**	0.0273
C	-1.8517	**-2.0822**	-1.8410	-1 9101	-1.4922	-1.8305	-1.8295-	-1.8707 .	1.7836	-2.0280	**-1.9506**	1.86 91

TABLE 7 : Non Dimensional Derivatives Estimated Directly Using Nonlinear Model

Flap (deg)	0.54	0.54	2.64,	22.64	4.75	4.65	7.22	7.22	9.03	9.03	11.3,	11.34
C_{xu}	-0.0336'	-0.0223	-0.0111	0.0033-	0.0059	0.0094	0.0015	0.0173	0.0145	0.0080	0.0176	0.0147
	4.5 *	6.9	10.9	32.1	67.1 .	20.5	95.2	7.4	11.5	12.1	6.7	6.1
C_{xw}	0.7113 -	0.7405.	0.1313	0.6693	0.6411	0.6920	0.6351-	0.6200	0.5945	0.5712	0.5345	0.5993
	0.8	1.0 -	0.6	0.6	1.3	12.9	0.7	0.9.	1-0	0.8	0.9	0.7
C_{x0}	-0.0744	-0.0902	0.10.49	-0.1014	-0.1108	-0.1166	-0.1042		-0.1126	-0.1017	-0.1105	-0.1106
	2.0	2.0	1.1	1.2	3.6	1.6	1.3	1.3	1.5	1.0	1.1	0.8
C_{zu}	0.4844	0.4590	0.4935	0.4068	0.4623 '	0.3533	0.4142	0.3939	0.2903	0.3677	0.3831	0.3456
	1.5	1.7	1.3	1.4	4.0	2.7	2.0	1.8	3.4	1.7	1.9	1.7 -
C_{zw}	-4.4262	-4.8759	-4.4890	-4.5900	-4.5326	-4.4517	-4.5539	4.7404	-4.6017	4.6563	-4.6093	-4.7913
	0.7	0.7	0.5	0.5	1.0	0.8	0.6	0.6	0.7'	0.6	0.6'	0.5
C_{zo}	-0.4993	-0.4129	-0.5356	0.431 û	0.5119	-0.4106	-0.4926	-0.4493	-0.3746	-0.4501	-0.5055	-0.4462
	1.4	2.0	1.1	1.4	3.6	2.1	1.6	1.5	2.6	1.5	1.5	1.3
C_{mu}	0.1100	0.1002	0.1085	0.0952 ;	0.0936 -	0.0834	0.0745	0.0751	0.0564'	0.0649	0.0696 .	0.0562
	1.7	2.0	1.4	1.7	5.2	2.8	2.5	2.1	4.1	2.5	2.7	2.6
C_{mw}	-1.0857	-1.0917	-1.0864	-1.0671	-1.0941	-1.0987	-1.0426	-1.0472 :	1.0620	1,0777	-1.0749	-1.0856
	0.5	0.6	0.4	0.4	0.8-	0.7	0.5	0.5	0.6	0.5	0.5	0.4
C_{mq}	-25.973	-26.496	-31.253	30.767	-25.825	29.017	-29.327	28.852	-28.698	-30.968	-31.572	-29.046
	1.7-	1.6	1.0	1.1'	2.5-.	2.1	1' 3 -	1.5	1.9	1.;4	1.4	1.3
C_{mo}	0.0251	0.0353	0.0296	0.0368	0.0454,.	0.0535-:	0.0568	0.0548	0.0755	0.0677	0.0601	0.0727
	7.8	6.3	5.3	4.7	11.0-	4.5	3.4	3.0	3.2	2.5	3.3	2.1
C	-1.5424.	-1.6210	-1.6786	1.7949	-1.5871	-1.6659	1.6358	-1.6008 .	-1.5722	1.6668	-1.6800	-1.6108
			0.5	0.5	1.0	0.9	0.6	0.6	0.8	0.6	0.6	0.6

% Standard deviation

Table 8: Linear Regression fit (y=A+Bx) to estimated Non-Dimensional:Derivatives-(Table 7) as a Function of Flap Deflection 4 (Absolute and Normalized `Values.)

	Derivative		Flap Sensitivity Coefficient	Normalized Variation' per degree flap deflection (B/A)*100	% Normalized variation over the experimental range of 10 degrees 10*(B/A)*100
	Cx	-0.02014	0.00366	18 17	**181.7**
2	C_{xw}	0.73771	-0.01566	2.12	**21.2**
3	Cxo	-0.09281	-0.00197	2.12	**21.2**
4	C_{x_A}	0.47456	-0.01183	2.50	**25.0**
5	Cxw	4.55594	-0.00911	0.20	2.0
6	C_{zo}	-0.46772	0.00130	0.28	2.8
7	C_{mu}	0.10953	0.00460	4.20	42.0
8	C_{mw}	-1.08563	0.00160	0.15	1.5
9	C_{mq}	-27.54030	-0.24367	0.89	**8.9**
10	C	0.02753	0.00398	14.46	**144.6**
11	C_{M,δ_e}	-1.60128	-0.00339	0.21	**2.1**

Table 9: Trimmed lift and drag coefficients of aircraft as a function of flap deflection (+) (By static balance and by Taylor series summation of the maximum likelihood_ estimates of derivatives)

Run	deg	deg	V ml	Kg	q P				
	0.54	7.05	104.7	7670	4320	0,578	0.543	0.085'	0.090
2	0.54	7.05	103.7	7642	4270	0.579	0.551	0.072	0.089
3	2.64	6.70	103.2	7597	4240	0.575	0.540	0.099	0.097
4	2.64	6.70	103.7	7539	4270-	0.571'	0.550	0.079	0.091
5	4.75	6.30	102.7	7472<	4140	0.566	0.576	0.093	0.093
6	4.75	6.30	105.1	7385,	4350	0.560	0.533	0.081	0.091
7	7.22	6.10	103.0	7351	4230	0.557	0.575	0.103	0.094
8	7.22	6.10	104.7	7310	4330	0.554	0.573	0.082	0.091
9	9.03	5.56	104.4	`7260	4320	0.550	0.538	0.092	0.090
10	9.03	5.56	103.7	7232	4270	0.548	0.520	0.086	0.089
11,	11.34	5.20	103.1	7200	4245	0.545	0.511	0.093	0.093
12	11.34	5.20	103.7	7170	4300	0.543	0.536	0.086	0.089

(+) Flight test at constant speed and altitude with different flap settings

f C $_{D}$S' Lift and drag coefficients obtained by mass balance (Equationl4, pp 19)

C_{L}, $C_{D\,ML}$: Lift and drag coeccifients obtained as Taylor series sum of individual estimated coefficients using parameter estimation from flight test data. (Equation 15, pp19)

Table 10: Trimmed lift and drag coefficients of aircraft as a function of angle of attack (*)
(By static balance and by Taylor series summation of the maximum likelihood estimates of derivatives)

Run No.	oG (deg)	V m/s	Kg	q Pa	CL e	CL.,	CDs	C DML
1	8.0	101.0	7600	3860	0.64	0.626		0.096
2	8.0	100.0	7550	3740	0.65	0.623	0.090	0.098
3	7.2	107.0	7491	4270	0.57	0.577		0.087
4	6.3	108.7	7455	4400	0.55	0.505	0.090	0.083
5	5.6	117.5	**7412**	5190	0.47	0.468	0,068	0.073
6	5.3	118.5	7344	5250	0.45	0.430	0.071	0.071
7	4.2	131.5	7308	6580	0.36	0.353	0.061	0.062
8	3.7	138.7	7270	7310	0.32	0.316	0.060	0.060
9	2.6	159.5	7212	9710	0.24	0.210	0.046	0.050

(*) : Flight tests at different filght conditions with constant flap deflection

C_{L_S} ; C_{D_S} : Lift and drag coefficients obtained by mass balance (Equation 14, ppl9)

C_{LK} ; C_{DML} Lift and drag coeccifients obtained as Taylor series sum of individual estimated coefiicients using parameter estimation from flight test data. (Equation 15, ppl9)

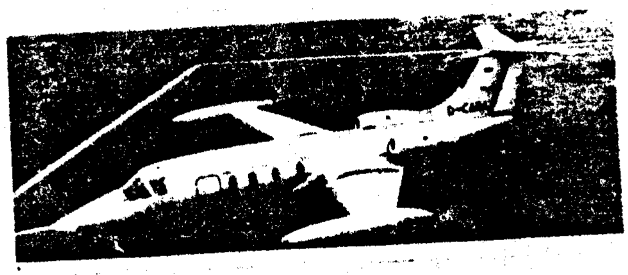


Fig. : **Test Aircraft**

Run.no.	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	10	[illegible]	[illegible]
Fuel Consumed(K9	530		600	655	[illegible]	795	[illegible]	865	905	940	970	995

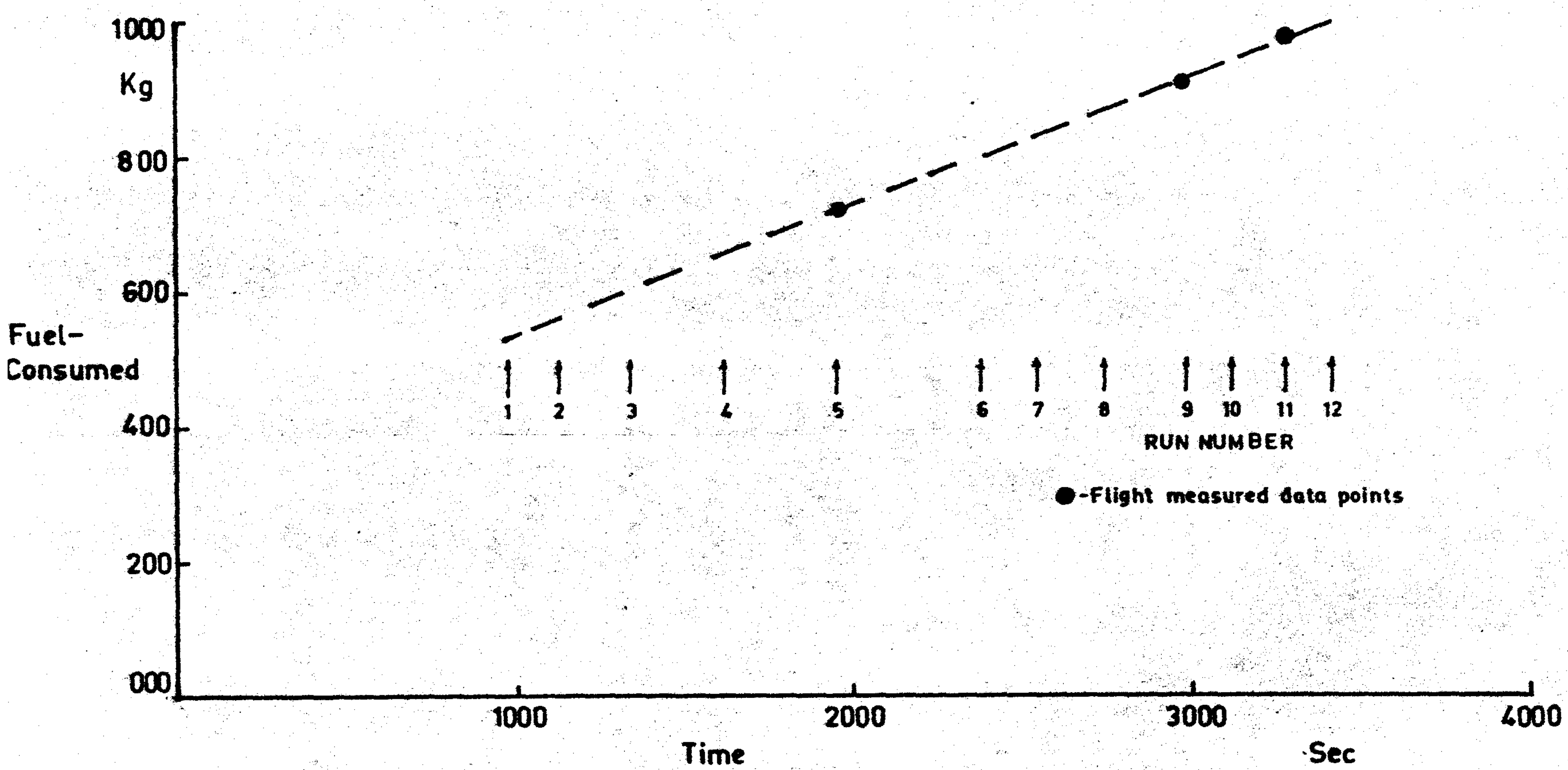


Fig.2 Details o Fuel consumption

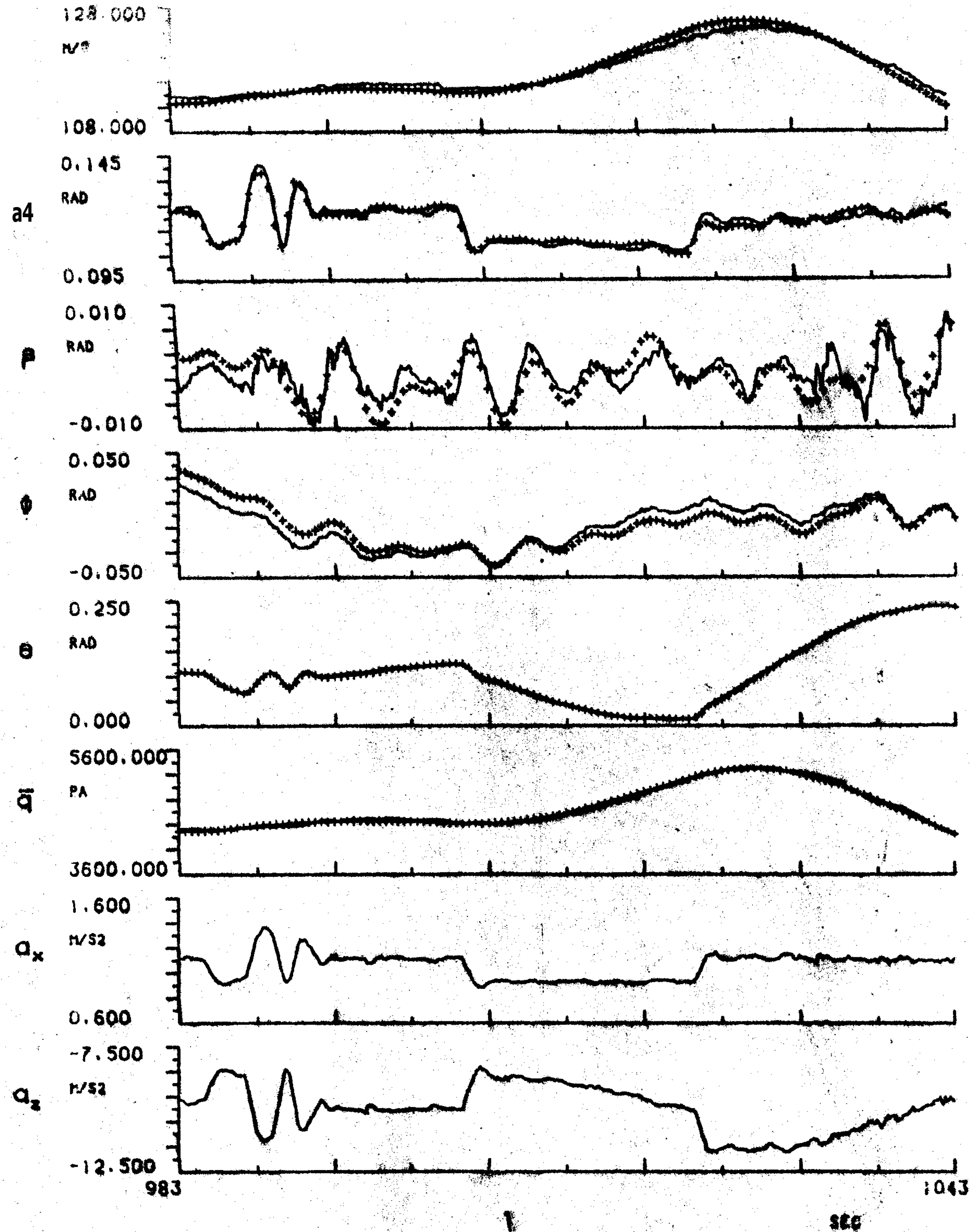


Fig. 3 (i)

Fig. 3 : CuIrYR.Bit# Pro*. to Co tibUltp Check of Aixcraf light Toot D*t4 4- - Meaeor* , +t+ R ttucted)

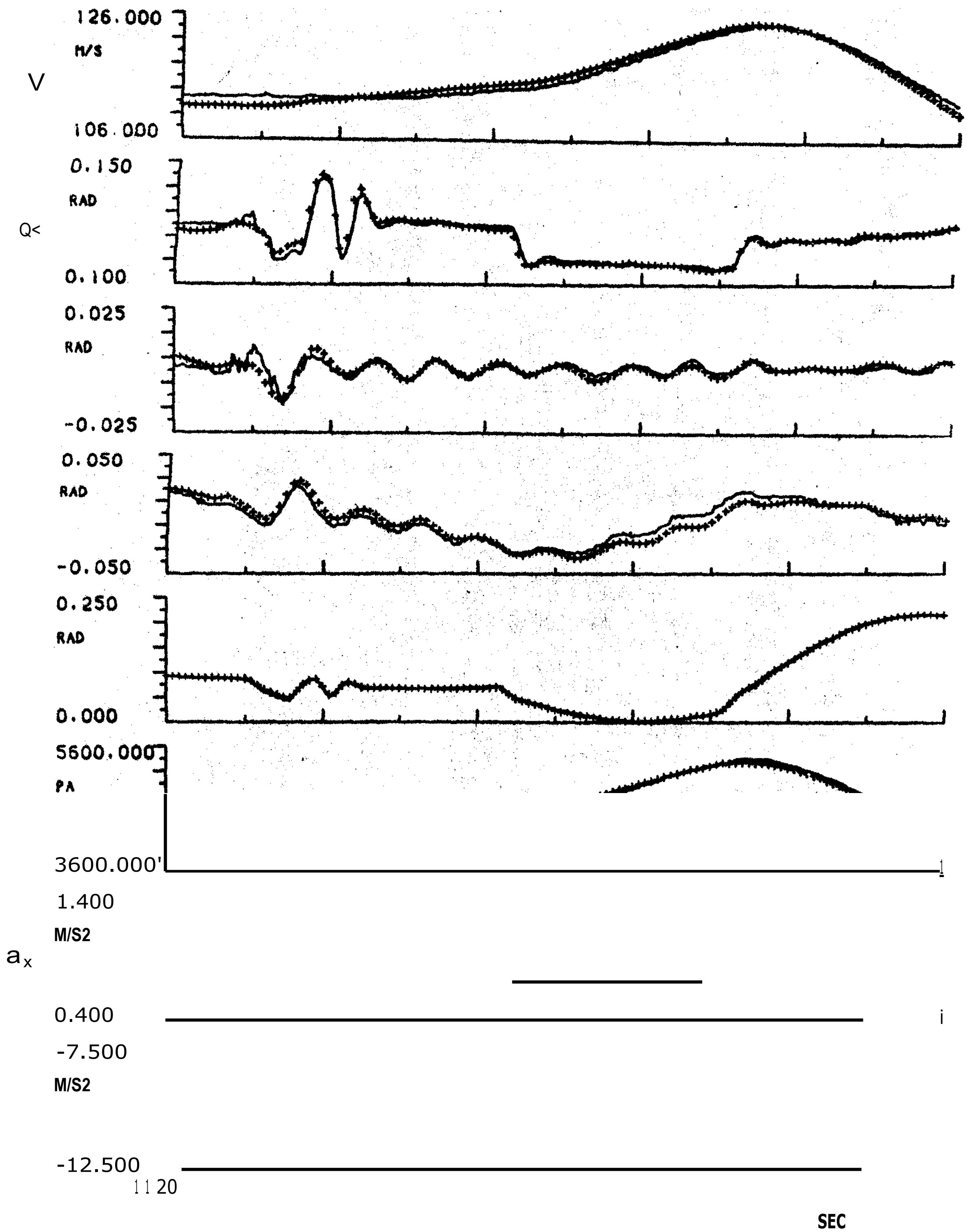

126.000
M/S
V
106.000
0.150
RAD
Q<
0.100
0.025
RAD
-0.025
0.050
RAD
-0.050
0.250
RAD
0.000
5600.000
PA
3600.000'
1.400
M/S2
a_x
0.400
-7.500
M/S2
-12.500
11 20
SEC

Fig. 3 (i)

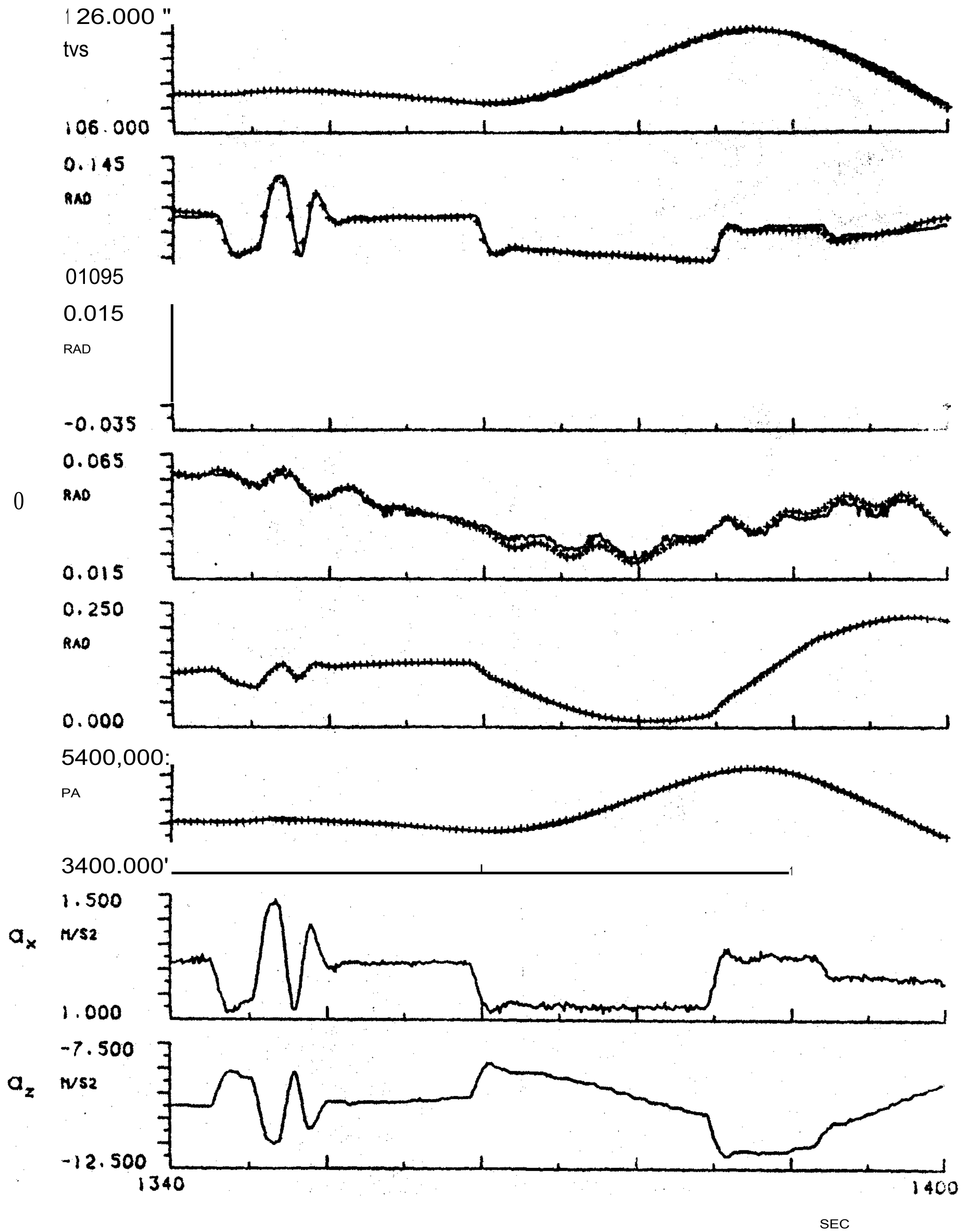
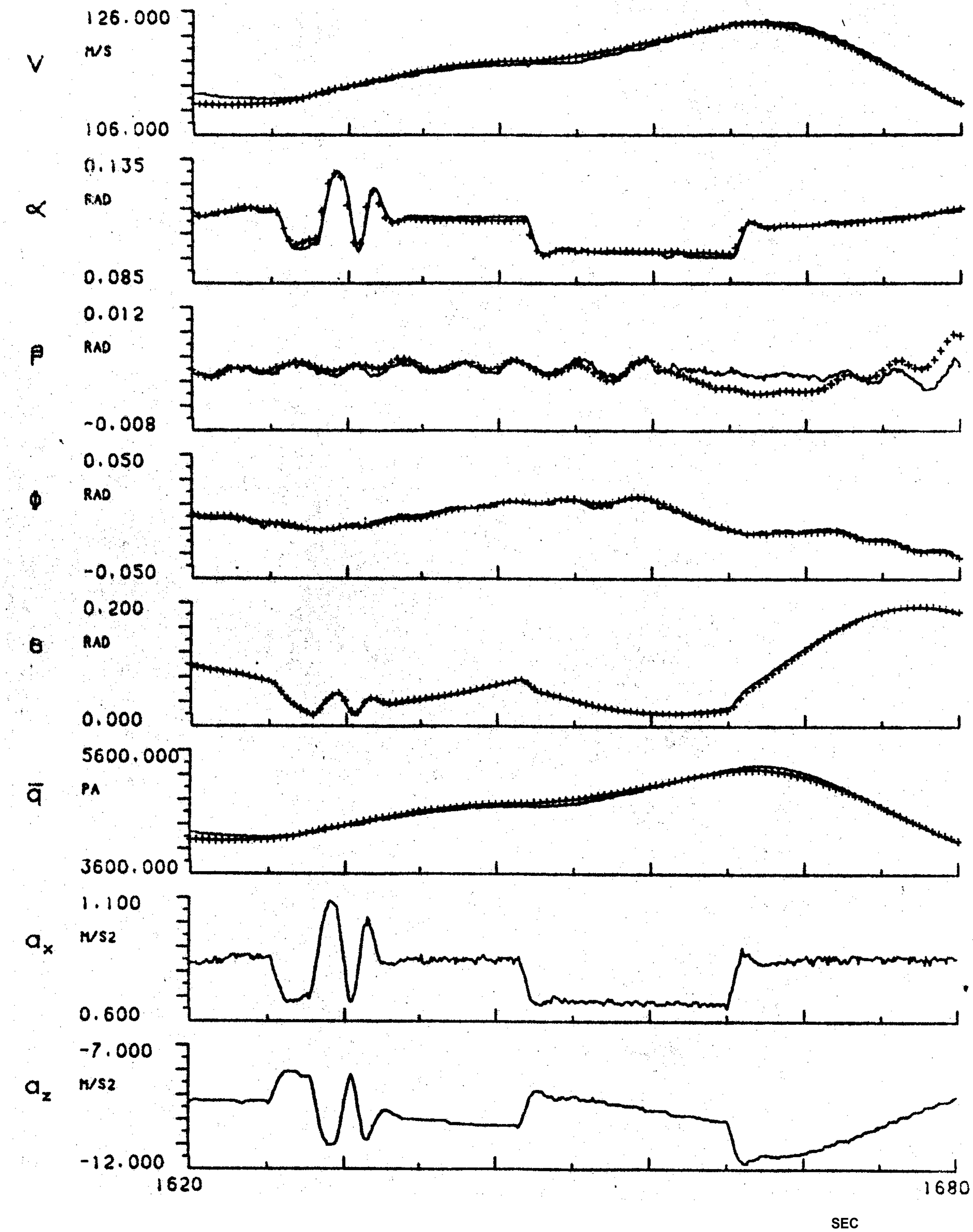
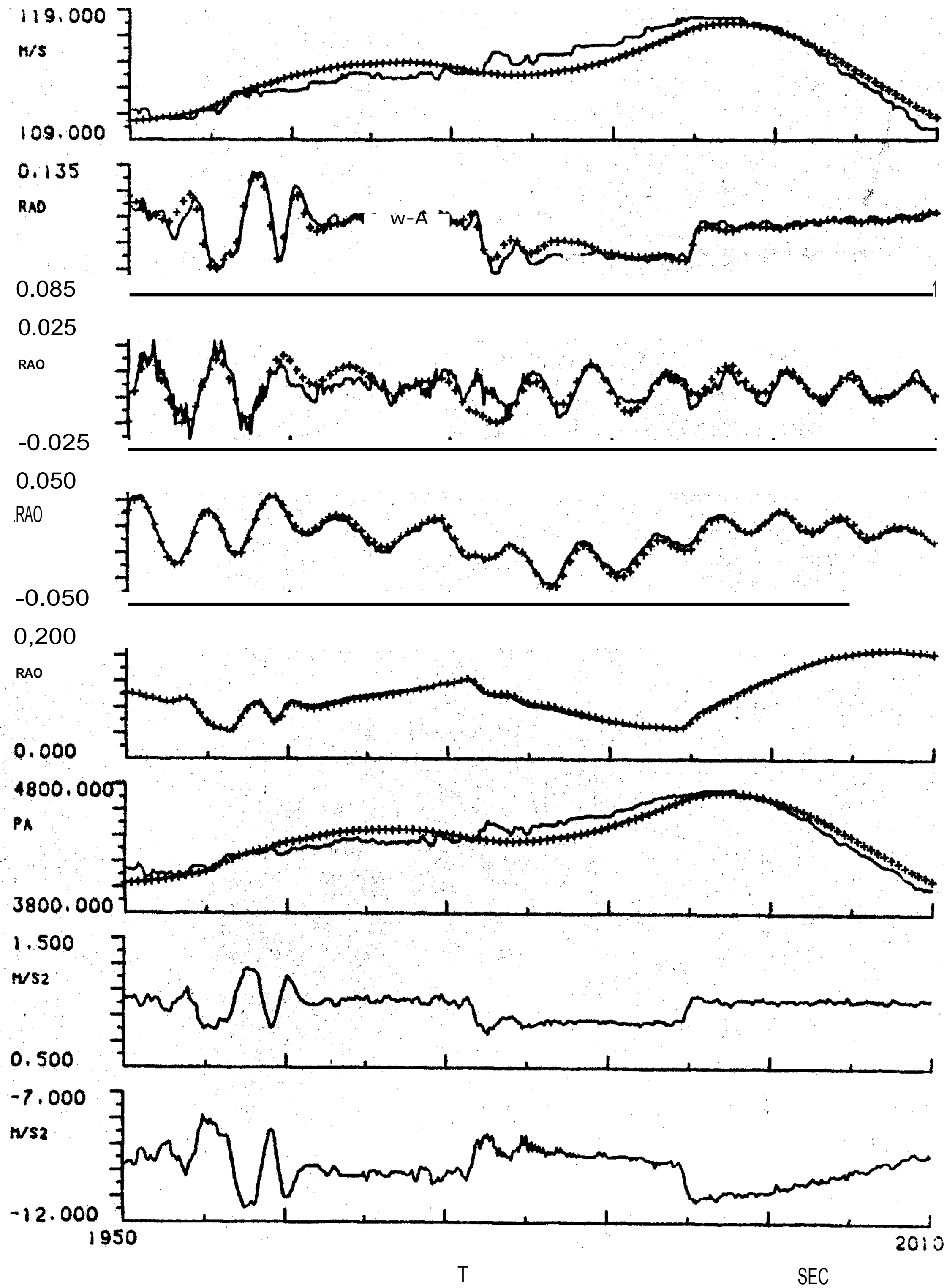

126.000 "
tvs
106.000
0.145
RAD
01095
0.015
RAD
-0.035
0
0.065
RAD
0.015
0.250
RAD
0.000
5400,000:
PA
3400.000'
1.500
a_x
M/S2
1.000
-7.500
a_z
M/S2
-12.500
1340
1400
SEC

Fig. 3 (iii)



V
126.000
M/S
106.000
α
0.135
RAD
0.085
β
0.012
RAD
-0.008
φ
0.050
RAD
-0.050
θ
0.200
RAD
0.000
q̄
5600.000
PA
3600.000
ax
1.100
M/S2
0.600
az
-7.000
M/S2
-12.000
1620
1680
SEC

Fig. 3 (iv)



119.000
M/S
109.000
0.135
RAD
w-A
0.085
0.025
RAO
-0.025
0.050
RAO
-0.050
0,200
RAO
0.000
4800.000
PA
3800.000
1.500
M/S2
0.500
-7.000
M/S2
-12.000
1950
2010
T
SEC

Fig

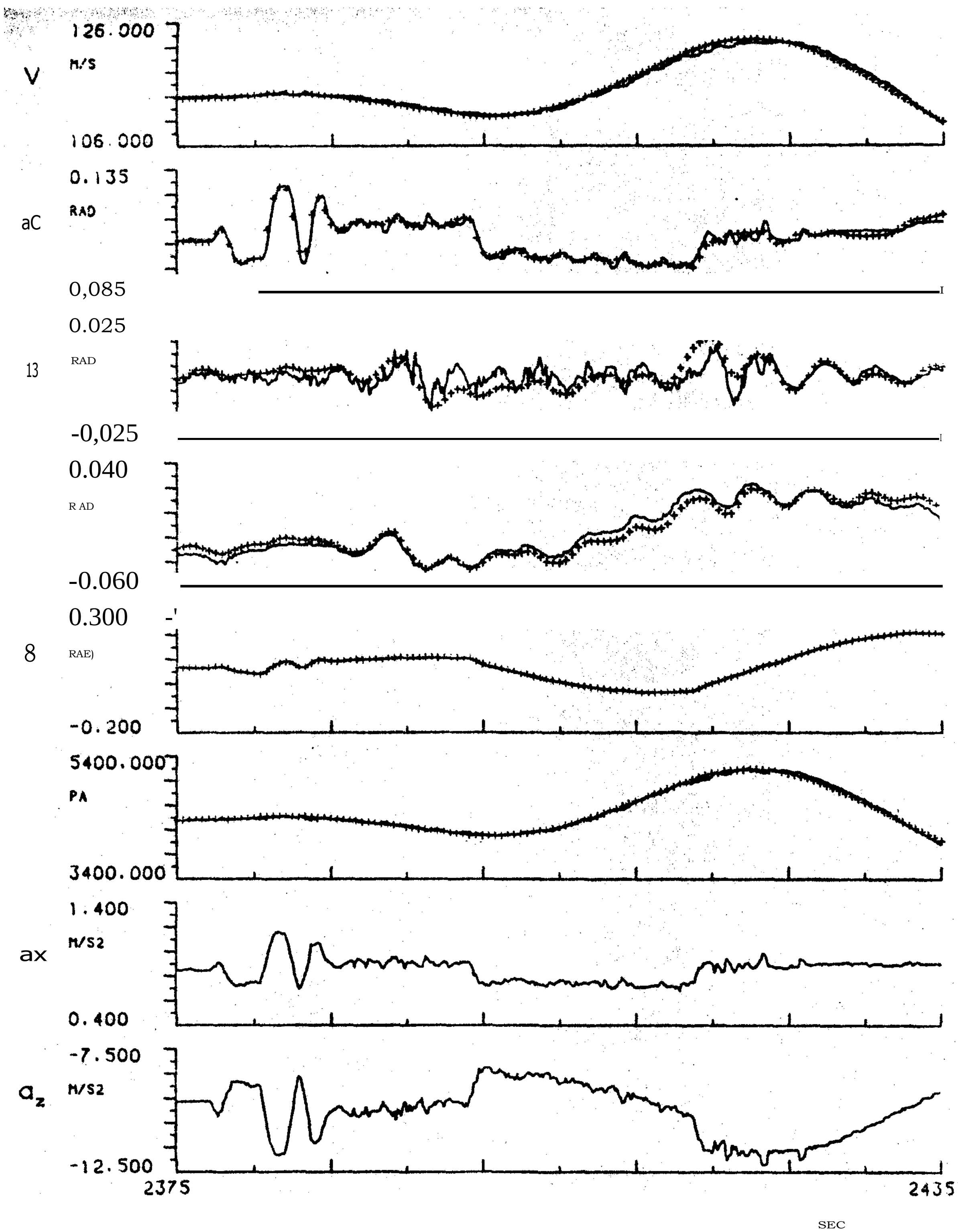

126.000
M/S
V
106.000
0.135
RAD
aC
0,085
0.025
RAD
13
-0,025
0.040
RAD
-0.060
0.300
RAE)
8
-0.200
5400.000
PA
3400.000
1.400
M/S2
ax
0.400
-7.500
M/S2
a_z
-12.500
2375
2435
SEC

Fig. 3 (vi)

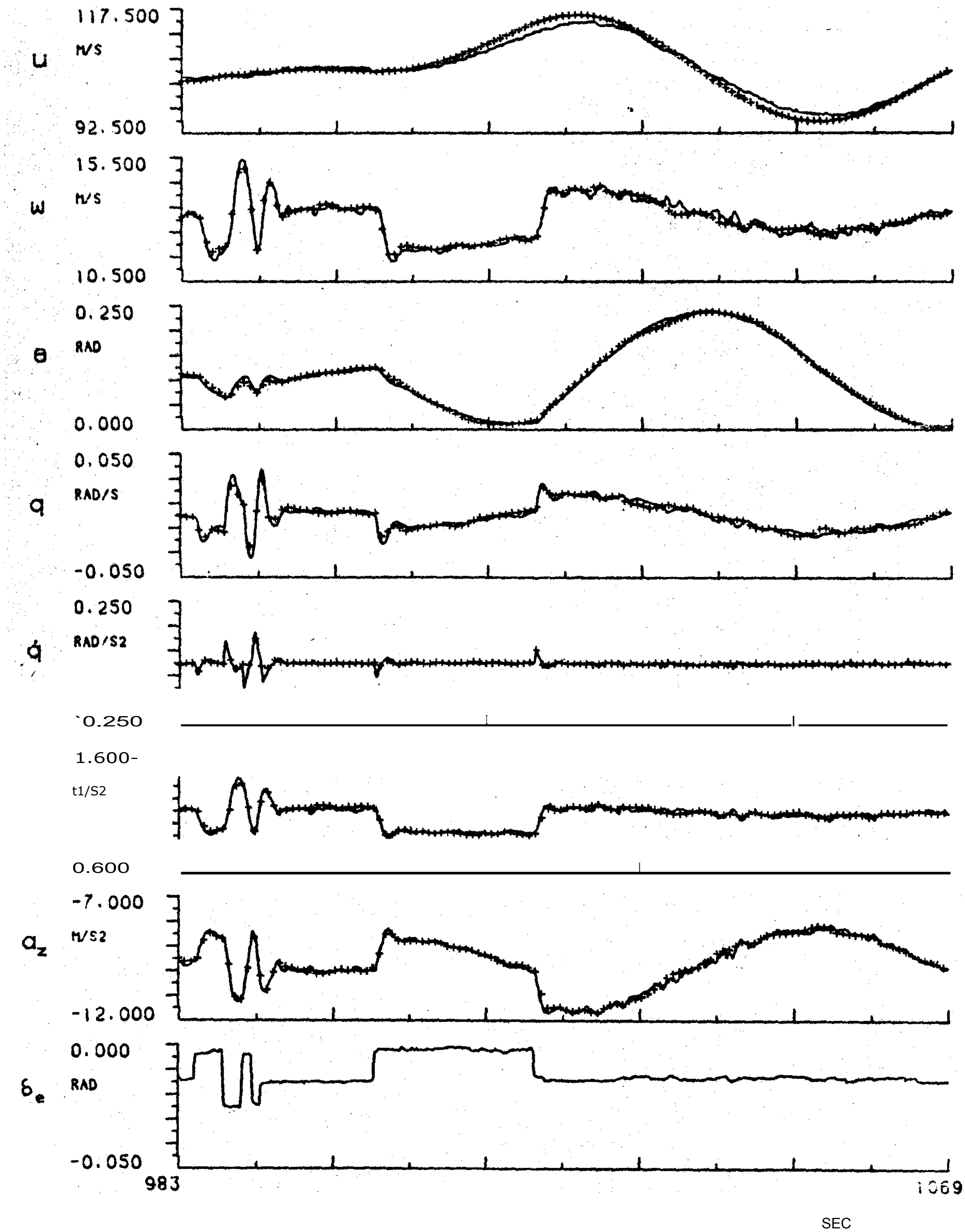


Fig. 4 (i)

Fig. 4 **Curve Fits from Parameter Estimation Using Dimensional Coefficients and Linear Model**
(—— Measured, ++++ **Estimated)**

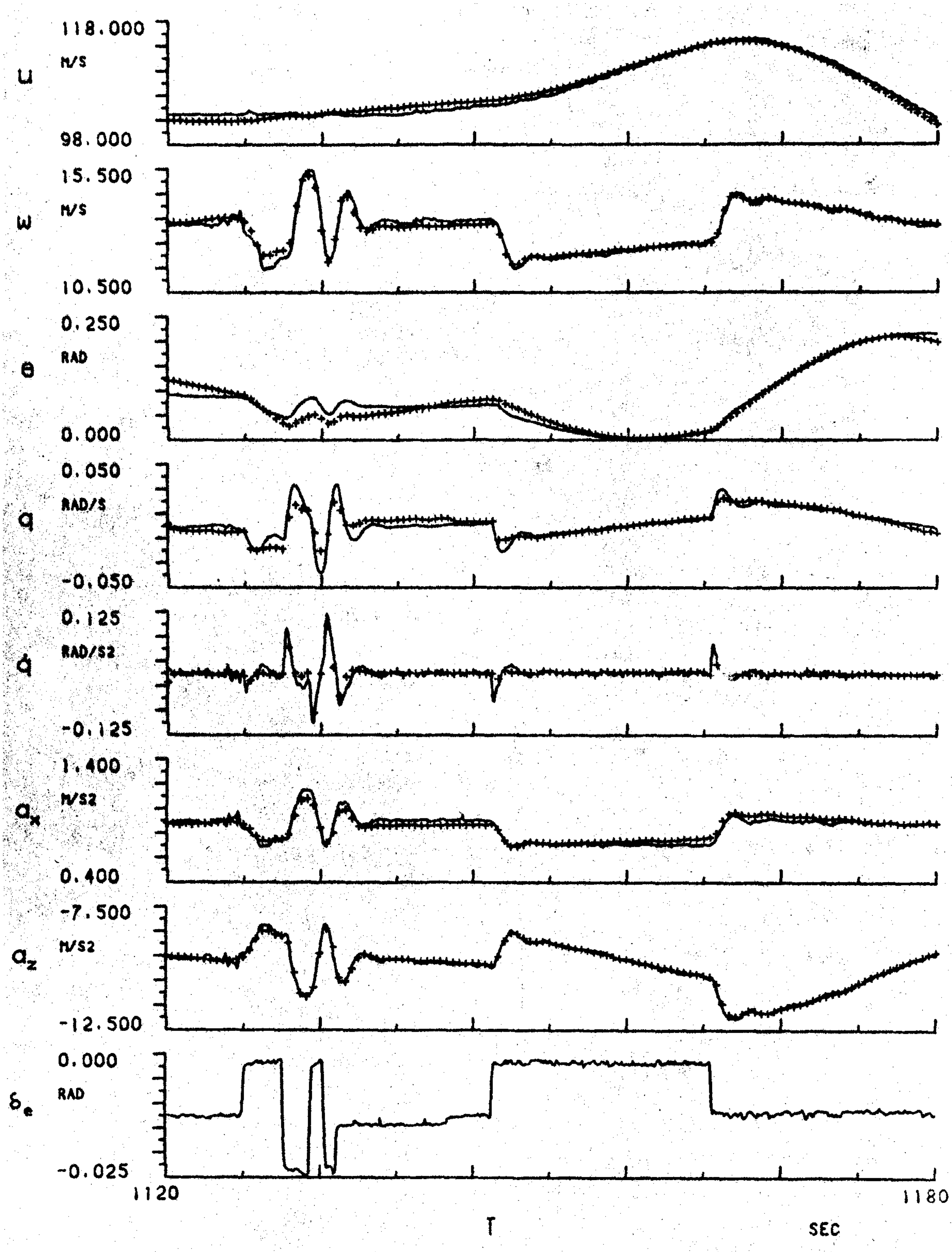
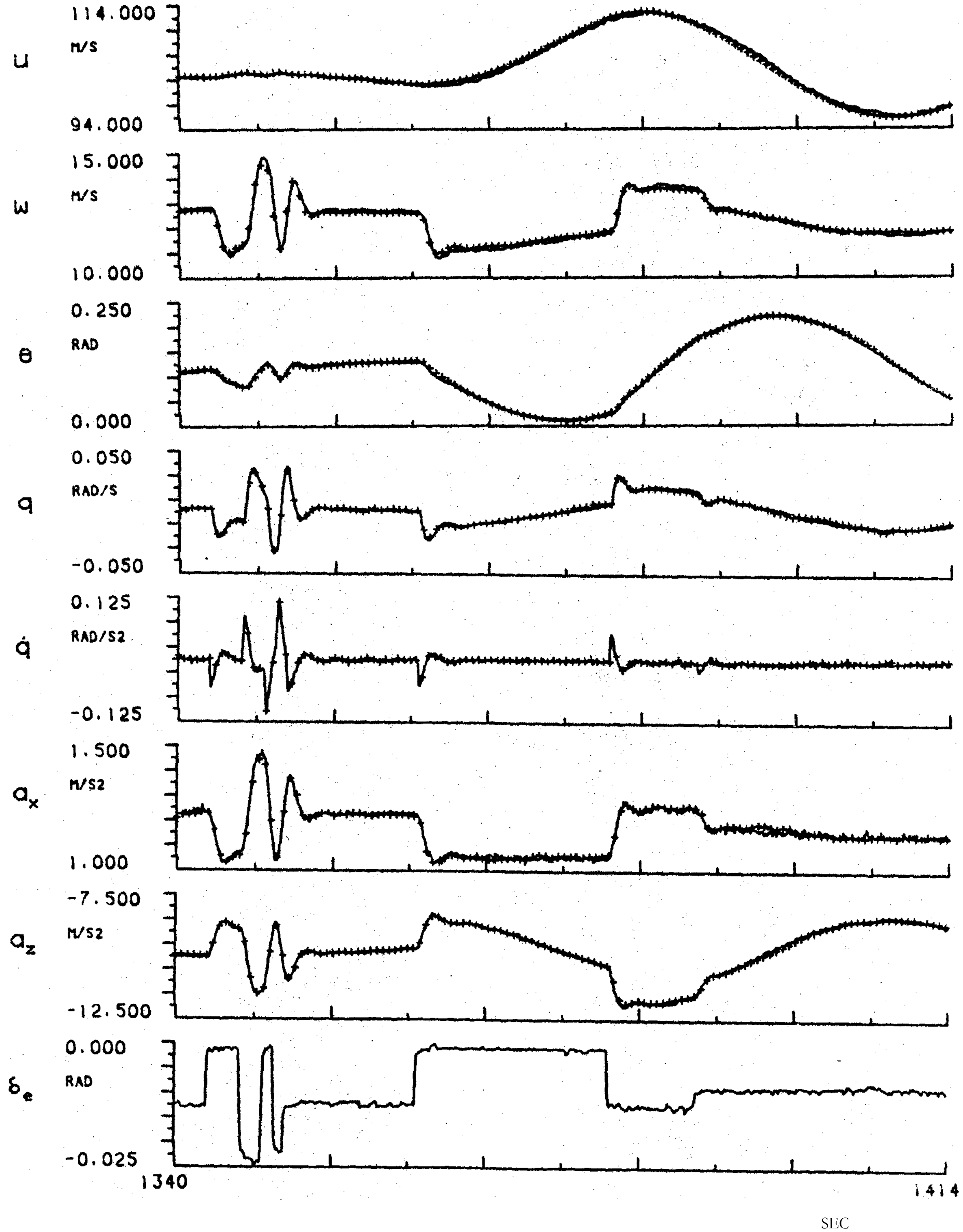
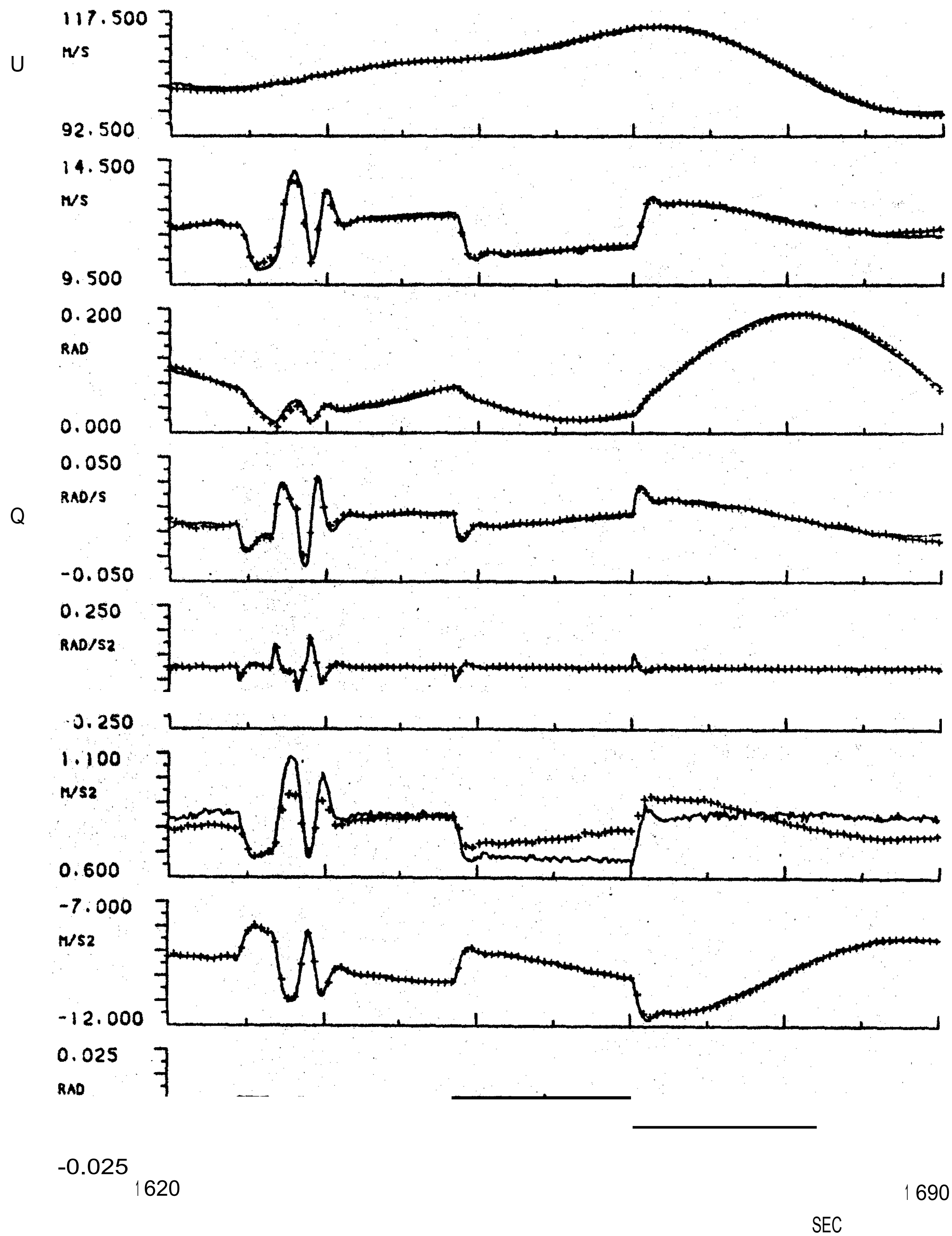

118.000
M/S
u
98.000
15.500
M/S
w
10.500
0.250
RAD
θ
0.000
0.050
RAD/S
q
-0.050
0.125
RAD/S2
q̇
-0.125
1.400
M/S2
a_x
0.400
-7.500
M/S2
a_z
-12.500
0.000
RAD
δ_e
-0.025
1120
1180
T
SEC

Fig. 4 (ii)

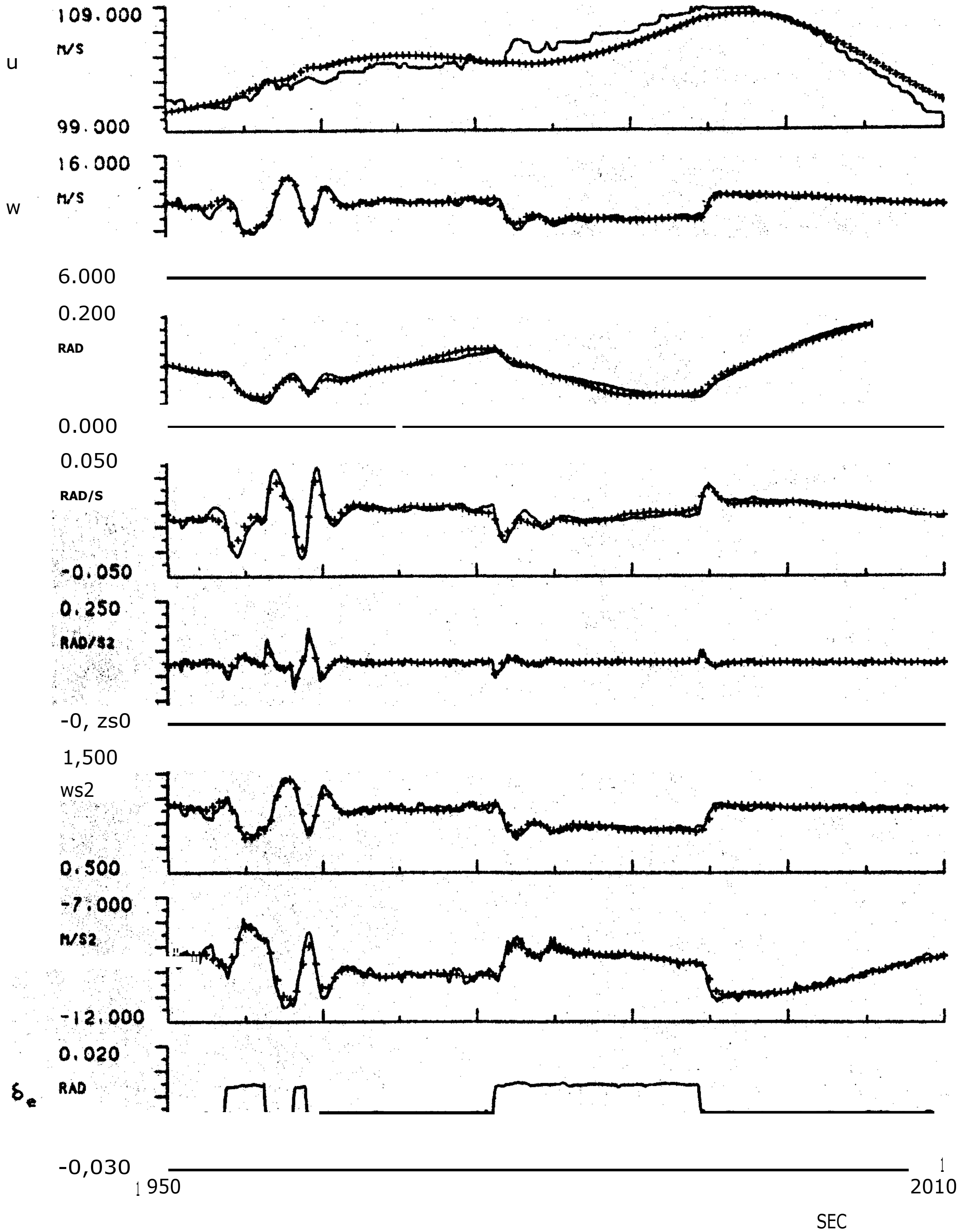

u
114.000
M/S
94.000
w
15.000
M/S
10.000
θ
0.250
RAD
0.000
q
0.050
RAD/S
-0.050
q̇
0.125
RAD/S2
-0.125
a_x
1.500
M/S2
1.000
a_z
-7.500
M/S2
-12.500
$δ_e$
0.000
RAD
-0.025
1340
1414
SEC

Fig, **4 (iii)**

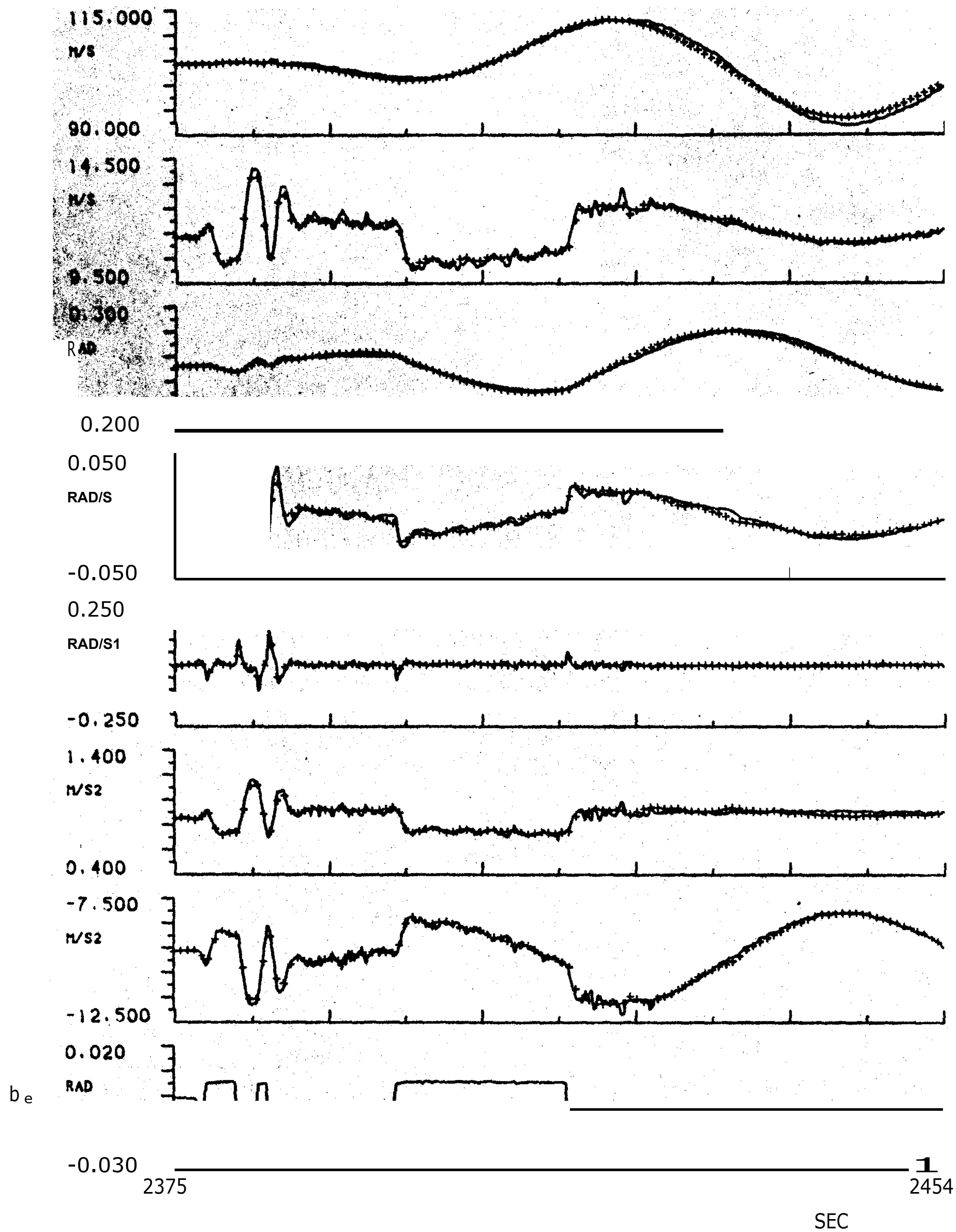


U
Q
117.500
M/S
92.500
14.500
M/S
9.500
0.200
RAD
0.000
0.050
RAD/S
-0.050
0.250
RAD/S2
-0.250
1.100
M/S2
0.600
-7.000
M/S2
-12.000
0.025
RAD
-0.025
1620
1690
SEC

Fig. 4 (iv)



u
109.000
M/S
99.000
w
16.000
M/S
6.000
0.200
RAD
0.000
0.050
RAD/S
-0.050
0.250
RAD/S2
-0, zs0
1,500
ws2
0.500
-7.000
M/S2
-12.000
δ_e
0.020
RAD
-0,030
1950
2010
SEC



115.000
M/S
90.000
14.500
M/S
9.500
0.300
RAD
0.200
0.050
RAD/S
-0.050
0.250
RAD/S1
-0.250
1.400
M/S2
0.400
-7.500
M/S2
-12.500
0.020
RAD
b_e
-0.030
2375
2454
SEC

Fig. 4 (vi)

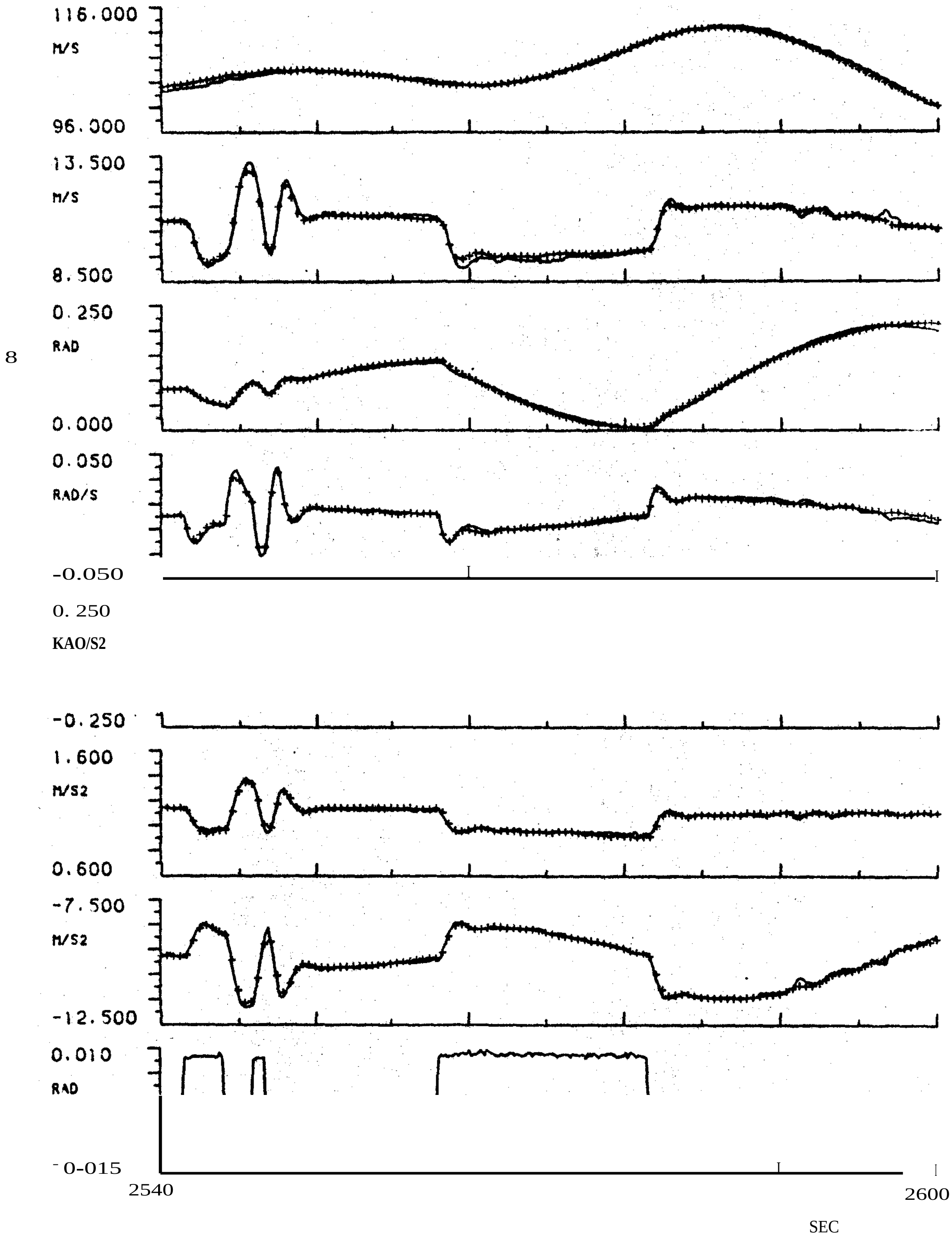
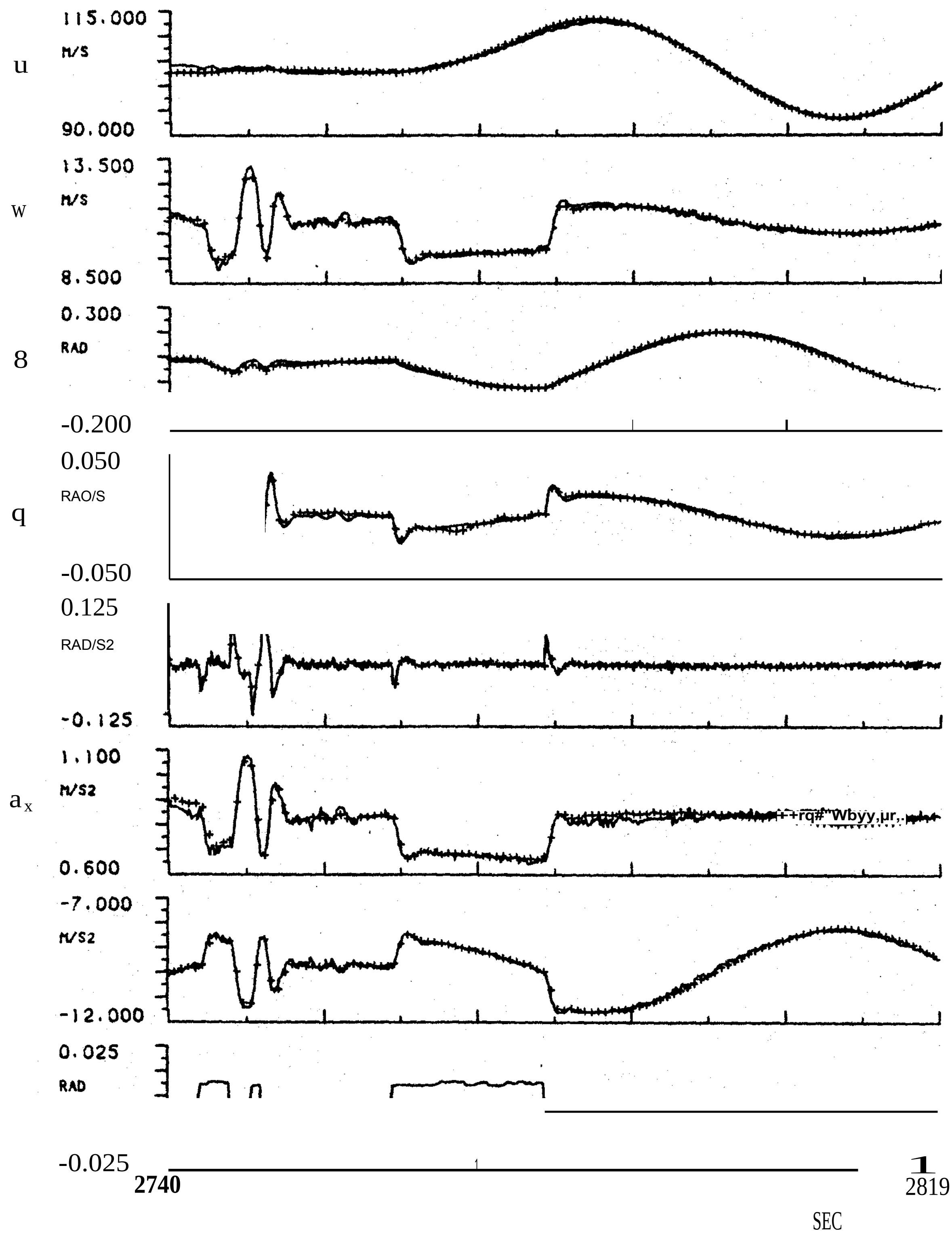
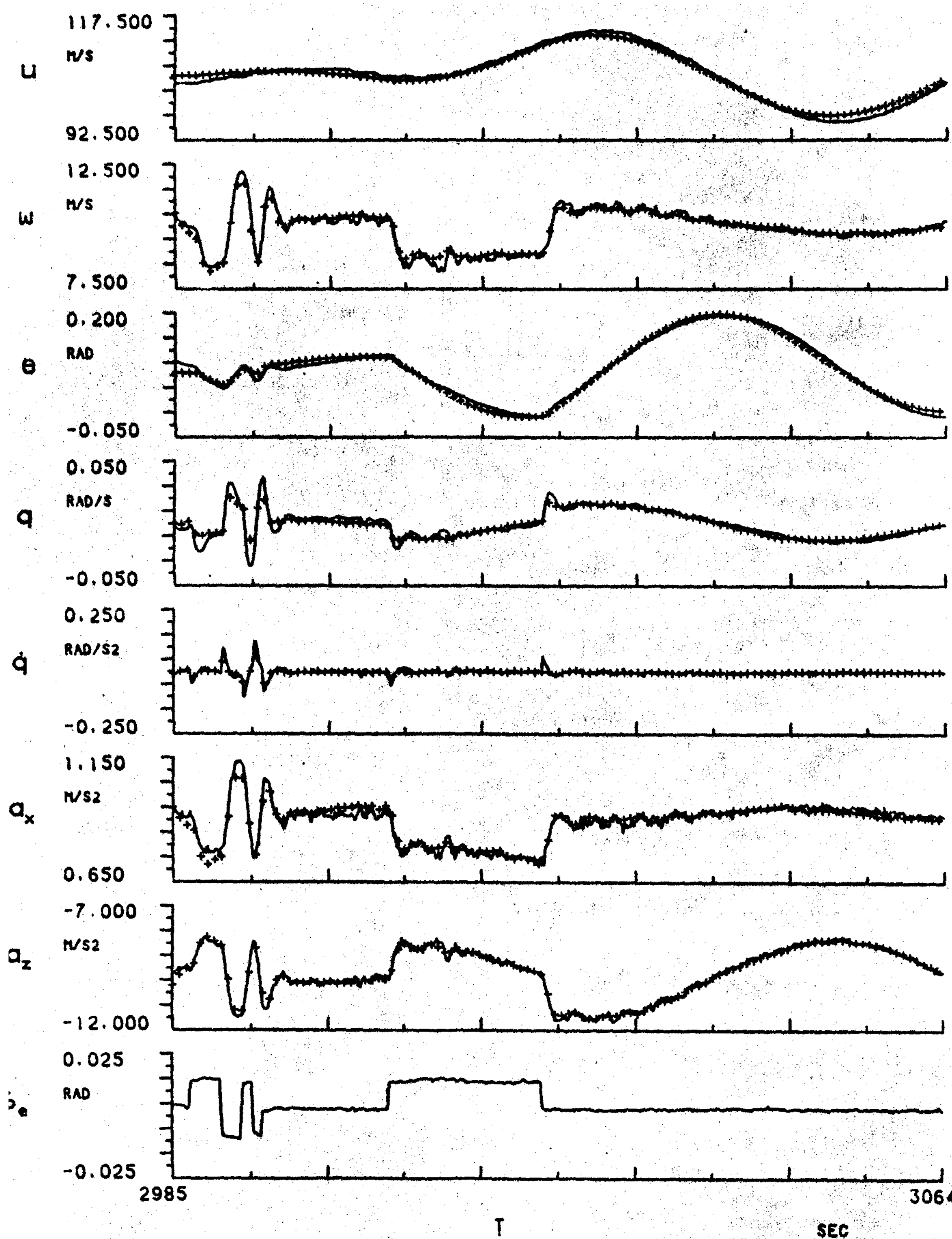

116.000
M/S
96.000
13.500
M/S
8.500
0.250
RAD
8
0.000
0.050
RAD/S
-0.050
0. 250
KAO/S2
-0.250
1.600
M/S2
0.600
-7.500
M/S2
-12.500
0.010
RAD
-0-015
2540
2600
SEC

Fig. 4 (vii)



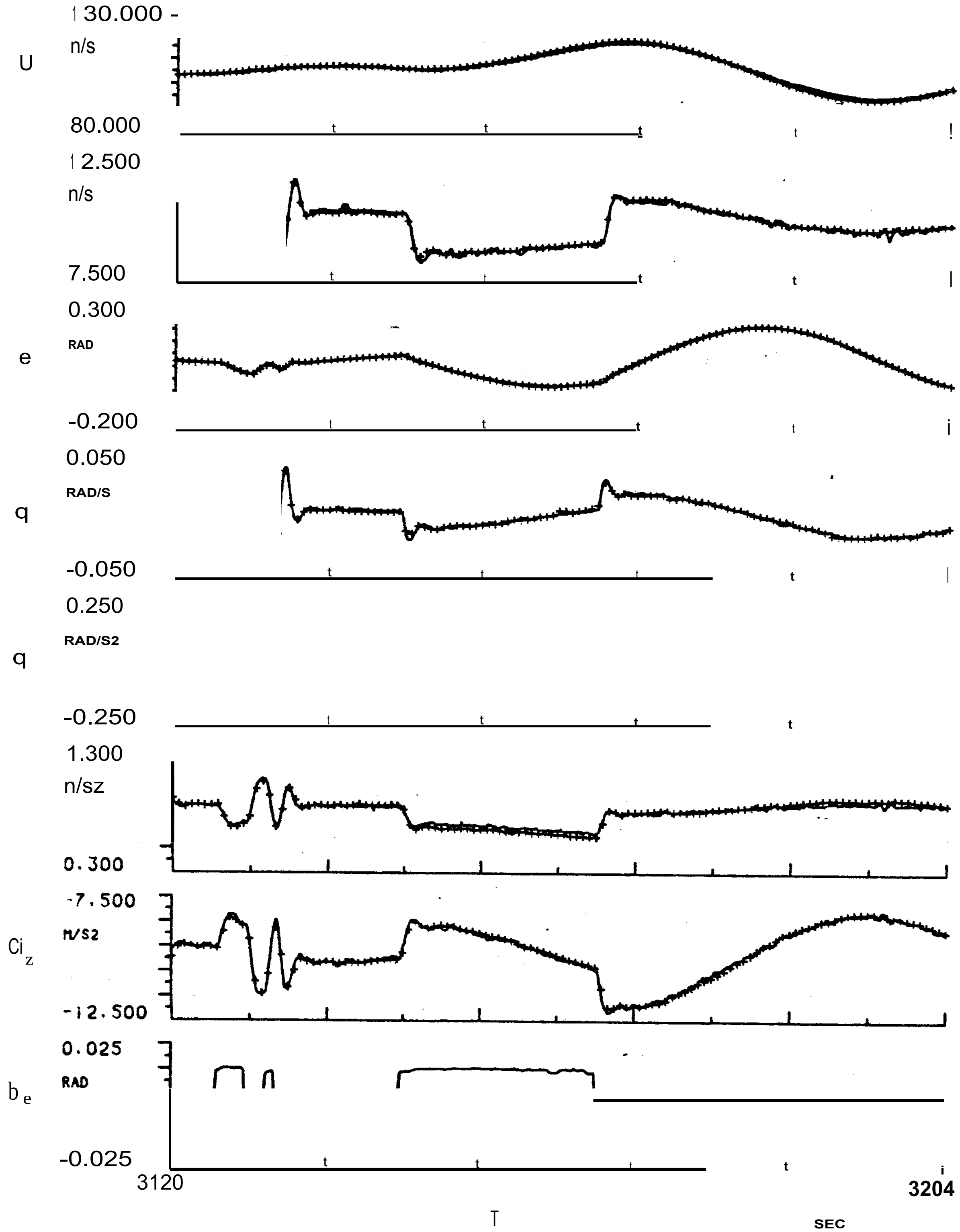
u
115.000
M/S
90.000
w
13.500
M/S
8.500
8
0.300
RAD
-0.200
q
0.050
RAO/S
-0.050
0.125
RAD/S2
-0.125
a_x
1.100
M/S2
0.600
-7.000
M/S2
-12.000
0.025
RAD
-0.025
2740
2819
SEC

Fig. 4 (viii)



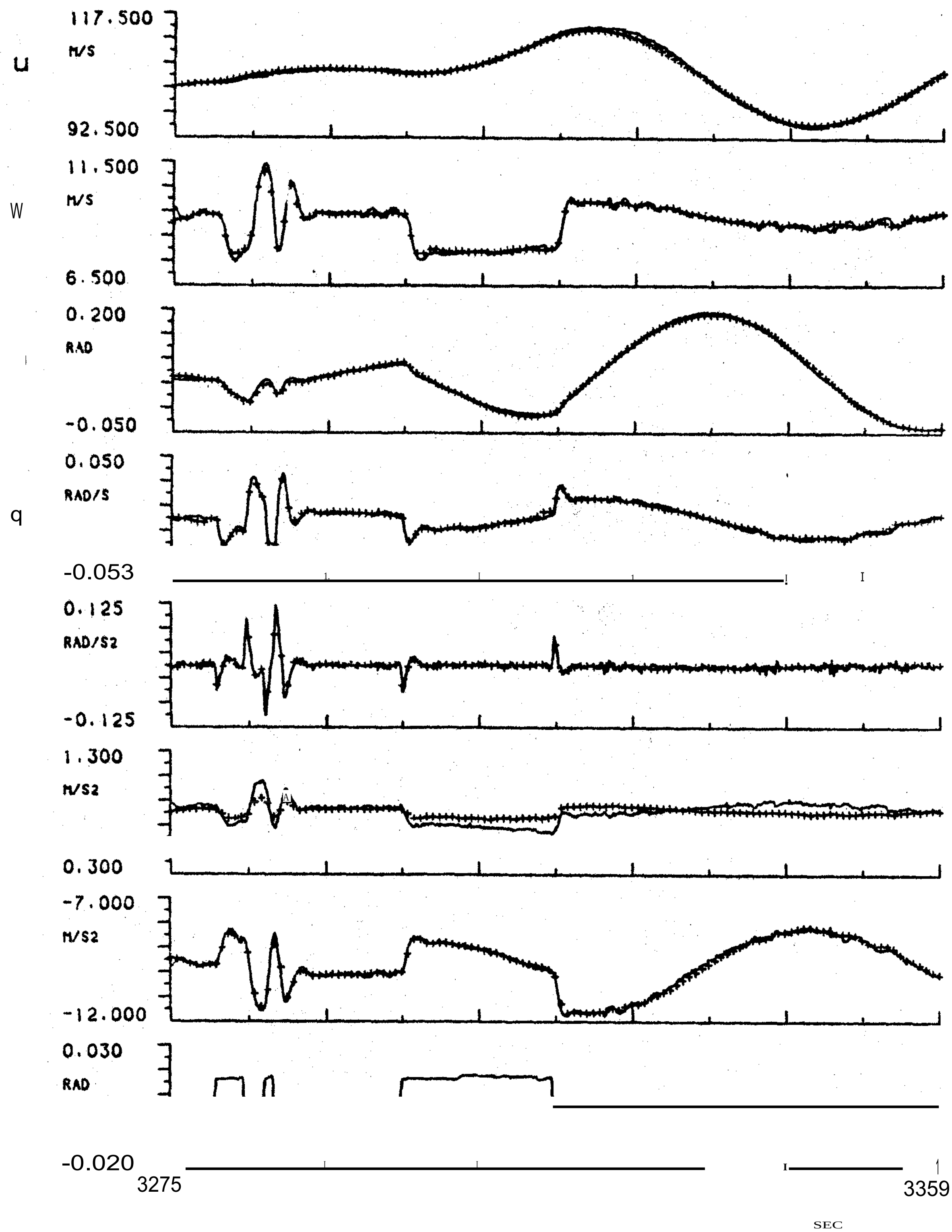
117.500
M/S
u
92.500
12.500
M/S
w
7.500
0.200
RAD
θ
-0.050
0.050
RAD/S
q
-0.050
0.250
RAD/S2
-0.250
1.150
M/S2
a_x
0.650
-7.000
M/S2
a_z
-12.000
0.025
RAD
-0.025
2985
3064
T
SEC

Fig. 4 (ix)



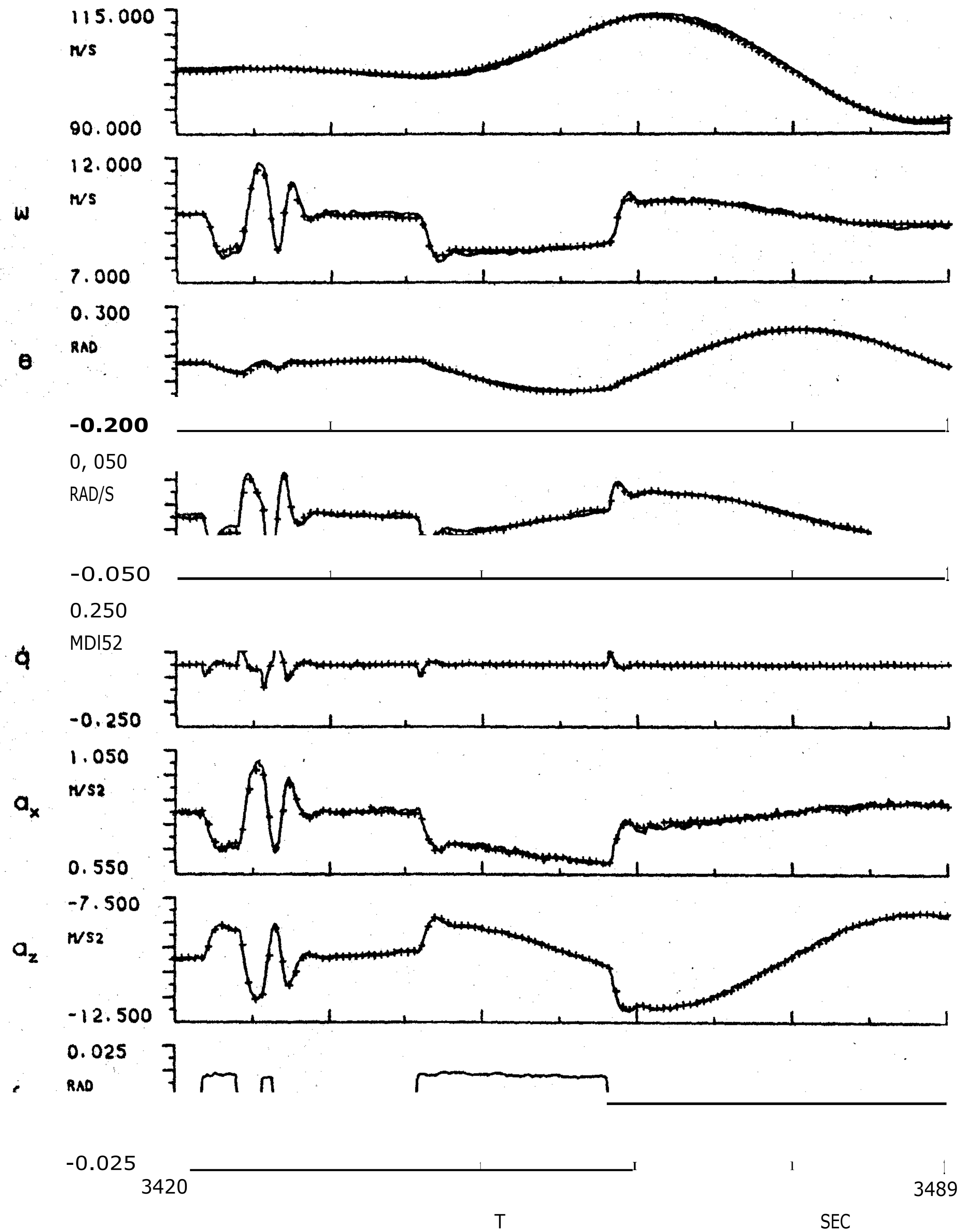
130.000
n/s
U
80.000
12.500
n/s
7.500
0.300
RAD
e
-0.200
0.050
RAD/S
q
-0.050
0.250
RAD/S2
q
-0.250
1.300
n/sz
0.300
-7.500
M/S2
Ci z
-12.500
0.025
RAD
b e
-0.025
3120
3204
T
SEC

Fig. 4 (x)



u
117.500
M/S
92.500
W
11.500
M/S
6.500
0.200
RAD
-0.050
q
0.050
RAD/S
-0.053
0.125
RAD/S2
-0.125
1.300
M/S2
0.300
-7.000
M/S2
-12.000
0.030
RAD
-0.020
3275
3359
SEC

Fig. 4 (xi)



115.000
M/S
90.000
12.000
M/S
w
7.000
0.300
RAD
θ
-0.200
0, 050
RAD/S
-0.050
0.250
MDI52
q̇
-0.250
1.050
M/S2
a_x
0.550
-7.500
M/S2
a_z
-12.500
0.025
RAD
-0.025
3420
3489
T
SEC

Fig. 4 (xii)

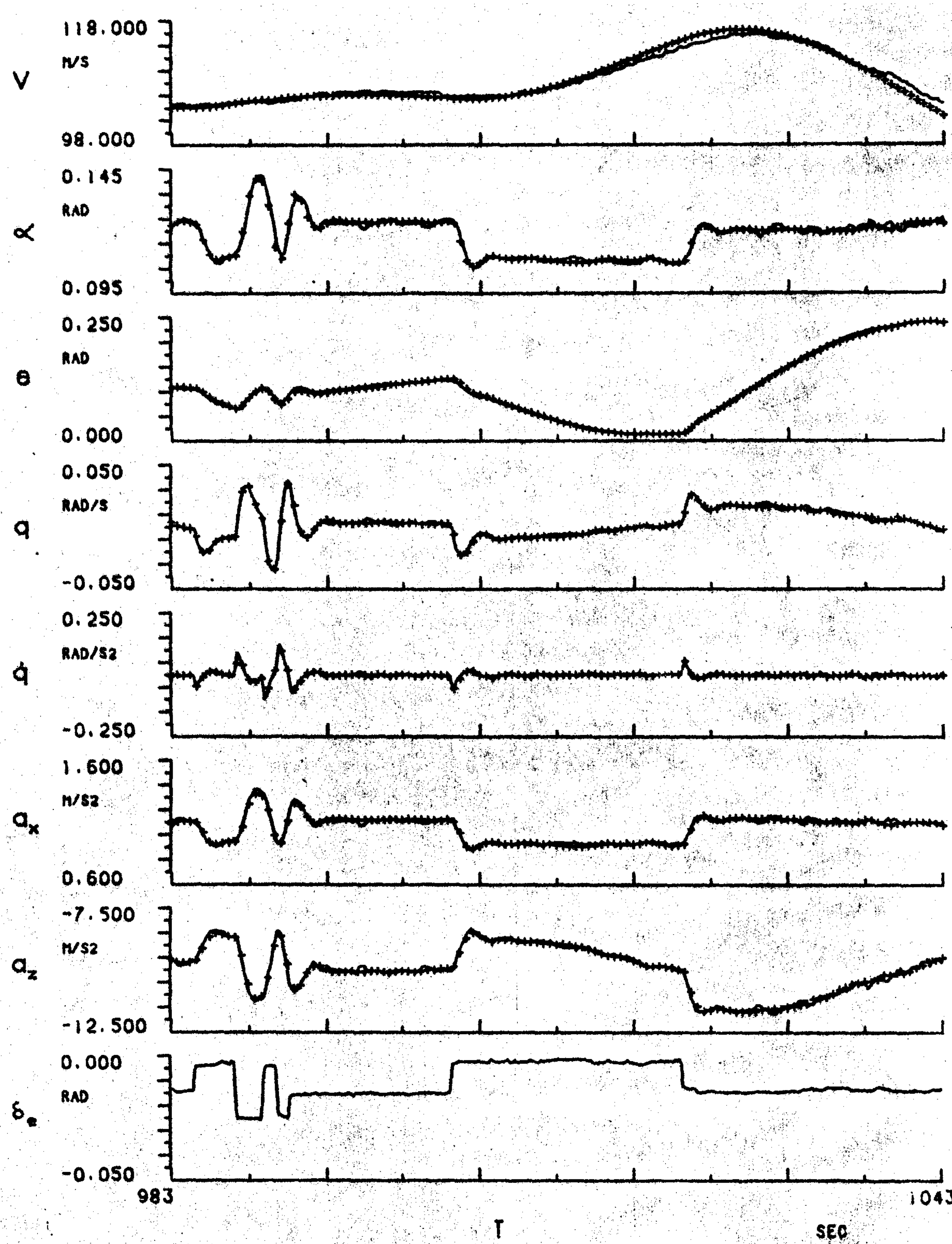


Fig. 5 (i)

Fig. 5 : *Curve Fits from Parameter* Estimation Using Non-Dimensional *Coefficients, and Nonlinear model* (——— Jeasured, ++++. Estimated)

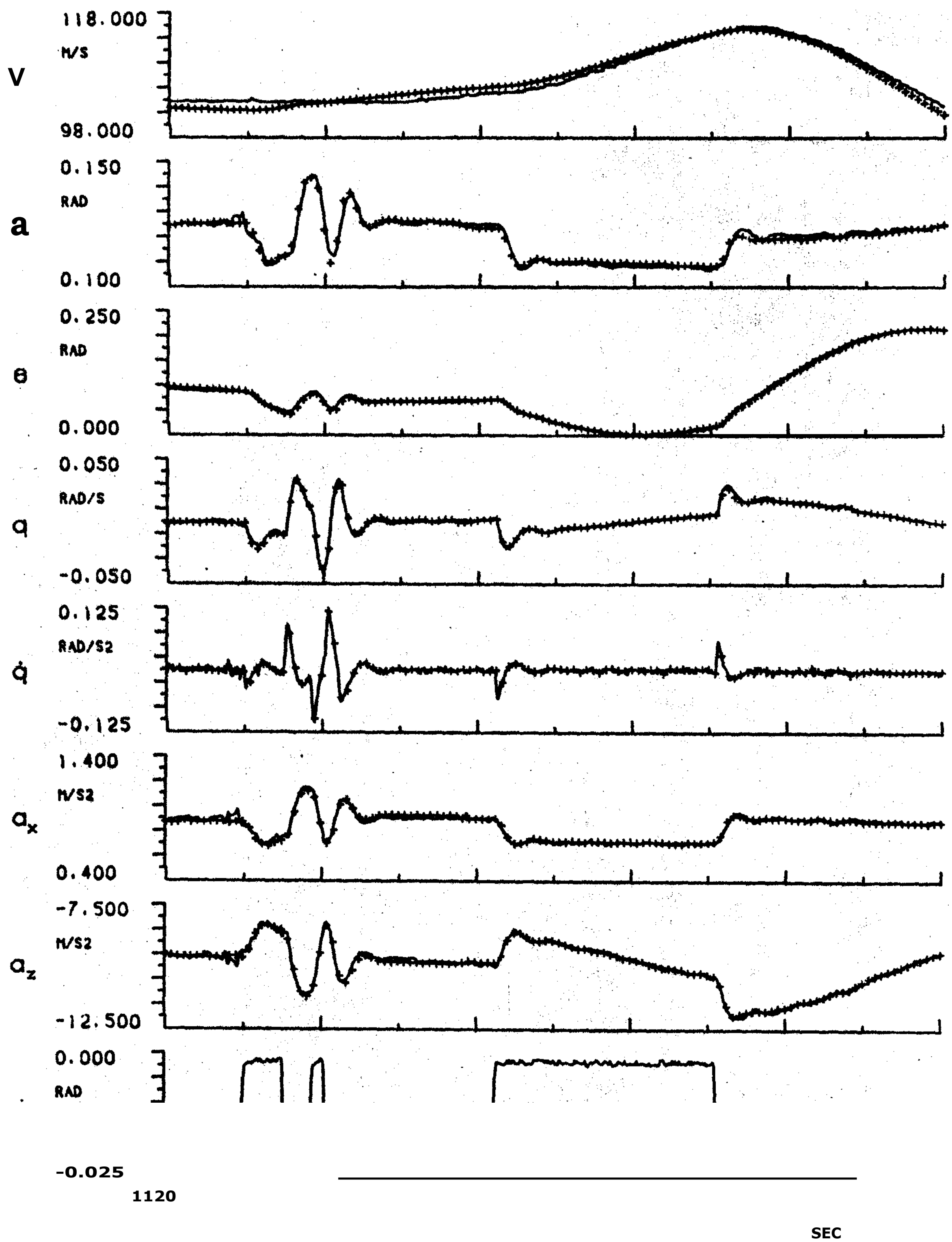

118.000
M/S
V
98.000
0.150
RAD
a
0.100
0.250
RAD
θ
0.000
0.050
RAD/S
q
-0.050
0.125
RAD/S2
q̇
-0.125
1.400
M/S2
a_x
0.400
-7.500
M/S2
a_z
-12.500
0.000
RAD
-0.025
1120
SEC

Fig. 5 (ii)

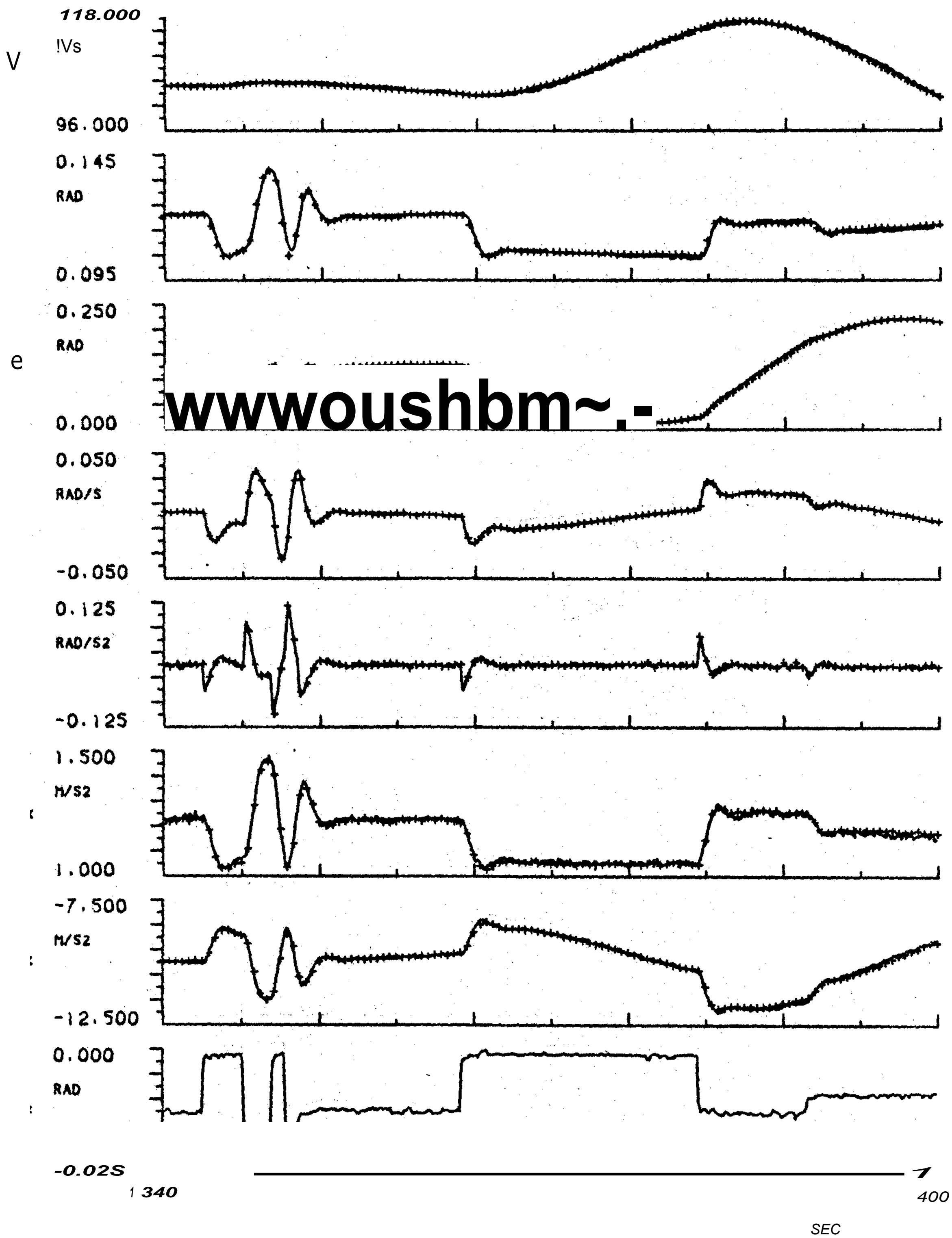
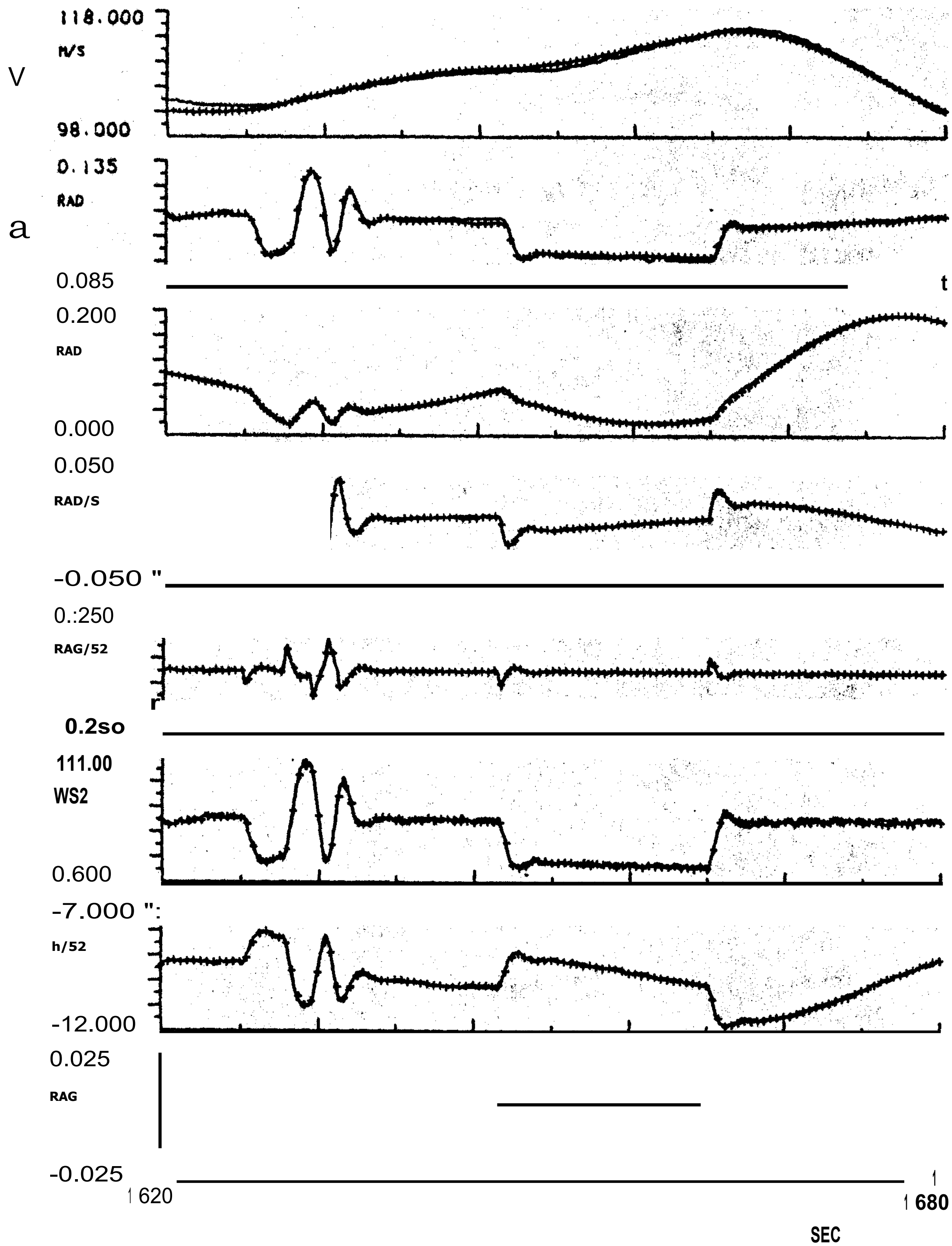
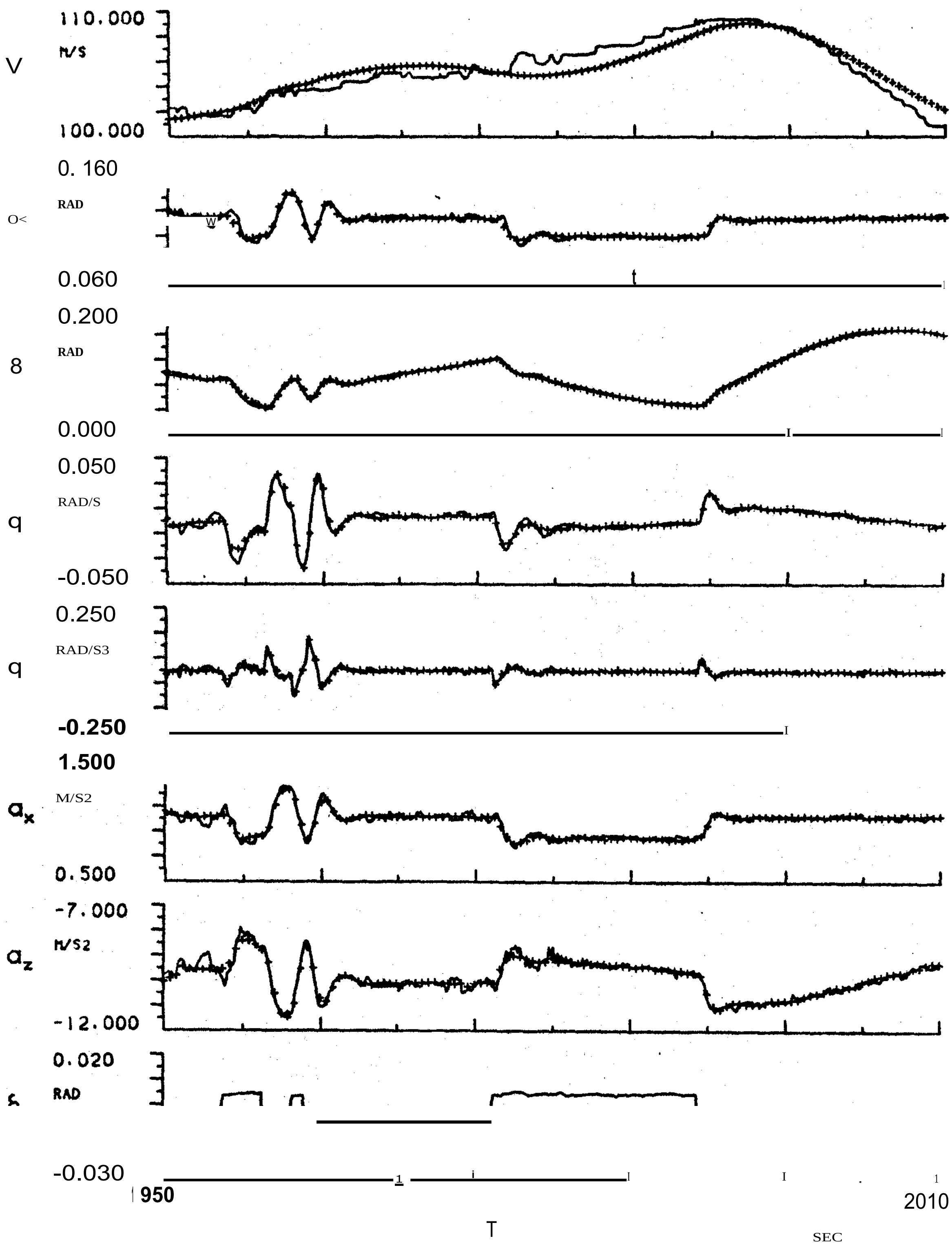

118.000
!Vs
V
96.000
0.145
RAD
0.095
0.250
RAD
e
wwwoushbm~.-
0.000
0.050
RAD/S
-0.050
0.125
RAD/S2
-0.125
1.500
M/S2
1.000
-7.500
M/S2
-12.500
0.000
RAD
-0.02S
1 340
400
SEC

Fig. 5 (iii)



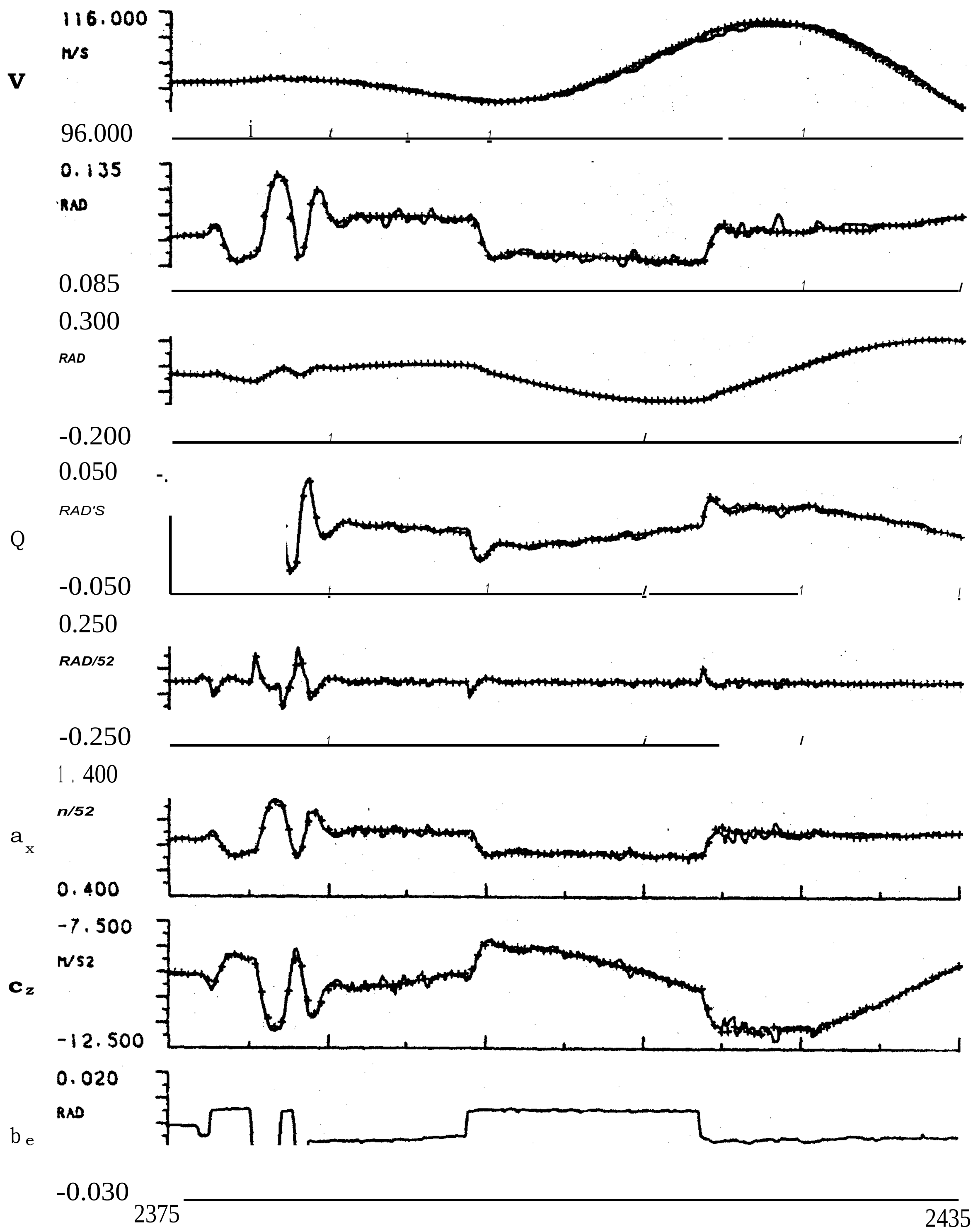
118.000
M/S
V
98.000
0.135
RAD
a
0.085
t
0.200
RAD
0.000
0.050
RAD/S
-0.050 "
0.:250
RAG/52
0.2so
111.00
WS2
0.600
-7.000 ":
h/52
-12.000
0.025
RAG
-0.025
1620
1680
SEC

Fig. 5 (iv)



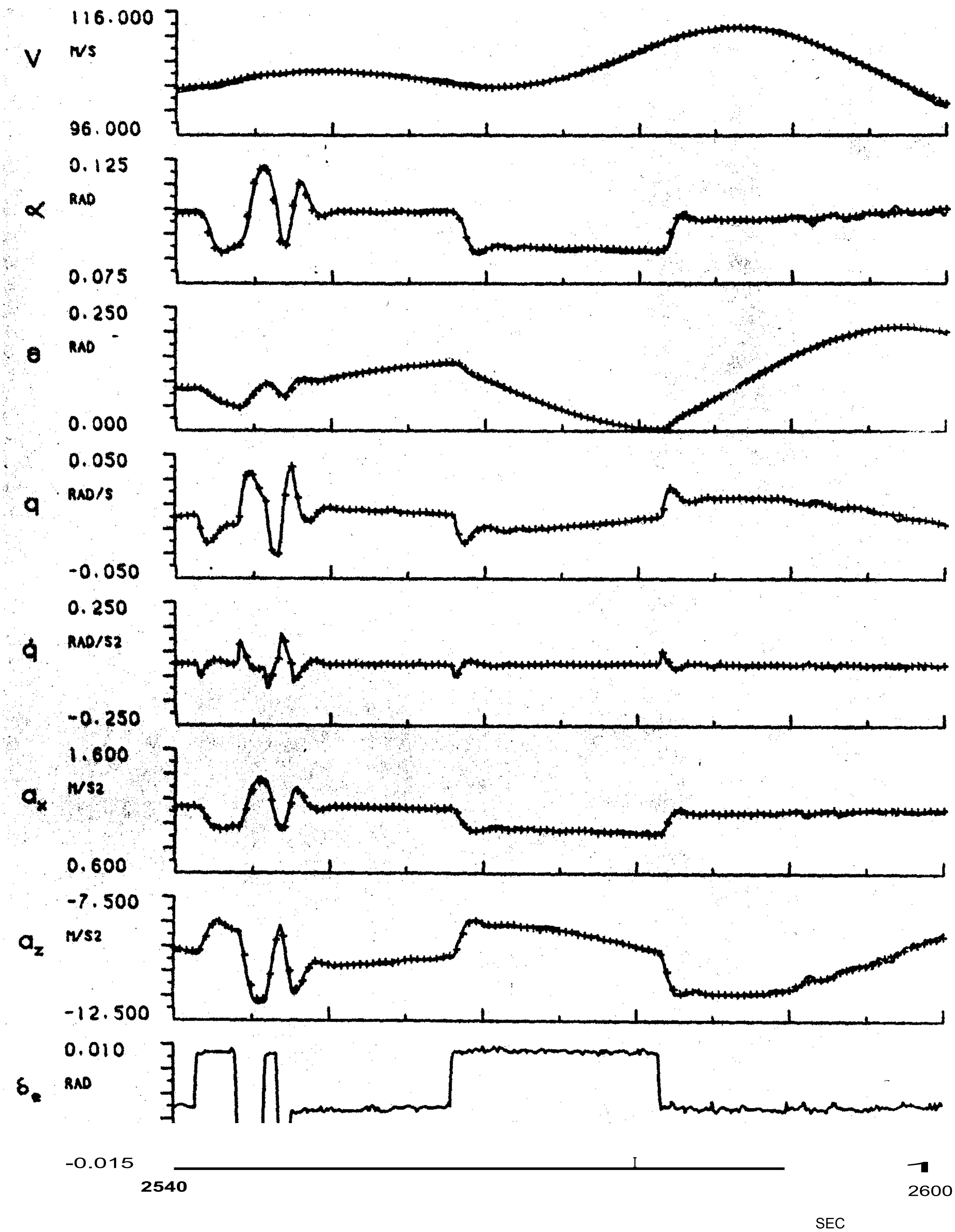
110.000
M/S
V
100.000
0.160
RAD
0.060
0.200
RAD
0.000
0.050
RAD/S
q
-0.050
0.250
RAD/S3
q
-0.250
1.500
M/S2
a_x
0.500
-7.000
M/S2
a_z
-12.000
0.020
RAD
-0.030
950
2010
T
SEC

Fig. 5 **v**



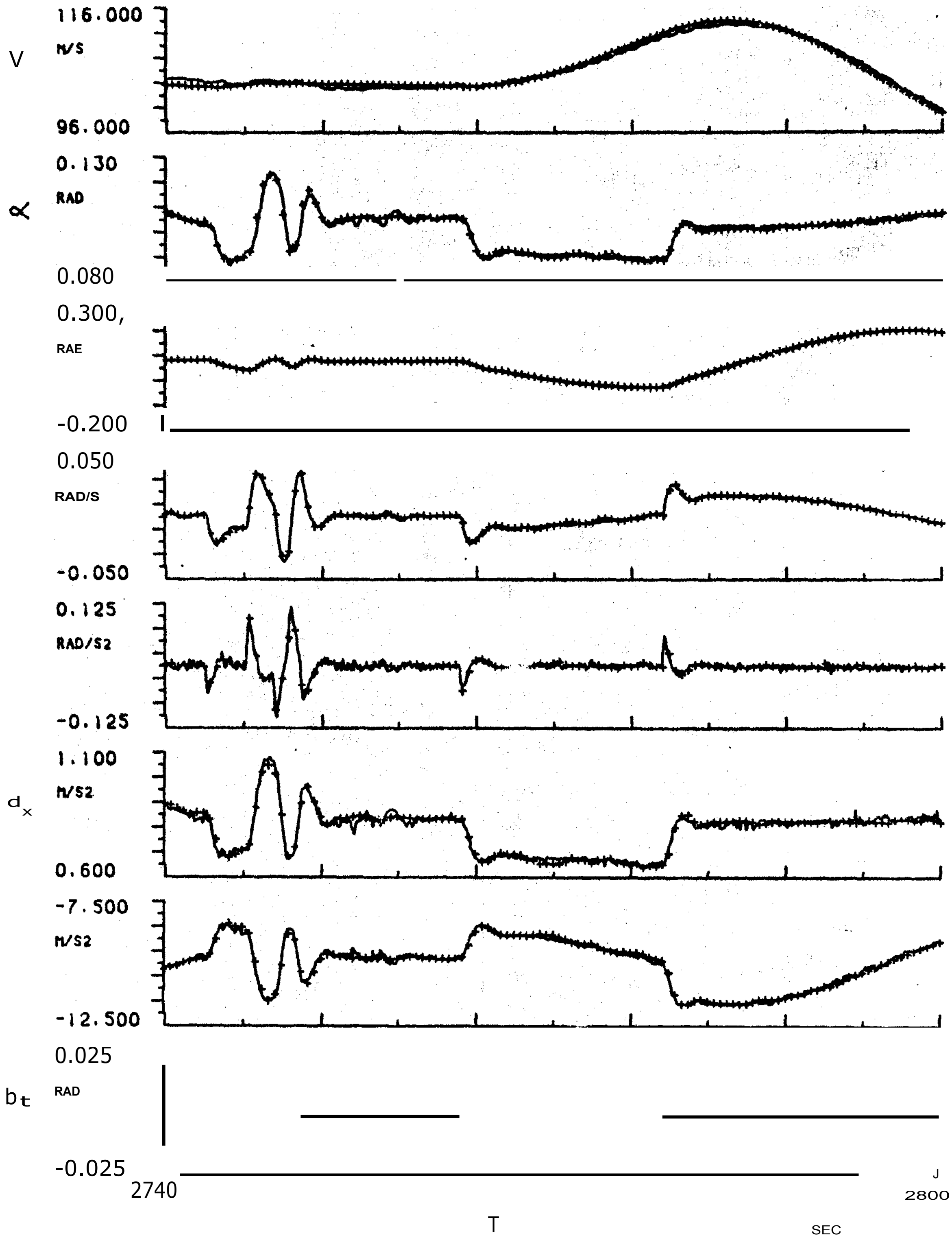
116.000
M/S
V
96.000
0.135
RAD
0.085
0.300
RAD
-0.200
0.050
RAD'S
Q
-0.050
0.250
RAD/52
-0.250
1.400
n/52
a x
0.400
-7.500
M/S2
c z
-12.500
0.020
RAD
b e
-0.030
2375
2435
SEC

Fig. 5 (vi)



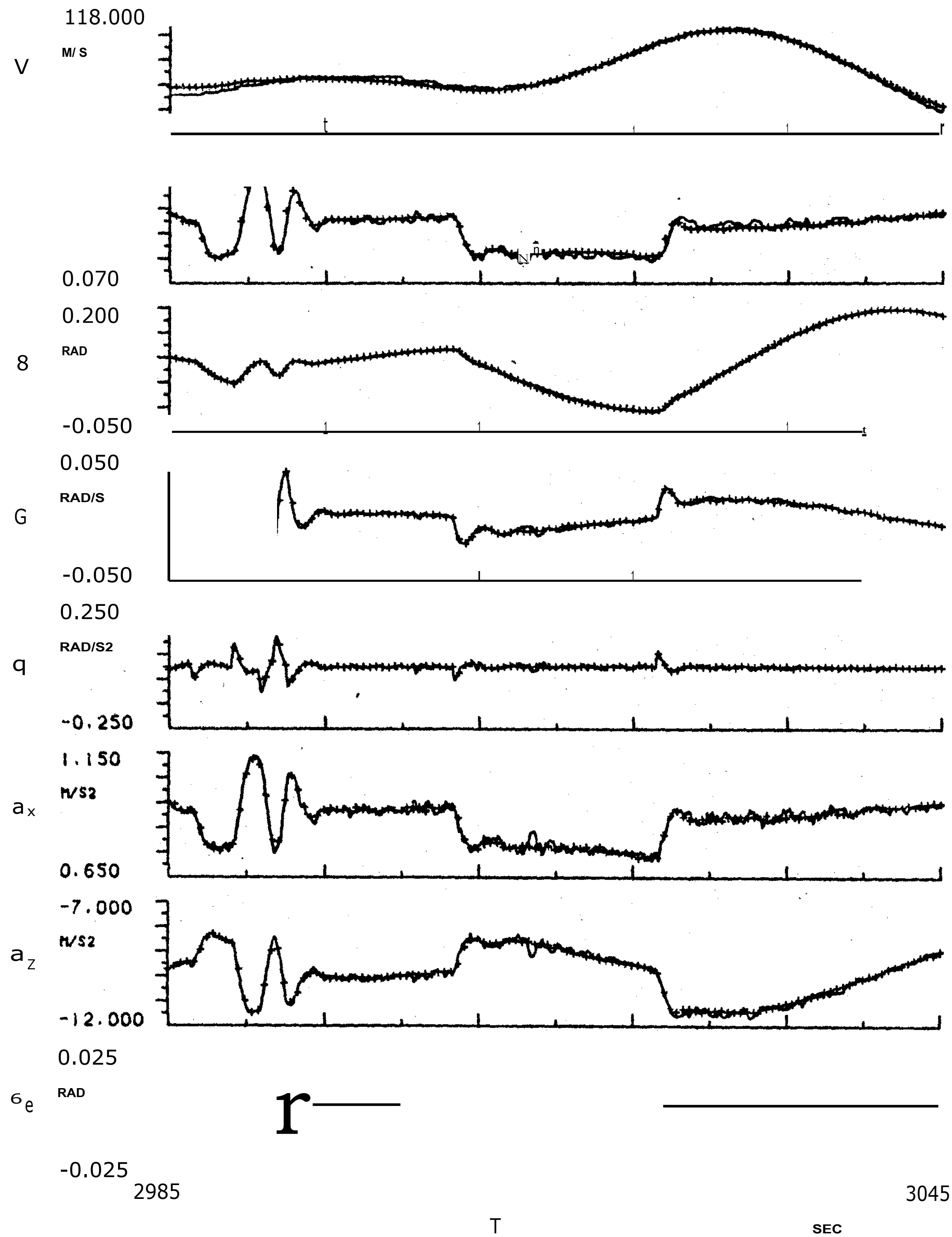
V
116.000
M/S
96.000
α
0.125
RAD
0.075
θ
0.250
RAD
0.000
q
0.050
RAD/S
-0.050
q̇
0.250
RAD/S2
-0.250
a_x
1.600
M/S2
0.600
a_z
-7.500
M/S2
-12.500
δ_e
0.010
RAD
-0.015
2540
2600
SEC

Fig. 5 (vii)



V
116.000
M/S
96.000
α
0.130
RAD
0.080
0.300,
RAE
-0.200
0.050
RAD/S
-0.050
0.125
RAD/S2
-0.125
d_x
1.100
M/S2
0.600
-7.500
M/S2
-12.500
b_t
0.025
RAD
-0.025
2740
2800
T
SEC

Fig, 5 **(viii)**



118.000
V
M/S
0.070
0.200
8
RAD
-0.050
0.050
G
RAD/S
-0.050
0.250
q
RAD/S2
-0.250
1.150
a_x
M/S2
0.650
-7.000
a_z
M/S2
-12.000
0.025
δ_e
RAD
-0.025
2985
3045
T
SEC

Fig. 5 (ix)

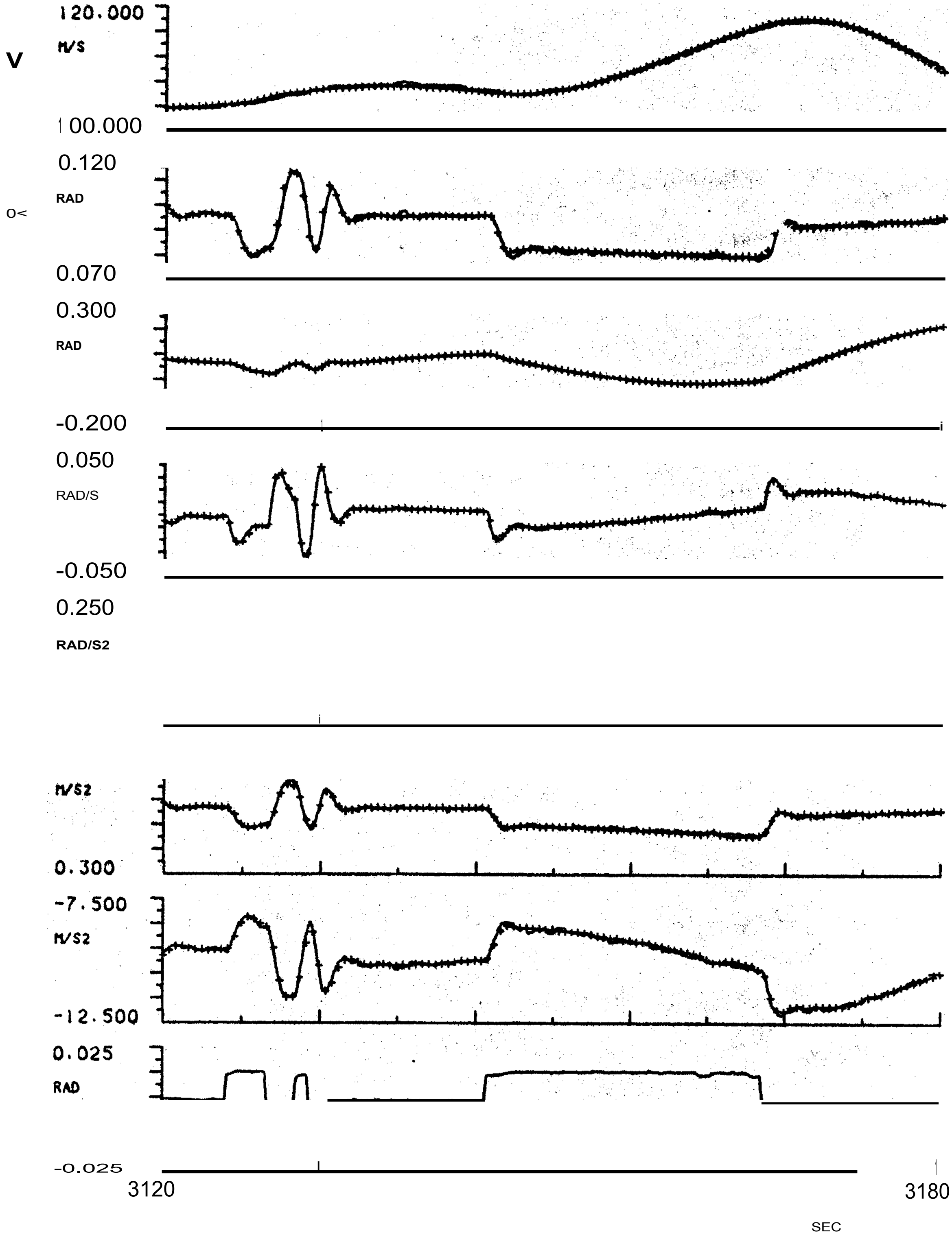
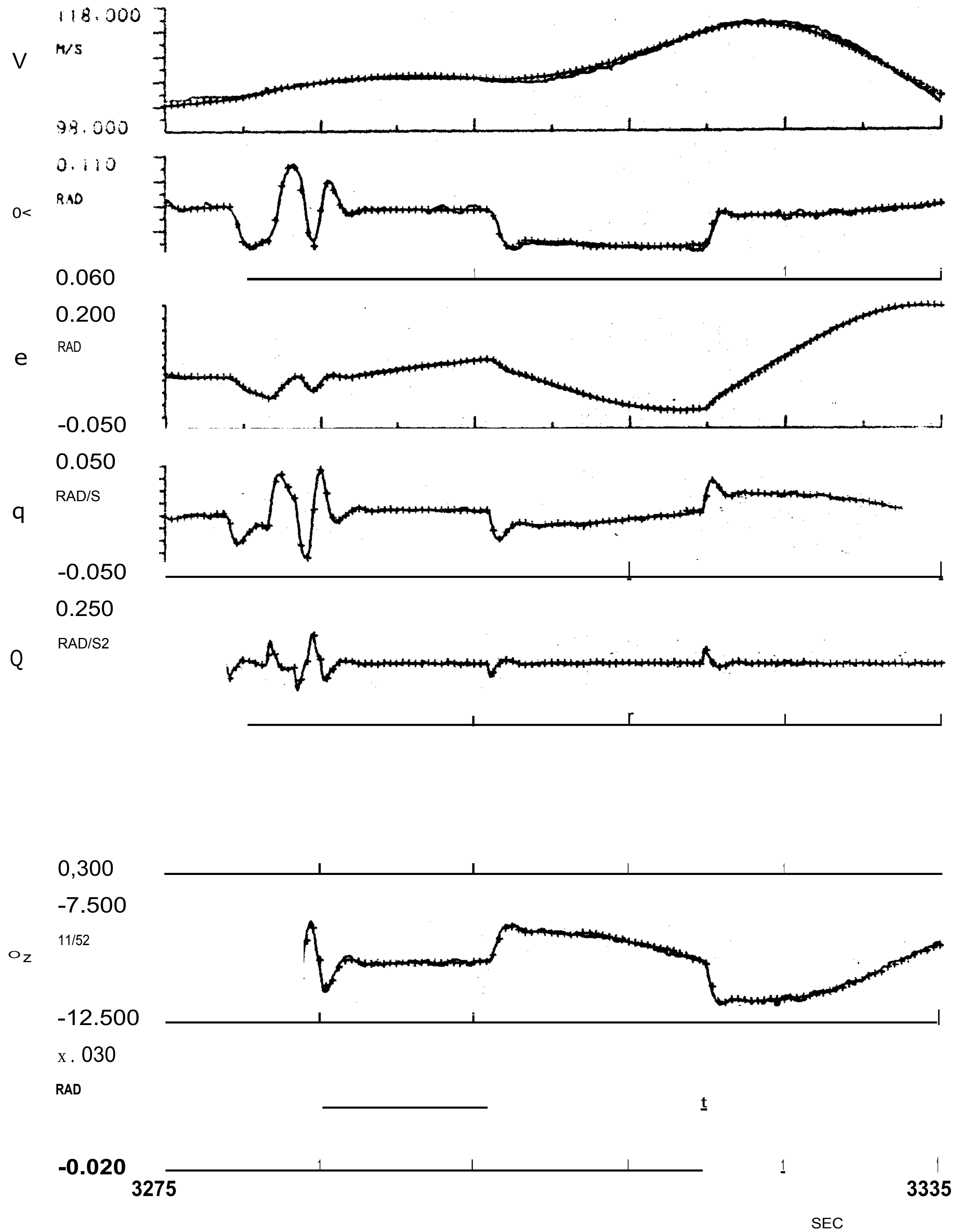
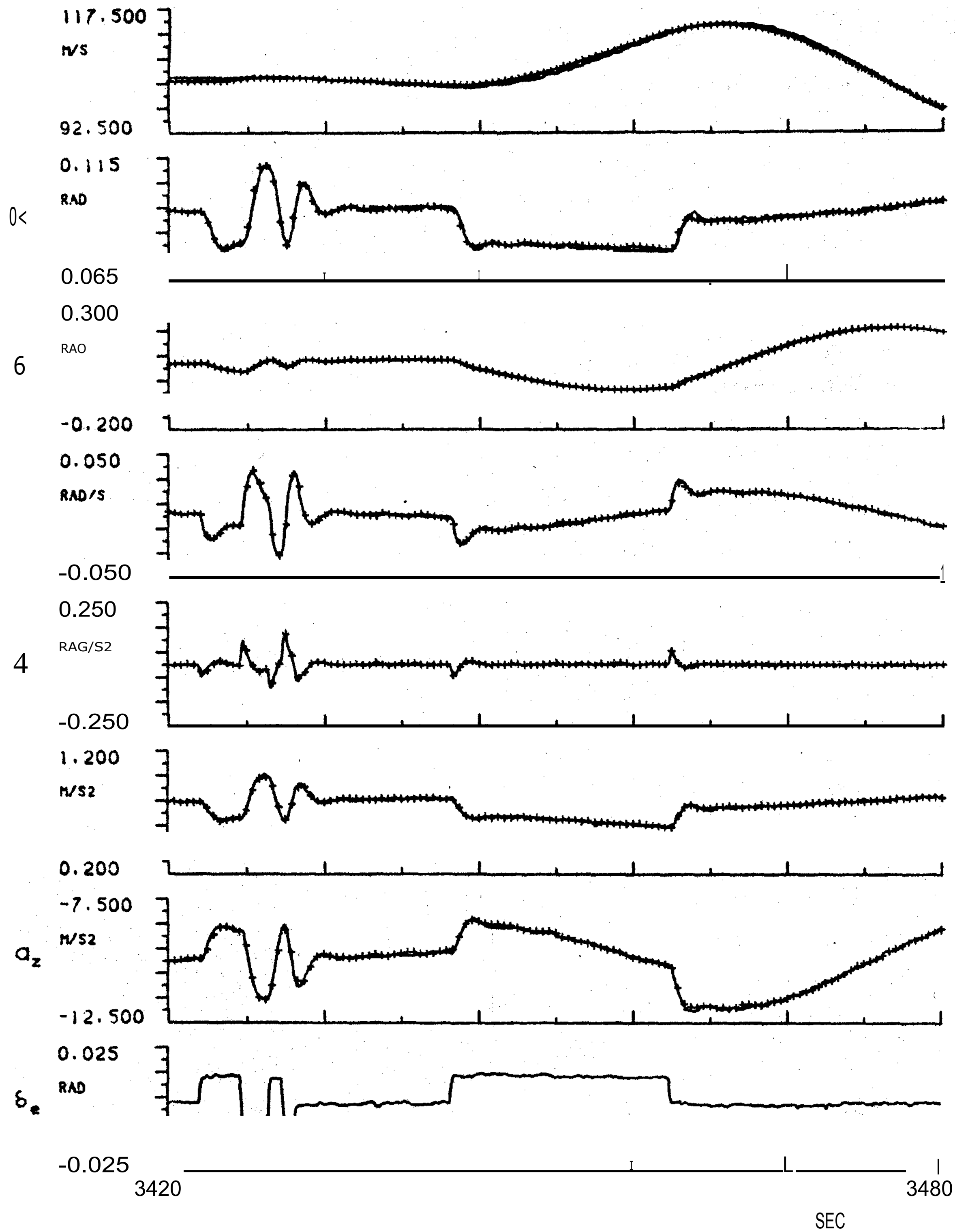

V
120.000
M/S
100.000
0.120
RAD
0<
0.070
0.300
RAD
-0.200
0.050
RAD/S
-0.050
0.250
RAD/S2
M/S2
0.300
-7.500
M/S2
-12.500
0.025
RAD
-0.025
3120
3180
SEC

Fig. 5 (x) ,:



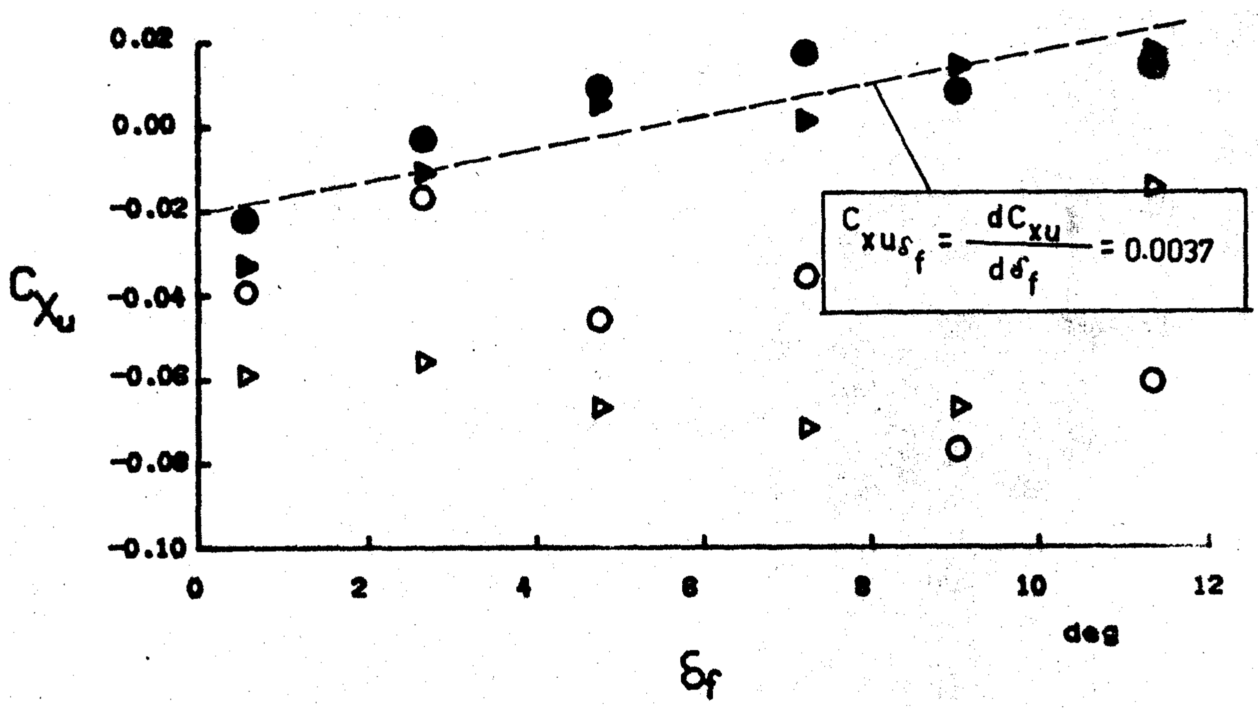
V
118.000
M/S
98.000
0.110
RAD
0<
0.060
0.200
RAD
e
-0.050
0.050
RAD/S
q
-0.050
0.250
RAD/S2
Q
0,300
-7.500
11/52
oz
-12.500
x.030
RAD
t
-0.020
3275
3335
SEC

Fig. 5 (xi)



117.500
M/S
92.500
0.115
RAD
0<
0.065
0.300
RAO
6
-0.200
0.050
RAD/S
-0.050
0.250
RAG/S2
4
-0.250
1.200
M/S2
0.200
-7.500
M/S2
a_z
-12.500
0.025
RAD
δ_e
-0.025
3420
3480
SEC

Fig. 5 (xii)



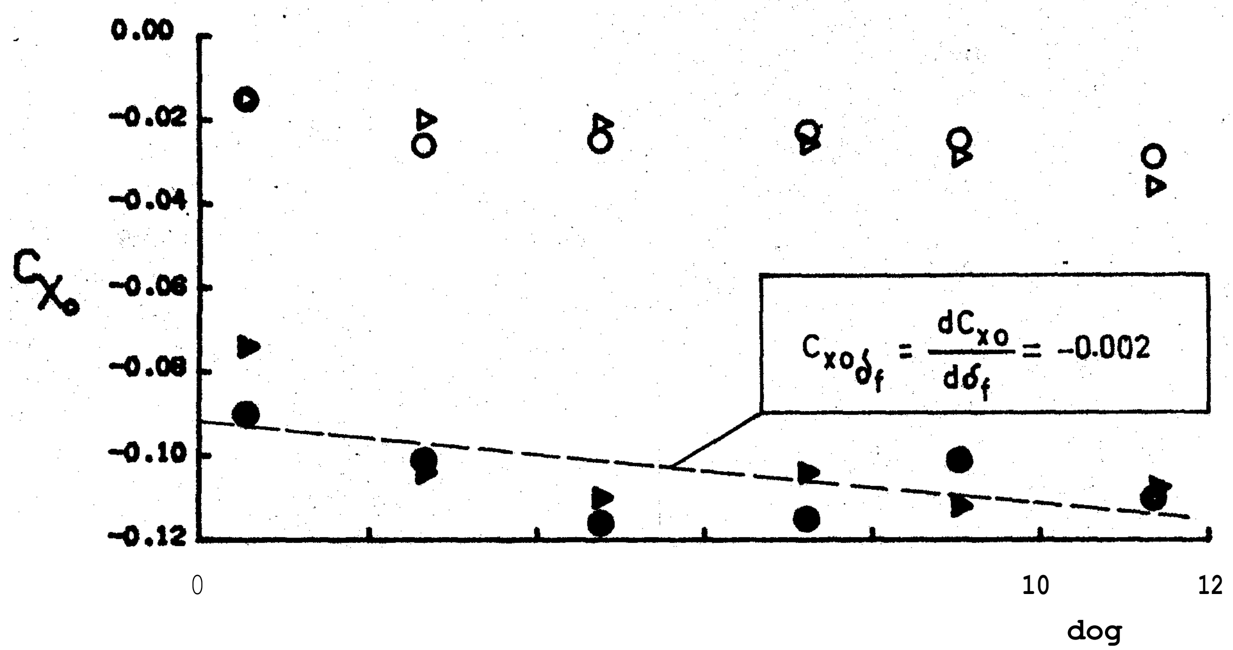


Fig. 6 (i)

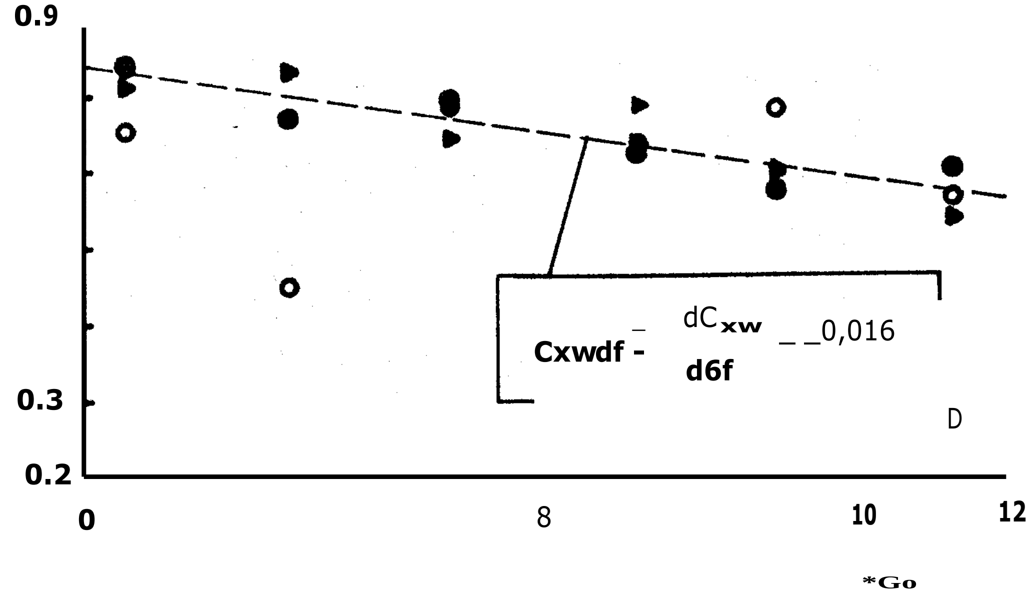
Fig. 6 Non-Dimensional Longitudinal, normal Force and Pitching Moment Coefficients C., Cx, Cm as Functions of Flap Position

bl First Run Non-Dimensional Coefficients Estimated Directly Using Nonlinear Model

Θ Repeat Run (Table 7)

D First Run Non-Dimensional Coefficients Converted From Estimated'Dimensional Derivatives

0 Repeat Run Using Linear Model (Table `6)


0.9
0.3
0.2
Cxwdf - dCxw / d6f _ _0,016
D
0
8
10
12
*Go

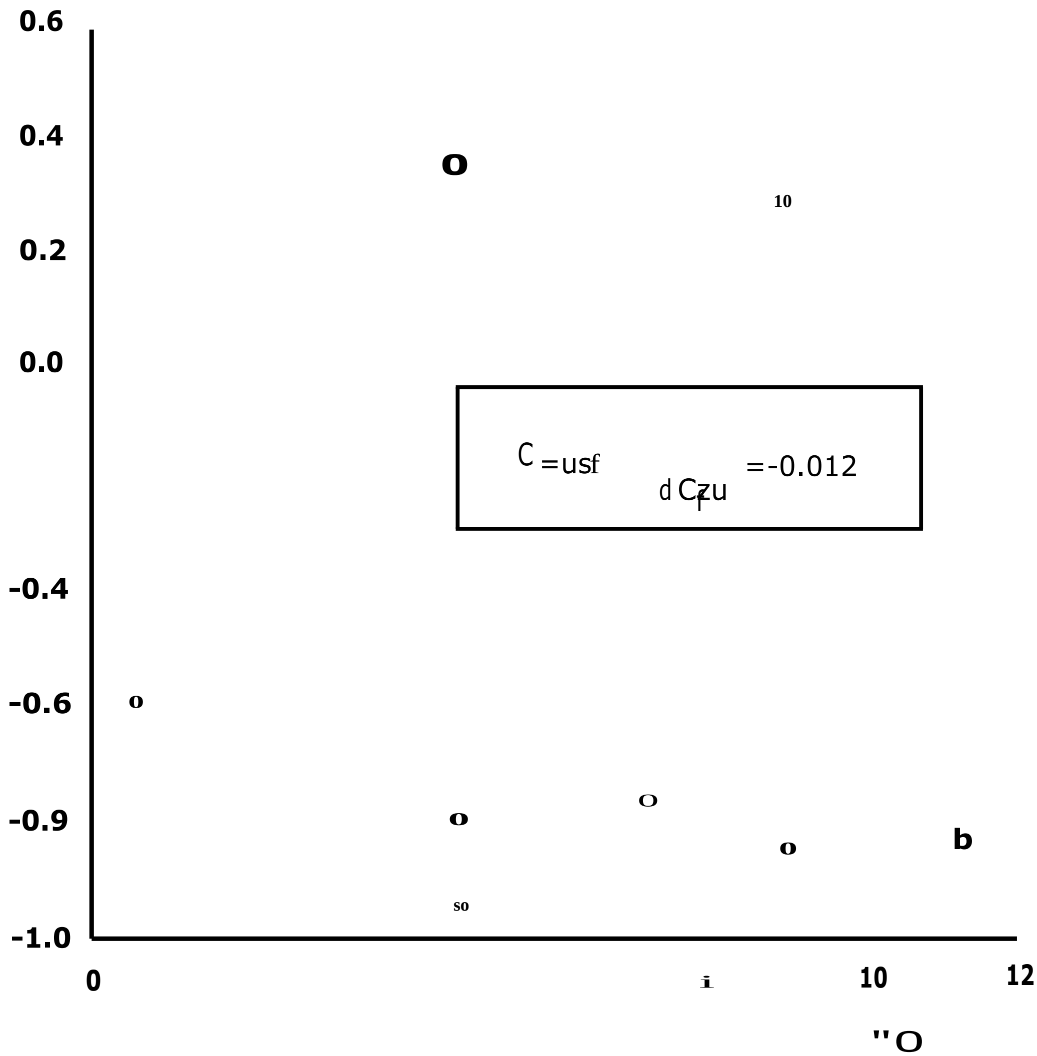
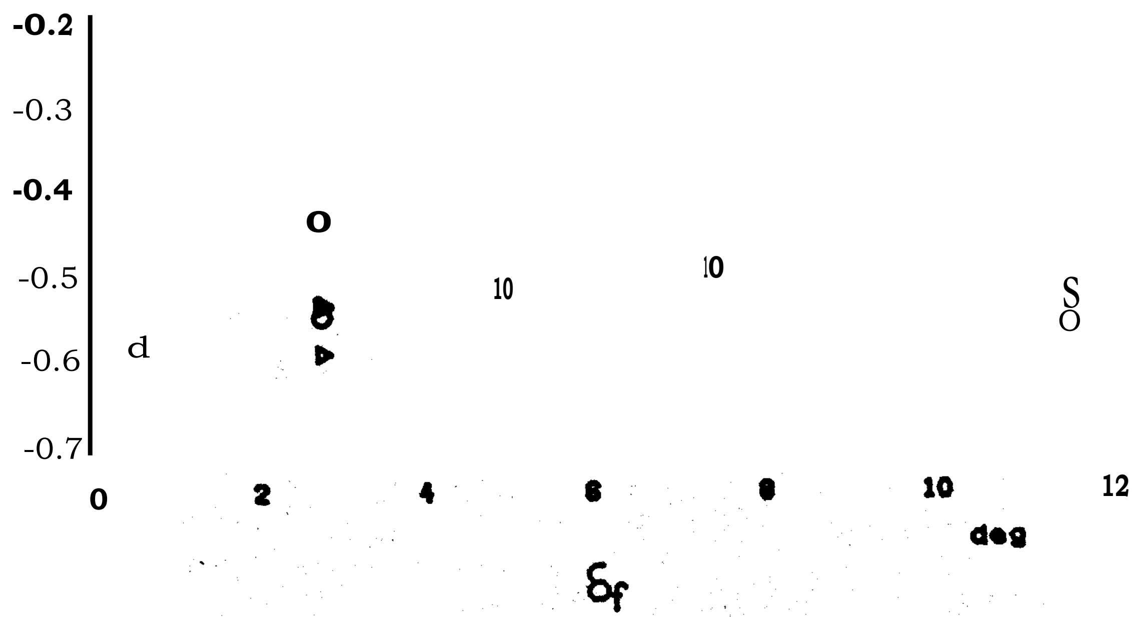

0.6
0.4
0.2
0.0
-0.4
-0.6
-0.9
-1.0
C =usf dCzu / f =-0.012
b
0
10
12
"O

Fig. 6 (ii)


-0.2
-0.3
-0.4
-0.5
-0.6
-0.7
d
0
2
4
6
8
10
12
deg
δ_f

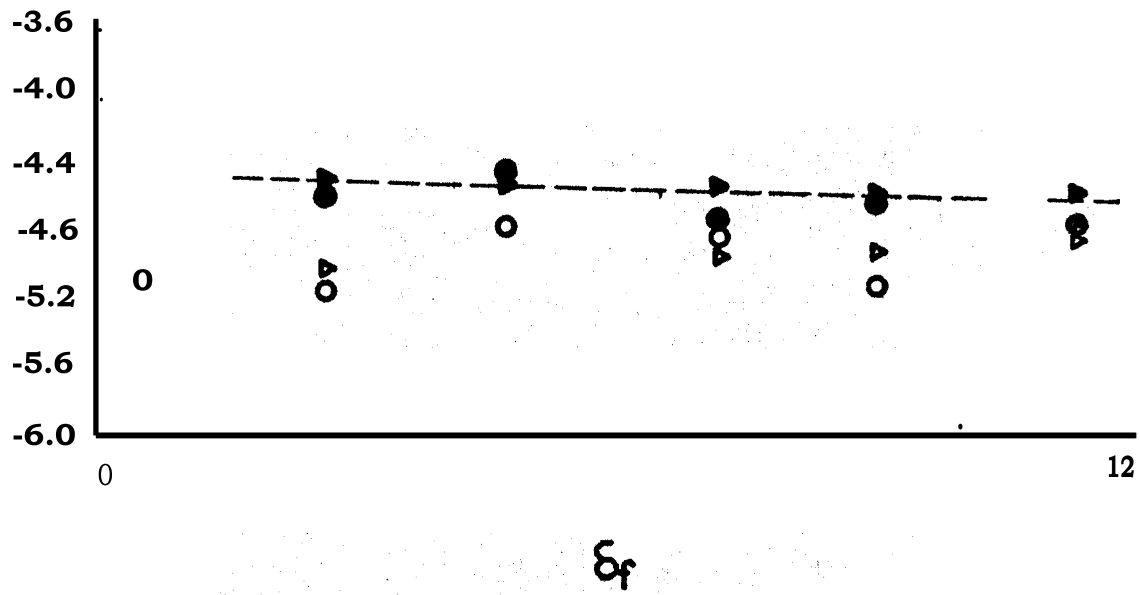
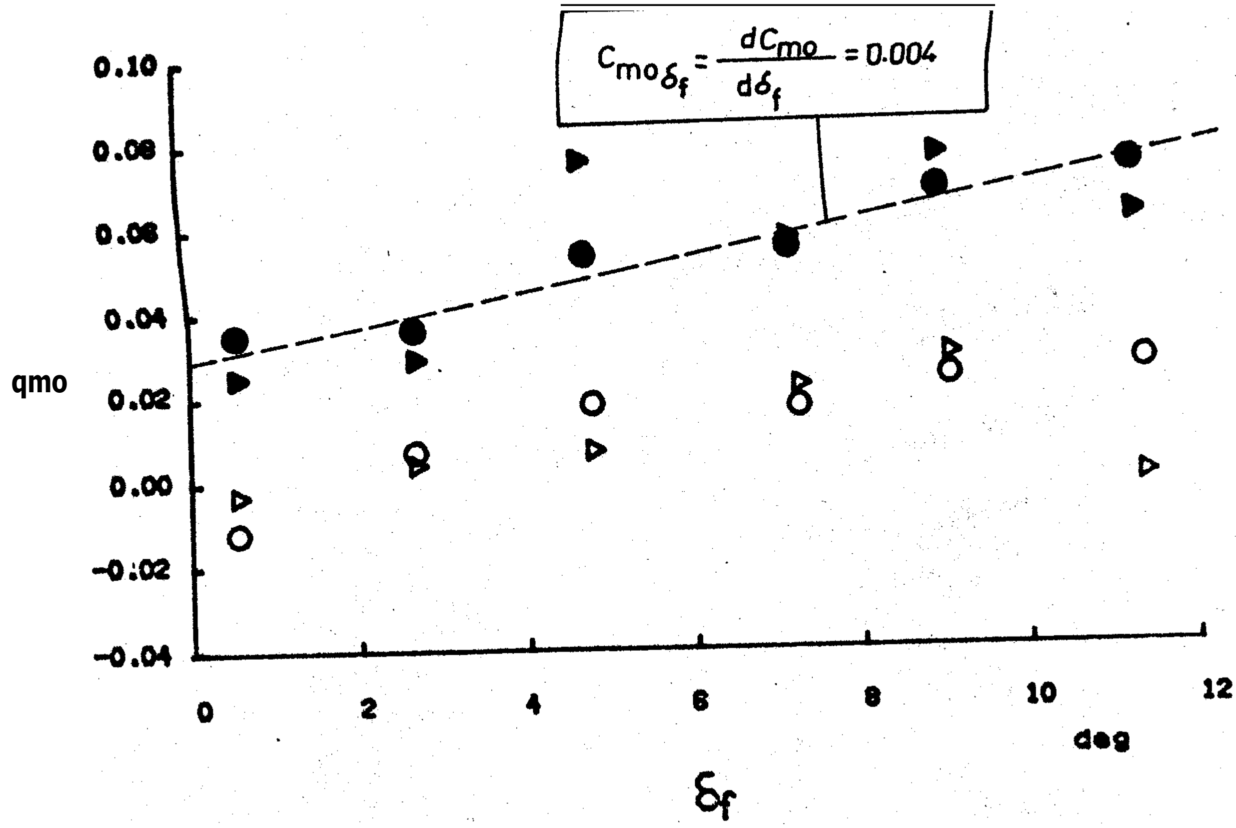
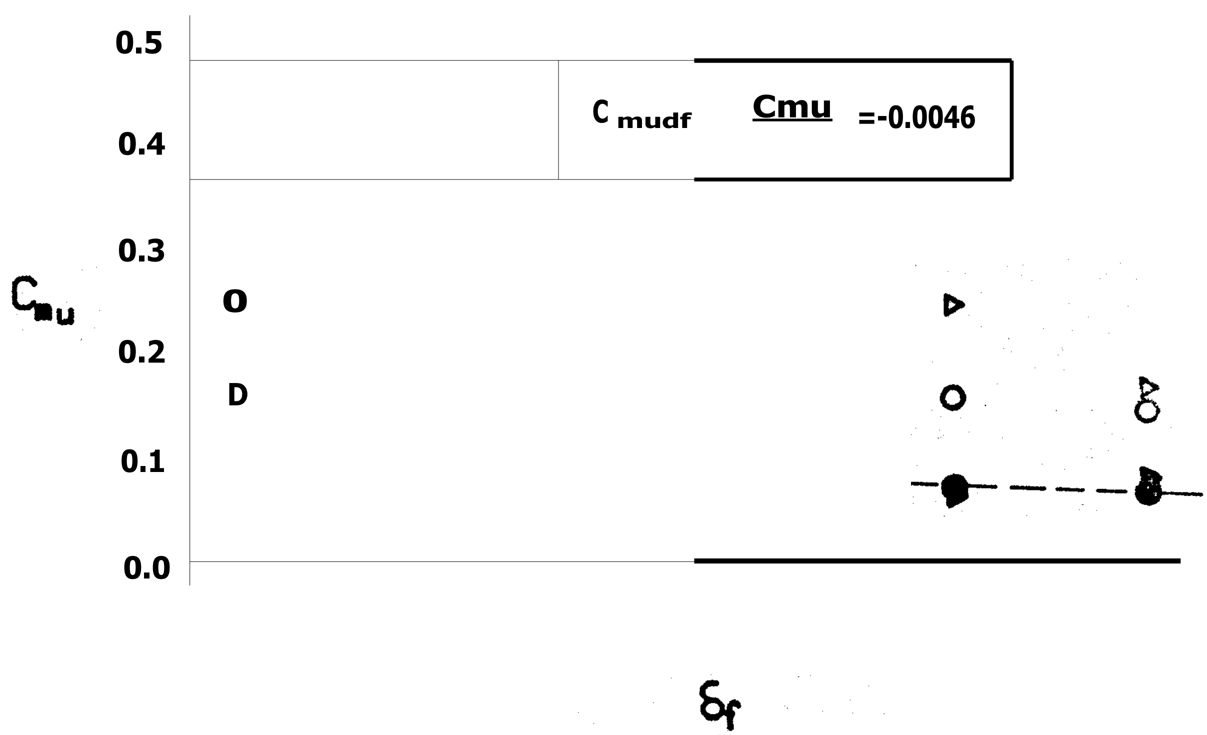

-3.6
-4.0
-4.4
-4.6
-5.2
-5.6
-6.0
0
12
δ_f

Fig. 6 (iii)

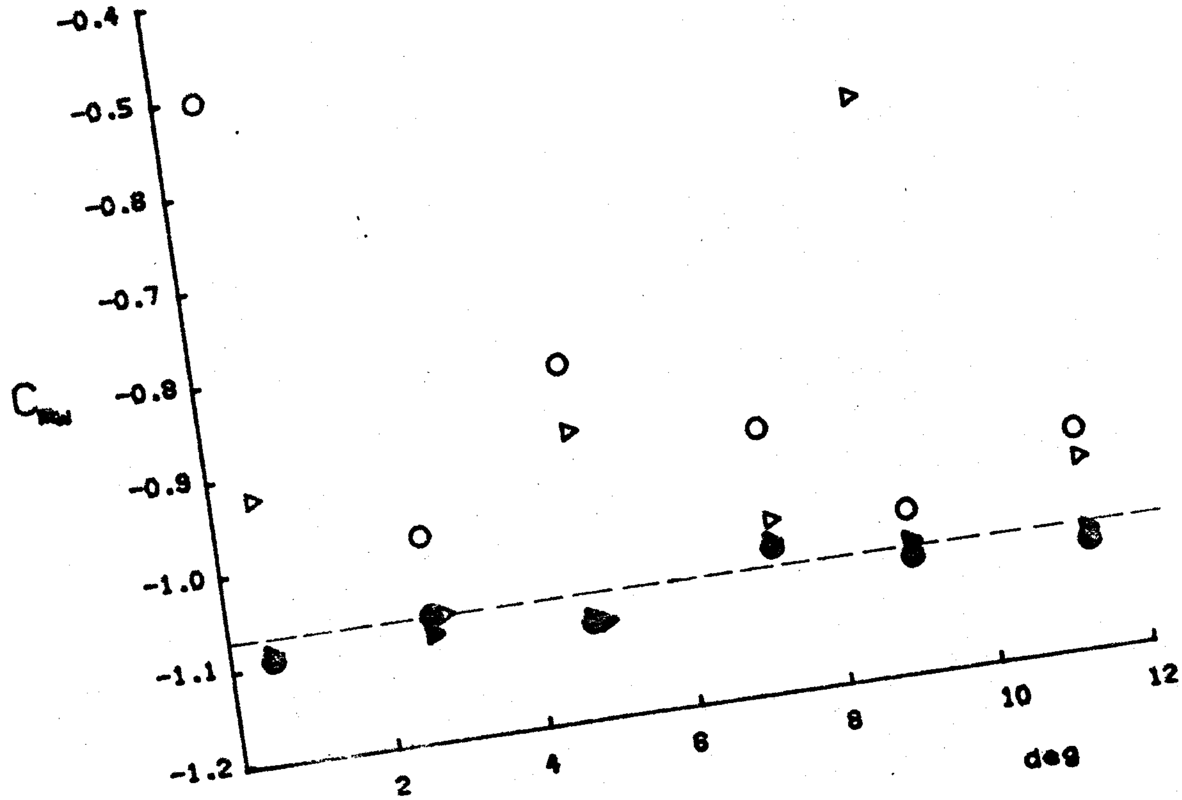


$C_{mo\delta_f} = \frac{dC_{mo}}{d\delta_f} = 0.004$
0.10
0.08
0.06
0.04
0.02
0.00
-0.02
-0.04
qmo
0
2
4
6
8
10
12
deg
δf

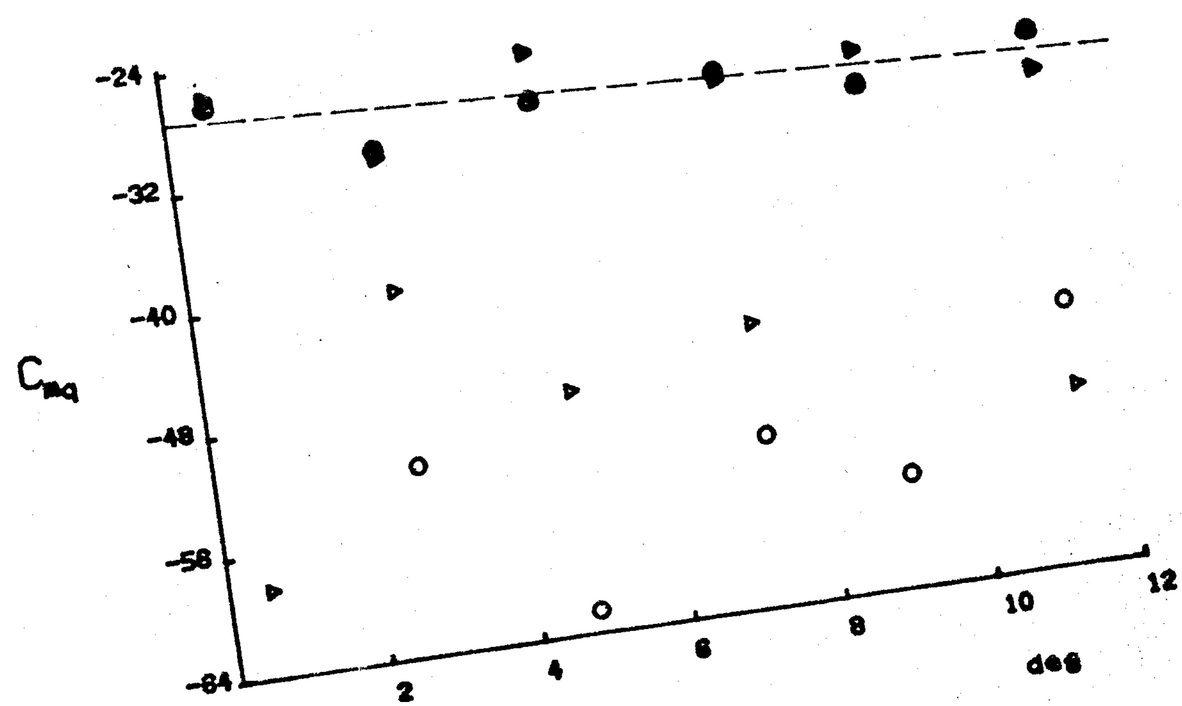


0.5
0.4
0.3
0.2
0.1
0.0
C mudf
Cmu
=-0.0046
Cmu
O
D
δf

Fig. 6 (iv)



-0.4
-0.5
-0.8
-0.7
-0.8
-0.9
-1.0
-1.1
-1.2
2
4
6
8
10
12
deg



-24
-32
-40
-48
-56
-64
2
4
6
8
10
12
deg

Fiq. 6 (v)

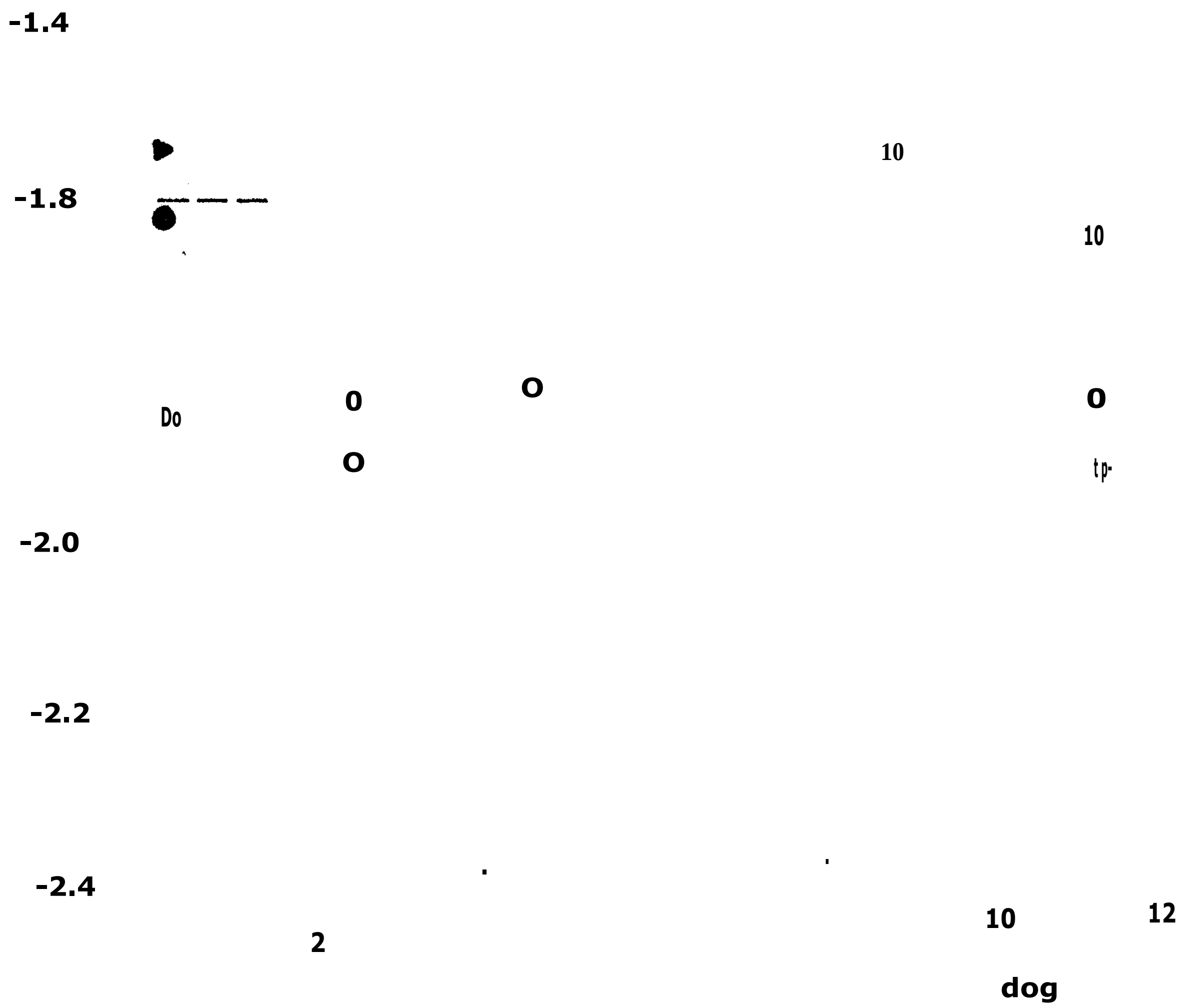


Fig. 6 vi)

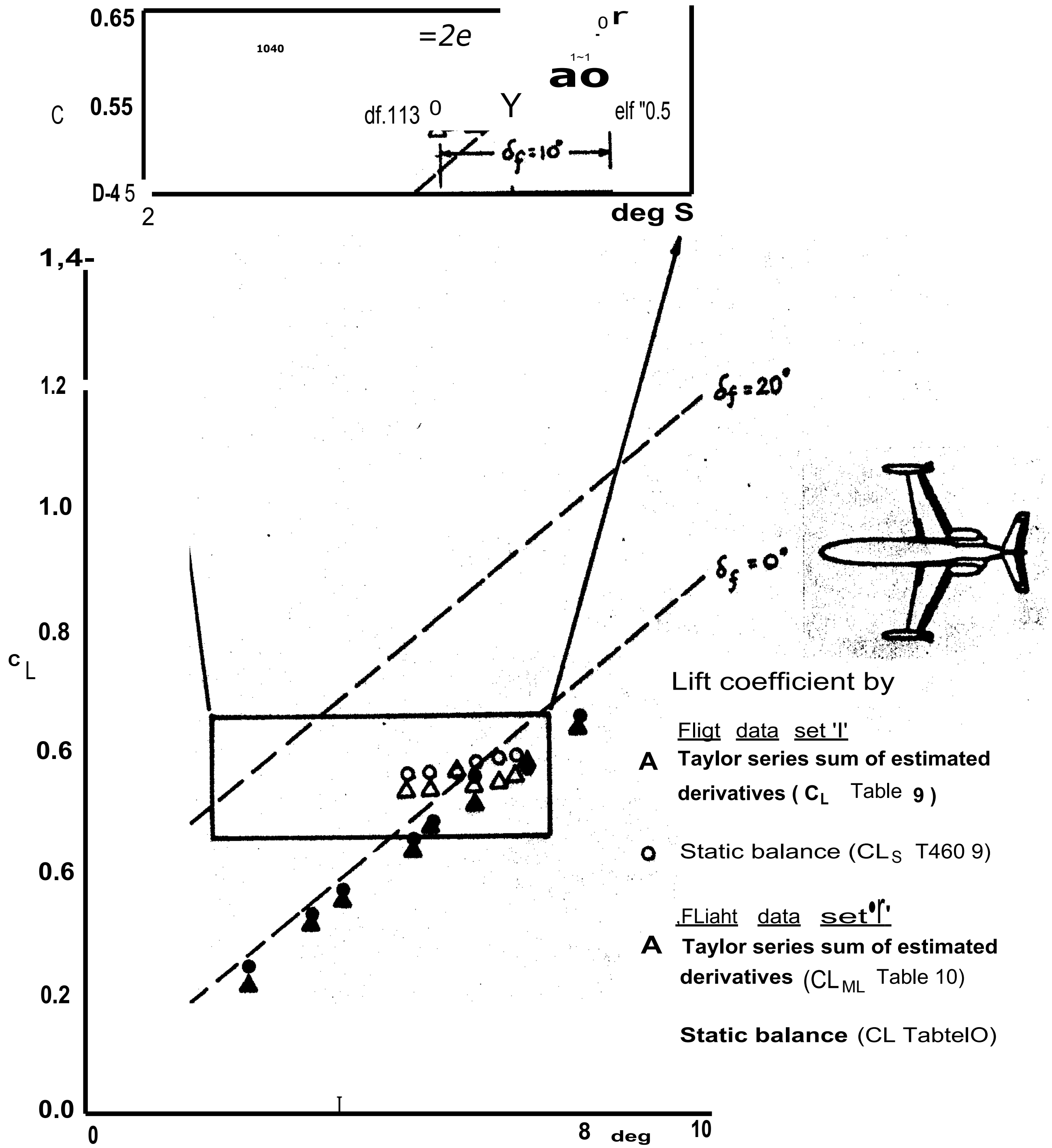


FIG. 7. ***Lift Coefficient (C_L) as a function of angle a attack***
(o) and flap deflection VI)

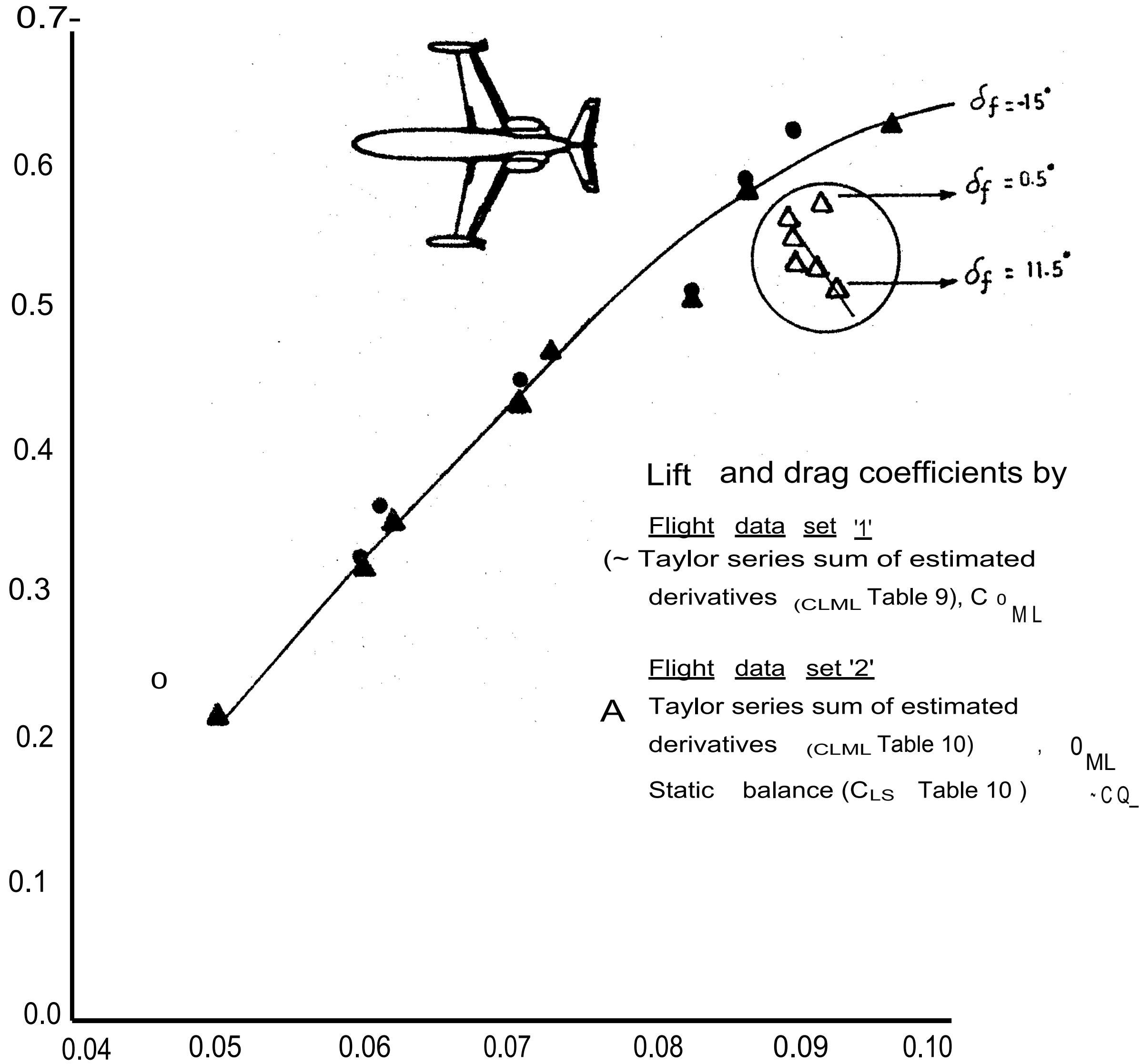


FIG. 8. Drag Polar C_L- as a function of flap deflection cf.

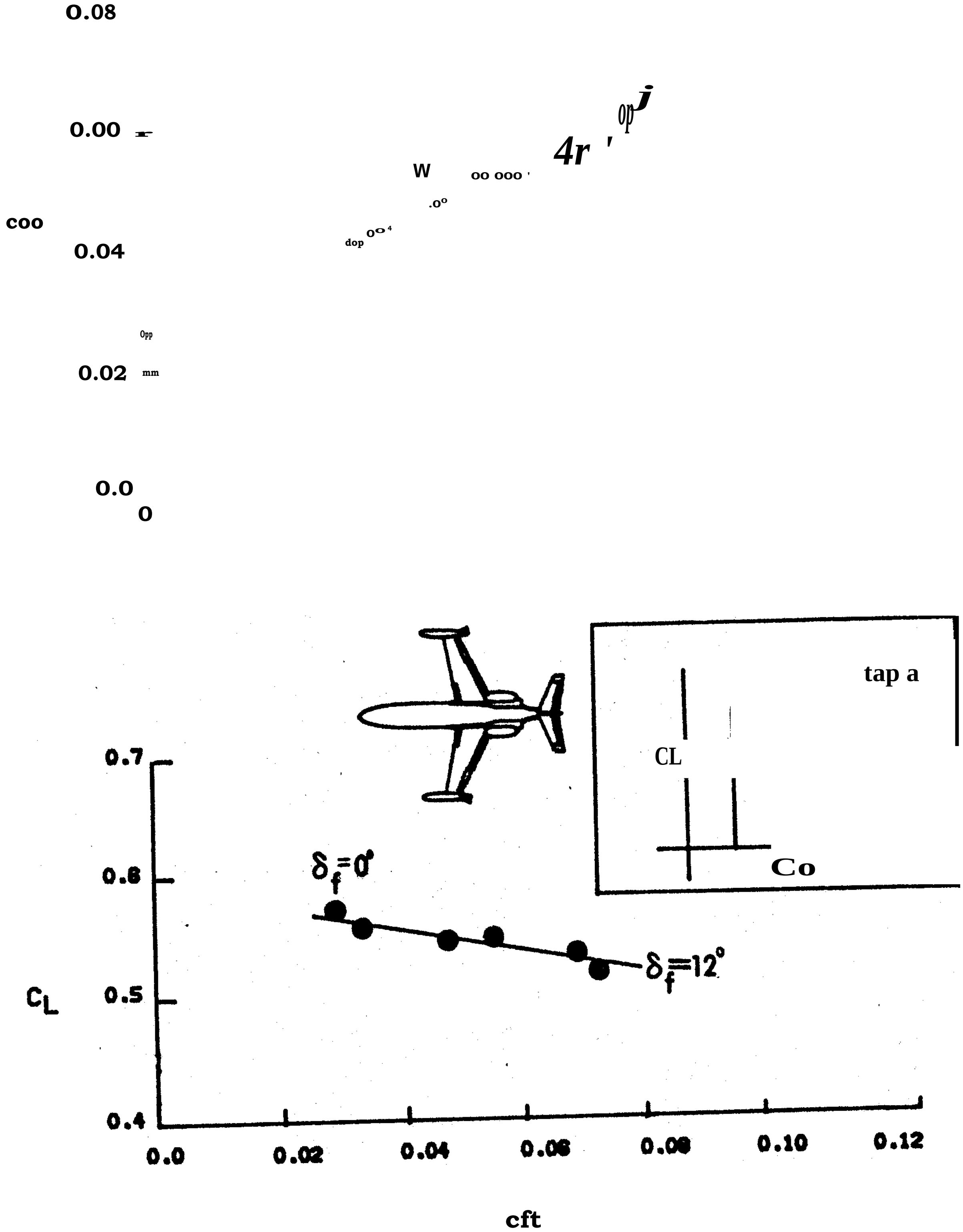


Fig. 9 : AERODYNAMIC CHARACTERISTICS AS A FUNCTION

OF FLAP DEFLECTION

Sheet ciassi ication

A oitica0 Lahog story	Documentation Sheet	Document Classification
Title	·:.cts of flap position on ,:-;.ituc.inal parameters of f4 · 320	Document No. TM SE 8602 Date of issue: Feb.1986
Author(s)	. Y ~ Jategaonkar malakrishna	Contents 75p, 9f, 10t, 1Gr
	~ diems Engineering Division	No. of copies: 25
External participation	Institute for Fl i ,ht Mechanics Dr LR, Braunschweig, FRG	NAL Project No. SE-223
Sponsor		Sponsor's Project No.

Approx.at H_-ad, Systems Engineering Division

Remarks

Keyulorci. P r:.m ter estimation, Flight test data, A.:,rodynamic derivatives

Abstract

Plr-w-eter estimation results to evaluate the effects of snail changes in flap position on the longitudinal 0_:-;ivatives of HFB-320 aircraft are presented in this report. Maximum likelihood estimation procedure used for kinematic consistency checking of flight test data and also for estimation of aerodynamic d,-r vatives. Linear and nonlinear models are uses: tc. estimate dimensional and non-dimensional derivat:.iv-as ~~ il~.