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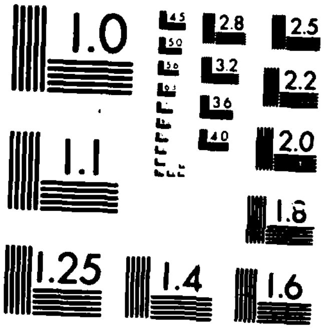

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MELBOURNE, VICTORIA

Structures Technical Memorandum 426

RIGID BODY STABILITY AUGMENTATION STUDIES FOR A WIND TUNNEL FLUTTER MODEL

by

C.J. LUDOWYK

Approved for Public Release

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RIGID BODY STABILITY AUGMENTATION STUDIES FOR A WIND TUNNEL FLUTTER MODEL

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C.J.LUDOWYK

SUMMARY

This report describes a simulation study undertaken by the author while attached to the RAE, Farnborough UK on a STUDS visit. The study was made in preparation for wind tunnel trials of flutter suppression control laws at DFVLR, Göttingen, West Germany.

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DATA AND NOTATION

Model Data: (See Fig. 1)

1. The model flies 3 kg of its own weight at 35 m/sec.

2. Overall model lift = 0.875 kg/taileron degree at 35 m/sec.

3. Span = 1.8 m.

4. Length = 2.0 m

Nomenclature:

K_{ACT}	Taileron actuator static gain constant.
K_1	Pitch rate weighting.
K_2	Pitch attitude weighting.
K_3	CG displacement weighting.
K_4	CG velocity weighting.
I_y	Pitch inertia about pivot (Initially 23.3 kg.m^2, modified to 21.2 kg.m^2).

M	Aircraft mass (Initially 66.6 kg. modified to 62.8 kg)		
S	Reference area (0.6 m^2).		
s	Laplace transform variable.		
V_{ZG}	Vertical velocity at CG, positive downwards (m/sec).		
V	Tunnel velocity (m/sec).		
X_g	Distance to CG	(Initially 1.441m) (Modified to 1.543m)	Positive to rear, measured from an origin near the nose
X_p	Distance to pivot	(Initially 1.51m) (Modified to 1.55m)	
X_{SPR}	Distance from mass relief spring to CG, positive for the CG in front of the spring. (Initially 0.068m, modified to 0.007m).		
q	Pitch rate, positive nose up (rad/sec).		
δ	Taileron angle (rad).		
δ_{TRIM}	Taileron trim angle (measured at -5 deg at 35 m/sec).		
δ_{SPAN}	Total taileron angle available (± 6 degrees).		

θ	Pitch angle, positive nose up (rads).
ρ	Density of air (1.225 kg/m^3).
τ_{ACT}	Taileron actuator time constant (0.01 sec).
τ_{INT}	Time constant of pseudo-integrator (6.0 sec).
τ_{NF}	Time constant of sensor noise filter. (0.03 sec).
ω_θ	Natural frequency of pitch spring (1.885 rad/sec).
ω_z	Natural frequency of heave spring (4.712 rad/sec).
ζ_θ	Assumed damping of pitch spring (2%).
ζ_z	Assumed damping of heave spring (5%).
Z_g	Vertical displacement at CG, positive downwards (m).
Z_{WA}	Wing lift coefficient (- 2.902/rad, acting at x = 1.657m).
Z_{WT}	Tail lift coefficient (- 0.672/rad, acting at x = 2.028m).
Z_q	Coefficient of lift due to pitch rate (- 0.581/rad acting at x = 2.072m)

NOTE:- There is a hidden (unit) length in the Z_q term due to Kaynes' method of derivation of the coefficients (Ref. 1).

$\hat{}$ Denotes estimated value.

$\dot{}$ Dot over a symbol denotes a derivative with respect to time.

1. INTRODUCTION

The author was attached to Flight Systems Department, RAE, Farnborough UK, from December 1982 to February 1984. Part of that time was spent on a task for Structures and Materials Department, under the direction of Mr. Colin Skingle of the Experimental Dynamics Group. This group was involved in a collaborative programme, under the auspices of GARTEur, on gust load alleviation and flutter suppression research. The programme was divided into three phases, classified as follows (Ref. 2):-

Phase 1: Production and demonstration of flutter suppression control laws on a wind tunnel aircraft model up to 1.7 Q_F, where Q_F is the flutter dynamic pressure.

Phase 2: Demonstration of flutter suppression control laws to 1.5 Q_F in the presence of gust inputs - both Dryden spectra and discrete gusts being programmable from a gust generator upstream of the model. My activities were confined to this phase.

Phase 3: Demonstration of flutter suppression and gust load alleviation (GLA) laws simultaneously, to 1.5 Q_F, using the gust generator.

The participating countries were UK (RAE/BAe), Holland (NLR), W. Germany (DFVLR/MBB) and France, using the wind tunnel facilities at DFVLR, Göttingen , W. Germany and a model supplied by DFVLR/MBB. Phase 1 was completed in 1982, Phase 2 completed in November 1983 and Phase 3 was scheduled for 1984.

2. SIMULATION STUDIES

The modified-Tornado model was mounted so that it was constrained to heave and pitch freedoms only (Fig. 1). The high pressure hydraulic supply to the model for control surface actuation was through looped flexible metal tubes, arranged like a safety pin, with low torsional and vertical stiffness. Phase 1 trials had shown that the rigid-body heave and pitch modes were very lightly damped. One of my activities in preparation for the Phase 2 trials was to investigate the rigid-body response of the model and to develop a stability augmentation system (SAS) control law for it. The RAE designed control system simulation and analysis computer package 'TSIM' (Refs. 3, 4) was used to program the rigid body and control law model simulation (Appendices A, B).

The roots of the author's rigid-body model modes were compared with those of a complete rigid-body plus aeroelastic model developed by Ian Kaynes of Structures Dept. RAE in order to verify the simpler model. The comparisons were performed for two values of X_{SPR} with a variety of tunnel speeds. These comparisons are depicted in Tables A and B and show that results of the simpler model, Type 2, are not significantly different from those of the full model, Type 1.

The root locus of the basic aircraft with changing velocity V is depicted in Fig. 2. for nominal values of X_{SPR} = 0.068m and X_G = 1.441m. The unstable rigid-body mode is apparent at V = 28 m/sec although in practice, the damping was such that this mode was marginally stable over the whole speed range.

Phase 1 trials also showed that with the model trimmed at nominal X_p and X_G values and X_{SPR} = 0.007m (i.e. mass relief spring approximately at the pivot), the tailplane trim angle was found to be large (approximately -5 degrees). Most of the available $\pm 6^0$ movement was used just for trimming and very little was left for using the tail as an active control. Suggested remedies were:-

(a) to move the CG rearward and hence reduce the required tail download to trim, and/or

(b) to move the spring attachment forward.

Both these options were explored using TSIM and both tended to destabilize the rigid aircraft modes even further, as evident in the root locus diagrams, Figs 3 and 4.

Therefore, the stability augmentation system (SAS) was required to provide basic stability and if possible, to extend the damping of these modes to a value more representative of that of free flying aircraft. Relevant feedback sensors available were accelerometers in the fuselage (nose and near the CG) and a pitch rate gyro.

As a design exercise, it was assumed that the designer had latitude in the values of X_{SPR} and X_G. Thus one design goal was to choose values of X_{SPR} and X_G for optimum 'basic aircraft' modal damping before applying the SAS control law. However as the study progressed, information was received from DFVLR, Göttingen that the spring and CG positions were immutable. The simulation studies showed that when the SAS law was applied, root sensitivity to variations in spring attachment position and CG position became small. Therefore even if these positions had been variable, there would have been little value in changing them.

The root locus of the basic aircraft with changing velocity V is depicted in Fig. 2 for the original values of X_{SPR} = 0.068m and X_G = 1.441m. The unstable rigid body mode is apparent at V = 28 m/sec. Fig. 3 shows the root locus for a fixed speed V = 35.0 m/sec with varying X_{SPR}. The stabilizing effect of increasing X_{SPR} (i.e. moving the location of the spring forward) is clearly shown. The similar stabilizing effect of moving the CG location further aft (increasing X_G) is shown in Fig. 4. These effects are summarised in the dual-parameter families of root loci in Figs. 5 and 6. From these plots, values of X_{SPR} = 0.068m and X_G = 1.6m were chosen to give optimum open-loop stability of the basic aircraft without feedback. This reference open-loop condition (Fig. 7) was used for initial studies of an SAS control law. Fig. 7 shows that the damping ratio of the dominant open-loop poles for the values chosen is 0.15 at 35 m/sec. The root sensitivity to X_{SPR} is quite large in this region, as shown in the time responses to a taileron step for X_{SPR} values of 0.07, 0.08 and 0.09m in Figs. 8, 9, and 10. One root has become positive real in Fig. 10 for a 0.01m increase in X_{SPR}.

3. CONTROL LAW DESIGN

The four states corresponding to the two Rigid Body Modes (RBM) of pitch and heave, i.e. pitch rate q, pitch angle θ, CG vertical displacement Z_G and velocity V_{ZG} were fed back according to the law

$$\delta = (K_1\ q + K_2\ \Theta + K_3\ Z_G + K_4\ V_{ZG}).\ \ G_A$$

where $G_A = \frac{K_{ACT}}{(1 + \tau_{ACT}.s)}$, the actuator transfer function.

Numerous root-loci and time responses were evaluated using trial weightings of the four states before the following RBM state feedback law was obtained

$$\delta = (0.4\ q + \theta + 0.2\ Z_G).G_A \qquad \text{for } X_G = 1.6\text{m}$$
$$\text{and } X_{SPR} = 0.07\text{m}$$

The root locus of this law for varying wind speed is shown in Fig. 11(a) using a first-order model for the taileron actuator, with time constant τ_{NF} of 0.01 seconds.

For practical implementation of this law, θ and Z_G were estimated as follows:-

since $\dot{\theta} = q$

$$\theta = \frac{q}{s}$$

Using pseudo-integration, the integral time constant τ_{INT} was selected to be about 10 times the time constant of the mode to be controlled. The pseudo-integrator was also gain-matched (for large s) with the perfect integrator.

This yielded $\hat{\theta} = \dfrac{\tau_{INT}.\ q}{1 + \tau_{INT}s}$

Similar reasoning on the estimates $\hat{V}_{ZG}$ and $\hat{Z}_G$ yielded

$$\hat{V}_{ZG} = \frac{\tau_{INT}.\ \ddot{Z}_G}{1 + \tau_{INT}.s}$$

$$\hat{Z}_G = \frac{(\tau_{INT})^2\ .\ \ddot{Z}_G}{(1 + \tau_{INT}.s)^2}$$

Thus the block diagram of Fig. 12 is obtained, with the addition of a sensor noise filter and first-order model of the actuator, with the estimate $\hat{\delta}$ given by

$$\hat{\delta} = \frac{K_{ACT}}{(1 + \tau_{ACT.s})} \quad \frac{K_1 . q}{(1 + \tau_{NFs})} + \frac{K_2 . \tau_{INT} . q}{(1 + \tau_{INT.s})(1 + \tau_{NF.s})}$$

$$+ \frac{K_3 . \tau_{INT}^2}{(1 + \tau_{INT.s})^2} + \frac{K_4 . \tau_{INT}}{(1 + \tau_{INT.s})}$$

The root locus of the estimated RBM state feedback is shown in Fig. 11(b) for the same weightings determined previously, i.e. $K_1 = 0.4$, $K_2 = 1.0$, $K_3 = 0.2$ and $K_4 = 0$. The comparison with Fig. 11(a) shows that the behaviour of the pitch-and heave-mode roots is essentially similar with estimated state feedback.

The TSIM models of the 'basic aircraft', aircraft with RBM state feedback, and aircraft with estimated RBM state feedback are given in Appendix B.

Subsequently, information from Göttingen on the final fixed positions of X_{SPR} and X_G were obtained (0.007m and 1.543m respectively) together with updated values of M and I_y.

The basic aircraft root locus for the updated data is shown in Fig. 13, where the aircraft becomes unstable at 24 m/sec and stays unstable through to 60 m/sec. (Compare with Fig. 2 where stability is re-established at about 55 m/sec.)

A similar study to that previously undertaken was performed to choose an acceptable control law. Because this was to be the final version for practical implementation, a simplified pitch-rate feedback law was evaluated and subsequently judged acceptable. The root locus of the pitch-rate feedback law is shown in Fig. 14. The time responses for a taileron step displacement of 5 degrees with three wind speeds are shown in Fig. 15. A similar time response for the best evaluated RBM state feedback law, in this case given by -

$$\delta = 0.5\, q + \Theta + 0.1\, Z_G - 0.1\, V_{ZG}$$

is shown in Fig. 16. This shows a reduction in an already small heave amplitude of about 10mm at the expense of slightly degraded pitch damping. Thus it is apparent there is little benefit from feeding back all the rigid body modes in this case. In practice, the law $\delta = 0.5\, q$ was used with success.

The time response at 35 m/sec for the modified data 'basic aircraft' is also shown in Fig. 17 where ± 1 metre heave amplitudes are seen to build up within 5 seconds.

This divergent rigid body oscillation was the mechanism which slammed the model into the end stops during the French team's trials. Fortunately, the damage was repaired quickly and did not significantly affect the performance of the subsequent RAE team.

4. CONCLUSION

This document has outlined some work undertaken for Structures and Materials Department, RAE as a small part of the author's attachment. Because the task was also a valuable learning exercise, the study of a suitable feedback law was made more comprehensively than normally justifiable on a simple model.

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APPENDIX A

Rigid aircraft equations of motion

Lift of whole aircraft, with tail and trim angle zero $= \rho S V^2 Z_{WA} . \alpha$

$\therefore$ Lift of Tail at Trim angle $\delta = \rho S V^2 . Z_{WT}\, \delta$

$$\alpha \approx \left(\frac{\dot{Z}_G}{V} + \Theta \right)$$

Spring stiffness force $= M\omega_z^2 \; (Z_G + X_{SPR} . \Theta)$

Spring damping force $= 2M\zeta_z\, \omega_z\, (\dot{Z}_G + X_{SPR} . q)$

Therefore the force eqn. is -

$$M\ddot{Z}_G = \rho S V^2 Z_{WA} \left({}^{\dot{Z}_G}/_{V} + \Theta \right) + \rho S V Z_q . q - M\omega_z^2 \; (Z_G + X_{SPR} . \Theta)$$

$$- 2M\zeta_z \omega_z \; (\dot{Z}_G + X_{SPR} . q) + \rho S V^2 Z_{WT} . \delta \qquad (1)$$

Similarly the Moment eqn. is -

$$I_y.\dot{q} = \rho SV^2 Z_{WA}\ (1.657 - X_G)(\dot{Z}_G/_V + \Theta) + \rho SVZ_q.\ (2.0723 - X_G).q$$

$$- M\omega_z^2\ .\ (Z_G + X_{SPR}\ .\ \Theta).\ \ X_{SPR}$$

$$- 2M\zeta_z\ \omega_z\ (\dot{Z}_G + X_{SPR}.q)\ .\ X_{SPR}$$

$$- I_y.\omega_\theta^2.\ \Theta - 2\ I_y\ \zeta_\theta\ \omega_\theta\ .\ q$$

$$+ \rho SV^2 Z_{WT}\ (2.028 - X_G)\ .\ \delta \qquad (2)$$

$$\dot{Z}_G = V_{ZG} \qquad (3)$$

$$\dot{\Theta} = q \qquad (4)$$

APPENDIX B - TSIM MODELS

B1. TSIM Variables and constants Nomenclature

TSIM Nomenclature	Previous Nomenclature
AREA	S
FREQZ	ω_Z
FREQTH	ω_θ
DAMPZ	ζ_z
DAMPTH	ζ_θ
DEL	δ
IY	I_y
KACT	K_{ACT}
K1 .... K4	K_1 .... K_4
RHO	ρ
TAUACT	τ_{ACT}
TAUINT	τ_{INT}
TAUNF	τ_{NF}
THETA	Θ
VZG	V_{ZG}
XG	x_G
XSPR	x_{SPR}
ZG	Z_G
ZWA	Z_{WA}
ZWT	Z_{WT}
ZQ	Z_q

APPENDIX B (CONT'D)

B2. TSIM model of basic aircraft rigid-body response

```
    GARTEUR MODEL RIGID-BODY EQUATIONS

    TEXT GARTEUR MODEL RIGID-BODY MODES
    STATE ZG,THETA,Q,VZG
    VARIABLE DEL,XSPR,V
    DERIVATIVE RHO,AREA,ZWA,ZQ,M,FREQZ,DAMPZ
    DERIVATIVE ZWI,XG,IY,FREQTH,DAMPTH
    WAIT
    K = RHO*AREA*V*V
    SVZG=K*ZWA*(VZG/V+THETA)/M + K*ZQ*Q/V/M
+           -FREQZ*FREQZ*(ZG+XSPR*THETA)
+           -2*DAMPZ*FREQZ*(VZG+XSPR*Q) + K*ZWI*DEL/M
    SQ   =K*ZWA*(1.657-XG)*(VZG/V+THETA)/IY
+           +K*ZQ*(2.0723-XG)*Q/V/IY
+           -M*FREQZ*FREQZ*(ZG+XSPR*THETA)*XSPR/IY
+           -2*M*DAMPZ*FREQZ*(VZG+XSPR*Q)*XSPR/IY
+           -FREQTH*FREQTH*THETA - 2*DAMPTH*FREQTH*Q
+           +K*ZWT*(2.028-XG)*DEL/IY
    STHETA=Q
    SZG   =VZG
    SIMSTOP
    END
```

B3. Aircraft rigid body model with full state feedback law

```
       GARTEUR MODEL IDEAL F/B SIMULATION

       TEXT GARTEUR MODEL RIGID-BODY MODES
       STATE ZG,THETA,Q,VZG,DEL
       VARIABLE XSPR,V
       DERIVATIVE RHO,AREA,ZWA,ZQ,M,FREQZ,DAMPZ
       DERIVATIVE ZWT,XG,IY,FREQTH,DAMPTH
       DERIVATIVE K1,K2,K3,K4,TAUACT
       WAIT
       K = RHO*AREA*V*V
       SVZG=K*ZWA*(VZG/V+THETA)/M + K*ZQ*Q/V/M
    +       -FREQZ*FREQZ*(ZG+XSPR*THETA)
    +       -2*DAMPZ*FREQZ*(VZG+XSPR*Q) + K*ZWT*DEL/M
       SQ   =K*ZWA*(1.657-XG)*(VZG/V+THETA)/IY
    +       +K*ZQ*(2.0723-XG)*Q/V/IY
    +       -M*FREQZ*FREQZ*(ZG+XSPR*THETA)*XSPR/IY
    +       -2*M*DAMPZ*FREQZ*(VZG+XSPR*Q)*XSPR/IY
    +       -FREQTH*FREQTH*THETA - 2*DAMPTH*FREQTH*Q
    +       +K*ZWT*(2.028-XG)*DEL/IY
       STHETA=Q.
       SZG  =VZG
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C      IDEAL CONTROL LAW WITH 1ST. ORDER ACTUATOR
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
       SDEL=-DEL/TAUACT+1/TAUACT*(K1*Q+K2*THETA+K3*ZG+K4*VZG)
       SIMSTOP

       END
```

APPENDIX B (CONT'D)

B4. Aircraft Rigid Body model with state-estimated feedback law

```
       GARTEUR MODEL WITH STATE-ESTIMATED F/B

       TEXT GARTEUR MODEL RIGID-BODY MODES
       STATE ZG,THETA,Q,VZG,DEL,X1,X2,X3,X4
       VARIABLE XSPR,V,TAUINT,KACT,TAUACT
       VARIABLE TAUNF,K1,K2,K3,K4
       DERIVATIVE RHO,AREA,ZWA,ZQ,M,FREQZ,DAMPZ
       DERIVATIVE ZWT,XG,IY,FREQTH,DAMPTH
       WAIT
       K = RHO*AREA*V*V
       SVZG=K*ZWA*(VZG/V+THETA)/M + K*ZQ*Q/V/M
     +     -FREQZ*FREQZ*(ZG+XSPR*THETA)
     +     -2*DAMPZ*FREQZ*(VZG+XSPR*Q) + K*ZWT*DEL/M
       SQ  =K*ZWA*(1.657-XG)*(VZG/V+THETA)/IY
     +     +K*ZQ*(2.0723-XG)*Q/V/1Y
     +     -M*FREQZ*FREQZ*(ZG+XSPR*THETA)*XSPR/IY
     +     -2*M*DAMPZ*FREQZ*(VZG+XSPR*Q)*XSPR/IY
     +     -FREQTH*FREQTH*THETA - 2*DAMPTH*FREQTH*Q
     +     +K*ZWT*(2.028-XG)*DEL/IY
       STHETA=Q
       SZG  =VZG
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C      CONTROL LAW WITH ESTIMATED STATE F/B, 1ST. ORD. ACTUATOR
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
       SX1 = (1/TAUNF)*(-X1 + Q)
       SX2 = (1/TAU1NT)*(-X2 + TAU1NT*X1)
       SX3 = (1/TAU1NT)*(-X3 + TAU1NT*SVZG)
       SX4 = (1/TAU1NT)*(-X4 + TAU1NT*X3)
       SDEL= (1/TAUACT)*(-DEL+KACT*(K1*X1+K2*X2+K4*X3+K3*X4))
       SIMSTOP
       END
```

TABLE A SIMULATION COMPARISON FOR $X_{SPR} = 0.0$

		Roots of Rigid body modes		
Simulation Type	V(m/sec)	Real	Imag.	Damping [illegible]
1	0	- 0.24 - 0.04	4.74 1.87	0.05 0.02
2	0	- 0.24 -0.04	4.71 1.88	0.05 0.02
1	20	- 0.67 0.05	5.12 2.77	0.13 0.02
2	20	- 0.76 + 0.05	4.74 3.32	0.16 - 0.01
1	40	- 1.18 + 0.02	6.75 3.51	0.17 - 0.005
2	40	- 1.42 + 0.27	6.18 4.39	0.22 - 0.06
1	60	- 1.58 - 0.02	9.16 3.74	0.17 0.005
2	60	- 1.66 + 0.08	8.63 4.63	0.19 - 0.02

TABLE B SIMULATION COMPARISON FOR X_{SPR} = 0.068m

Simulation Type	V(m/sec)	Roots of rigid body modes		
		Real	Imag.	Damping Ratio
1	0	- 0.24 - 0.04	4.81 1.82	0.05 0.02
2	0	- 0.24 - 0.04	4.74 1.87	0.05 0.02
1	20	- 0.65 - 0.06	5.27 2.55	0.12 0.02
2	20	- 0.67 - 0.04	5.12 2.73	0.13 0.02
1	40	- 1.13 -0.03	6.93 3.19	0.16 0.01
2	40	- 1.17 + 0.02	6.71 3.47	0.17 - 0.005
1	60	- 1.55 - 0.05	9.30 3.41	0.16 0.02
2	60	- 1.57 -0.02	9.06 3.70	0.17 0.004

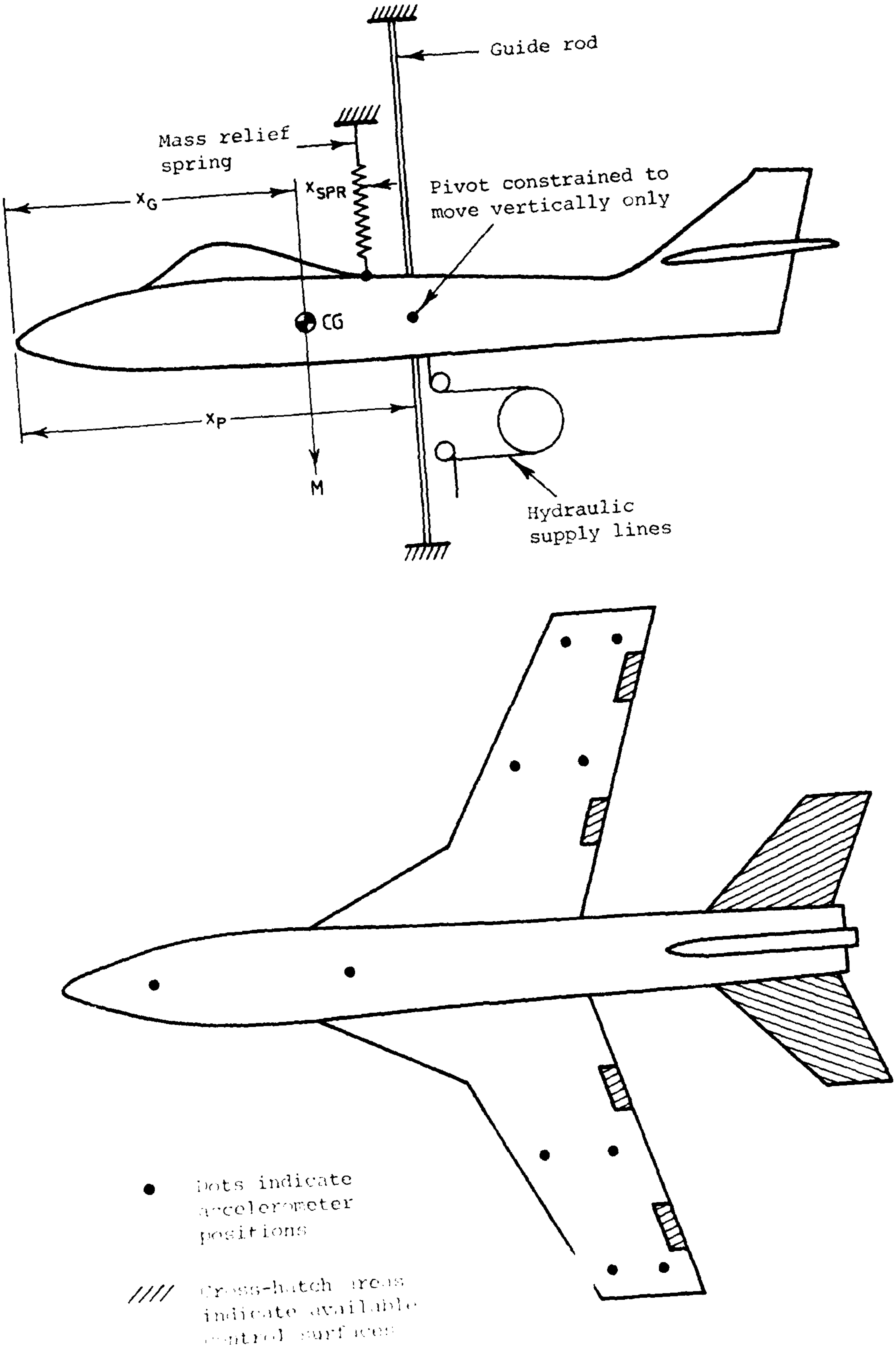
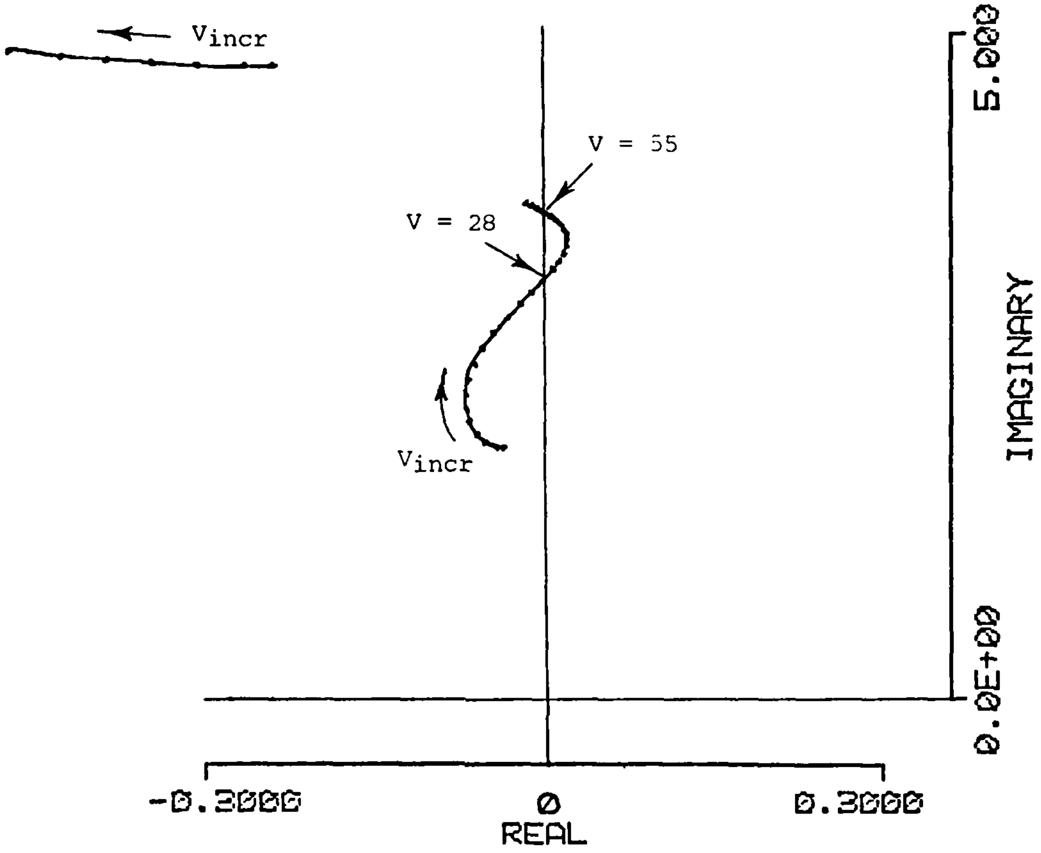
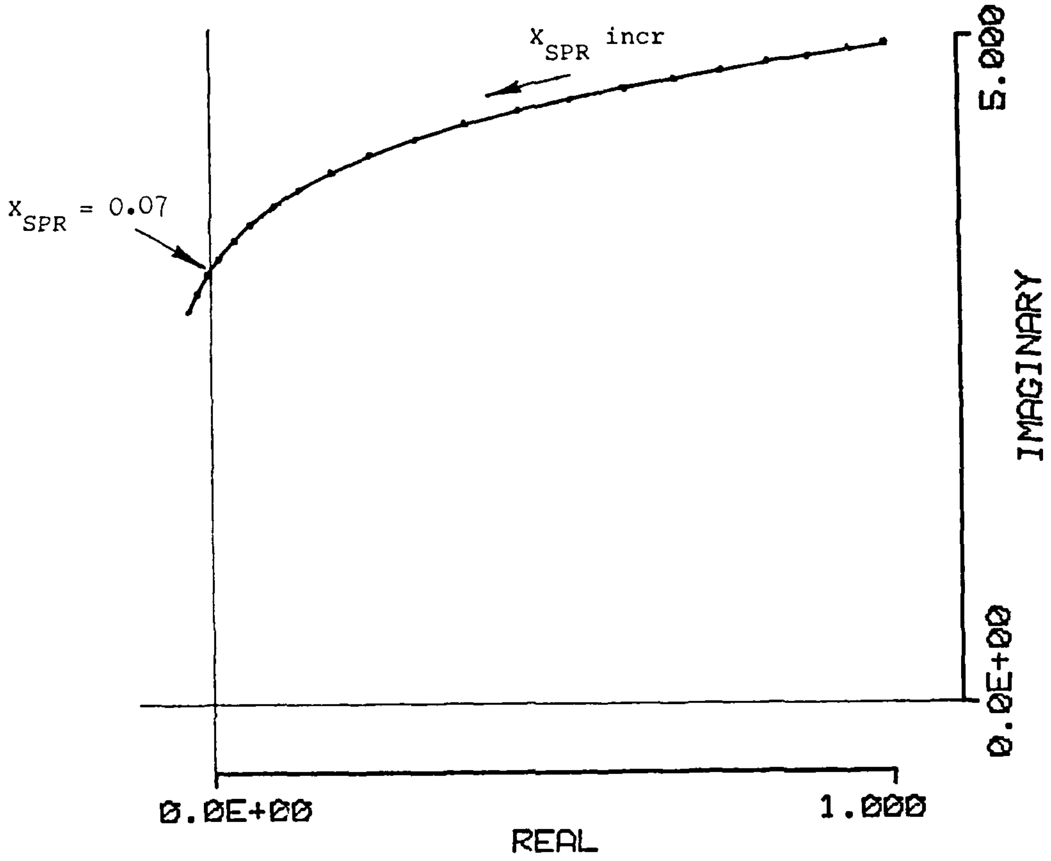


FIG. 1 GARTEUR MODEL



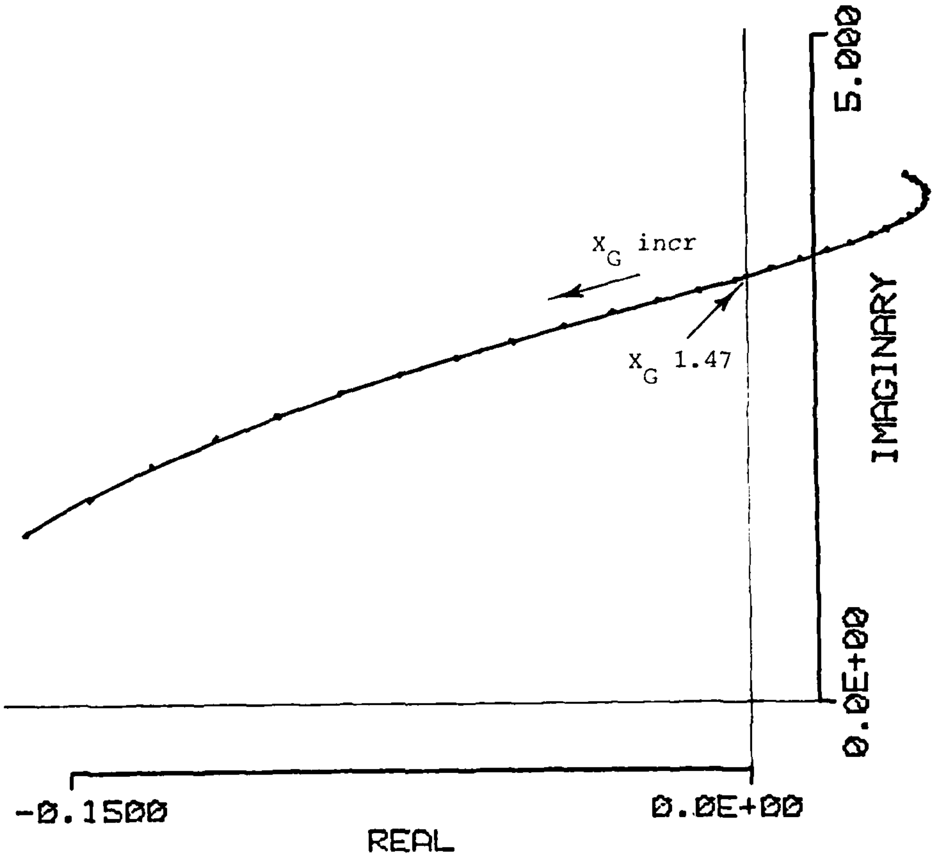
$X_{SPR} = 0.068$m; $X_G = 1.441$m; $0.01 \leq V \leq 60.0$ in steps of 2.0 m/sec

FIG 2. ROOT LOCUS - BASIC A/C (VARIATION WITH V)



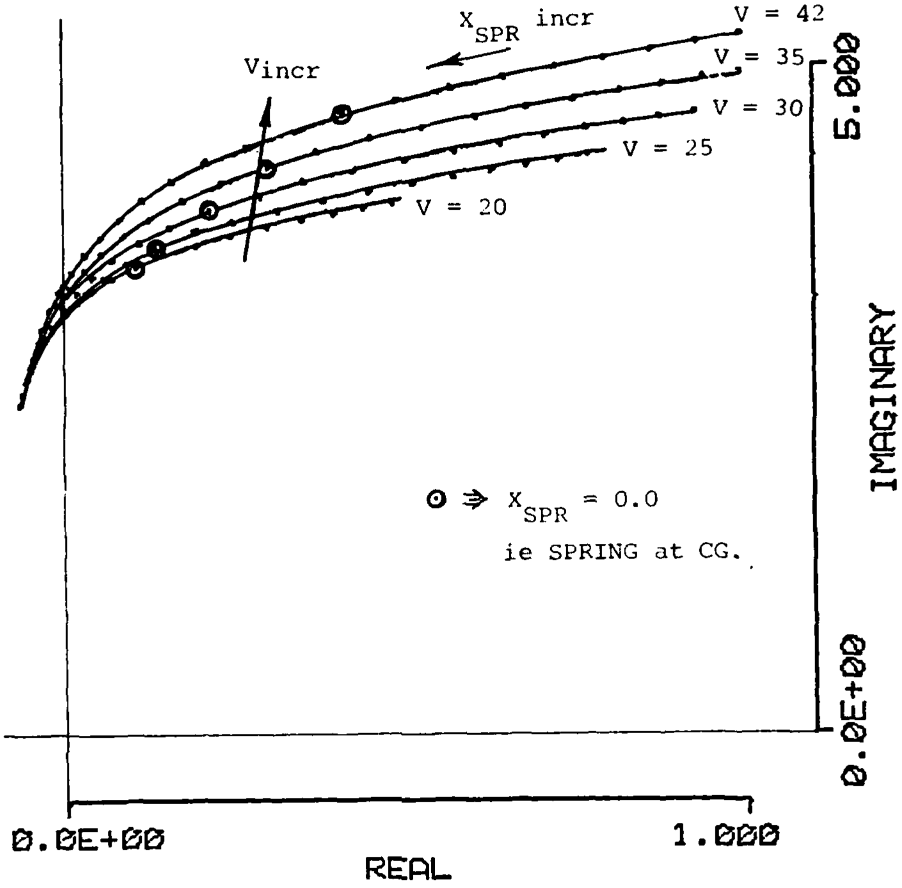
X_G = 1.441m; V = 35.0 m/sec; $-0.1 \le X_{SPR} \le 0.1$ in steps of 0.01m

FIG. 3 ROOT LOCUS - BASIC A/C (VARIATION WITH X_{SPR})



$V = 35.0$ m/sec; $X_{SPR} = 0.068$m; $1.3 < X_G < 1.6$ in steps of 0.01m

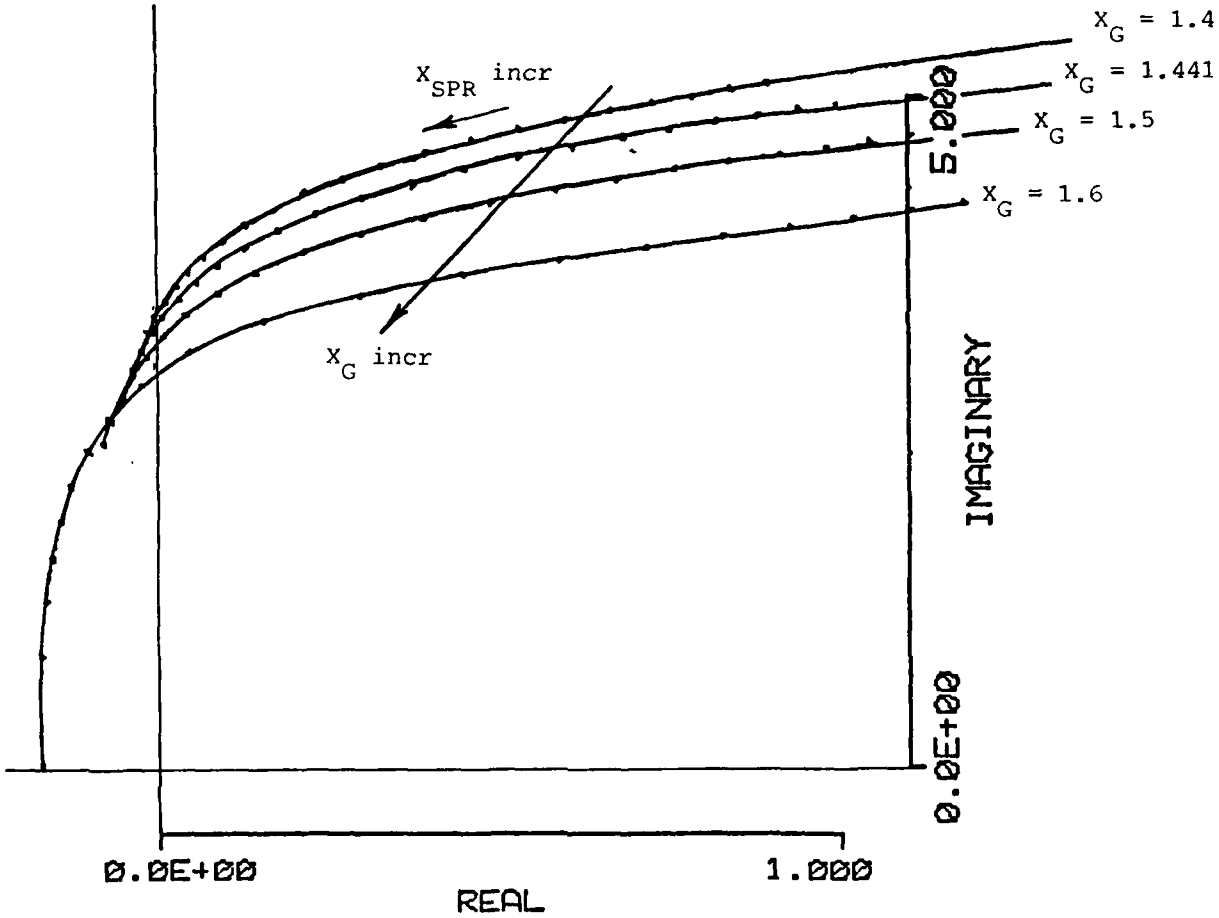
FIG. 4 ROOT LOCUS - BASIC A/C (VARIATION WITH X_G)



$X_G = 1.441\text{m}; -0.1 \leq X_{SPR} \leq 0.1$ in steps of 0.01m;

$20 \leq V \leq 40$ in steps of 5 m/sec

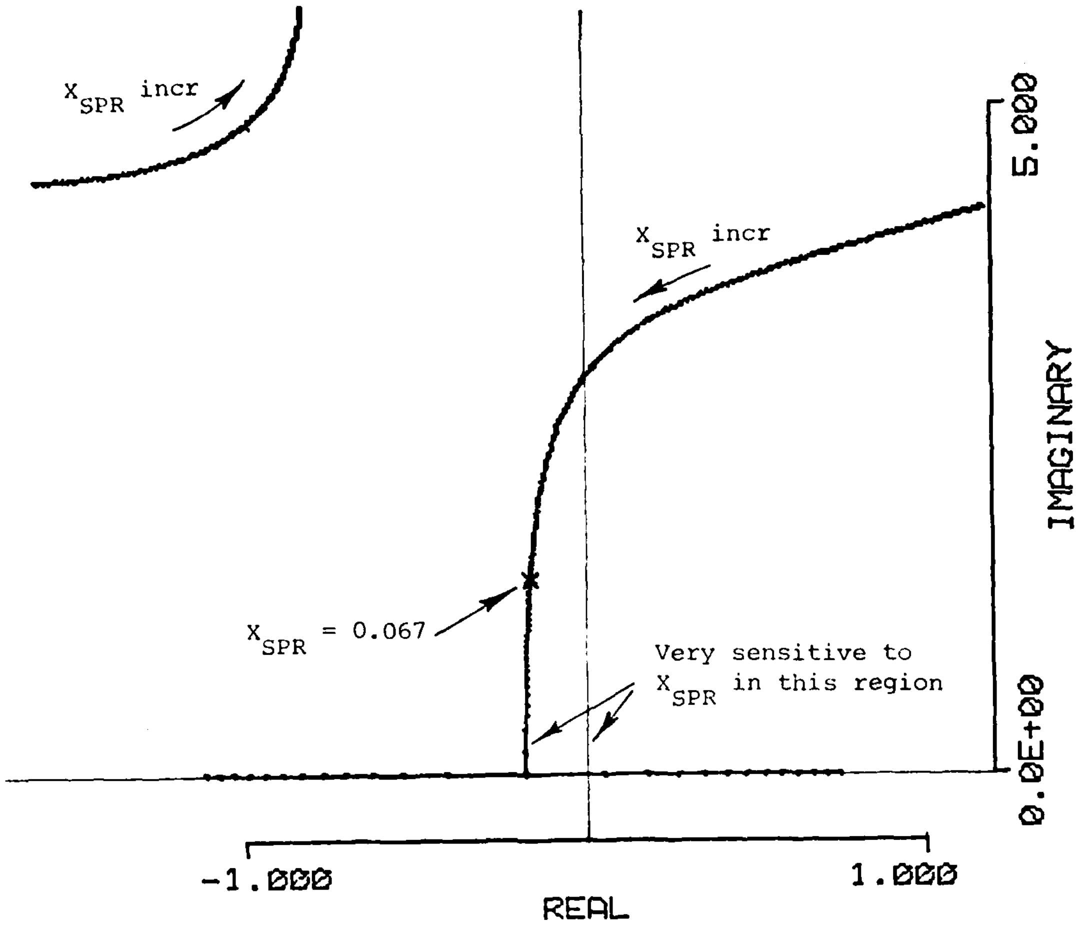
FIG. 5 ROOT LOCUS - BASIC A/C (VARIATION WITH V AND X_{SPR})



V = 35 m/sec; - 0.1 ≤ X_{SPR} ≤ 0.1 in steps of 0.01m

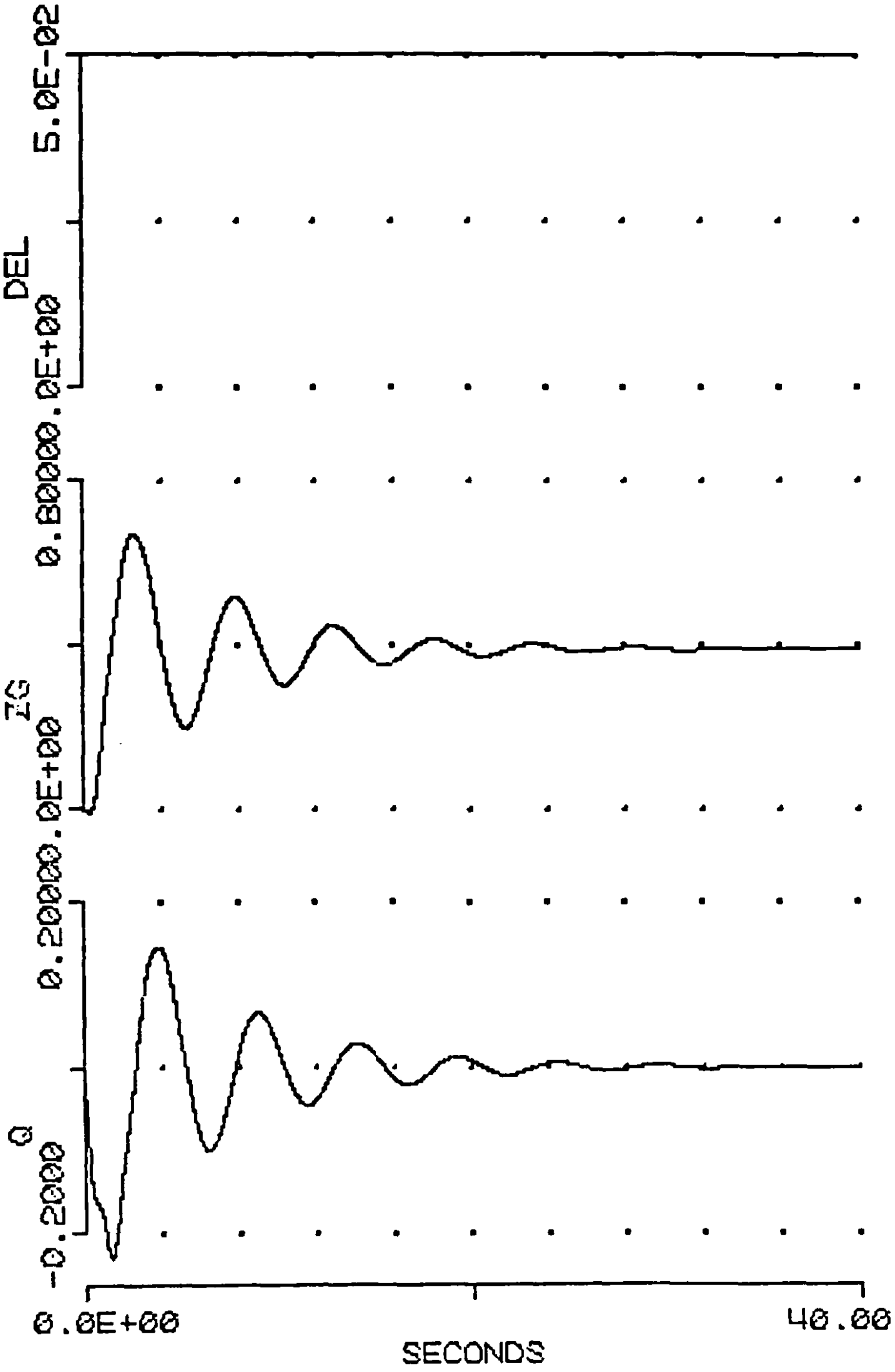
X_G varies as shown

FIG. 6 ROOT LOCUS - BASIC A/C (VARIATION WITH X_{SPR} AND X_G)



V = 35 m/sec; X_G = 1.6m; -0.1 X_{SPR} 0.1 in steps of 0.0001m

FIG. 7 ROOT LOCUS - BASIC A/C

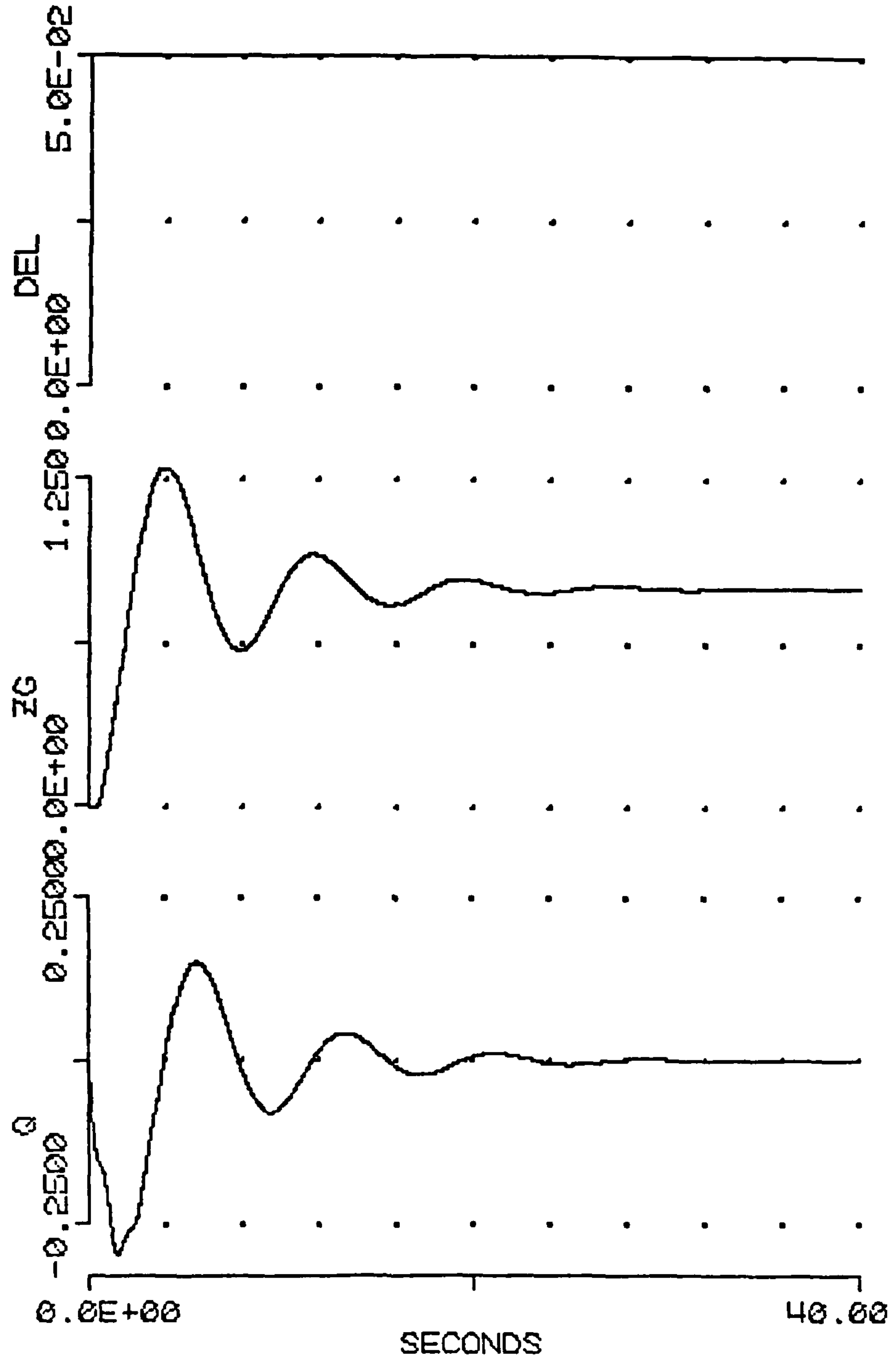


X_{SHF} = 0.07m

X_G = 1.6m

V = 35 m/sec

FIG. 8 TIME RESPONSE - BASIC A/C

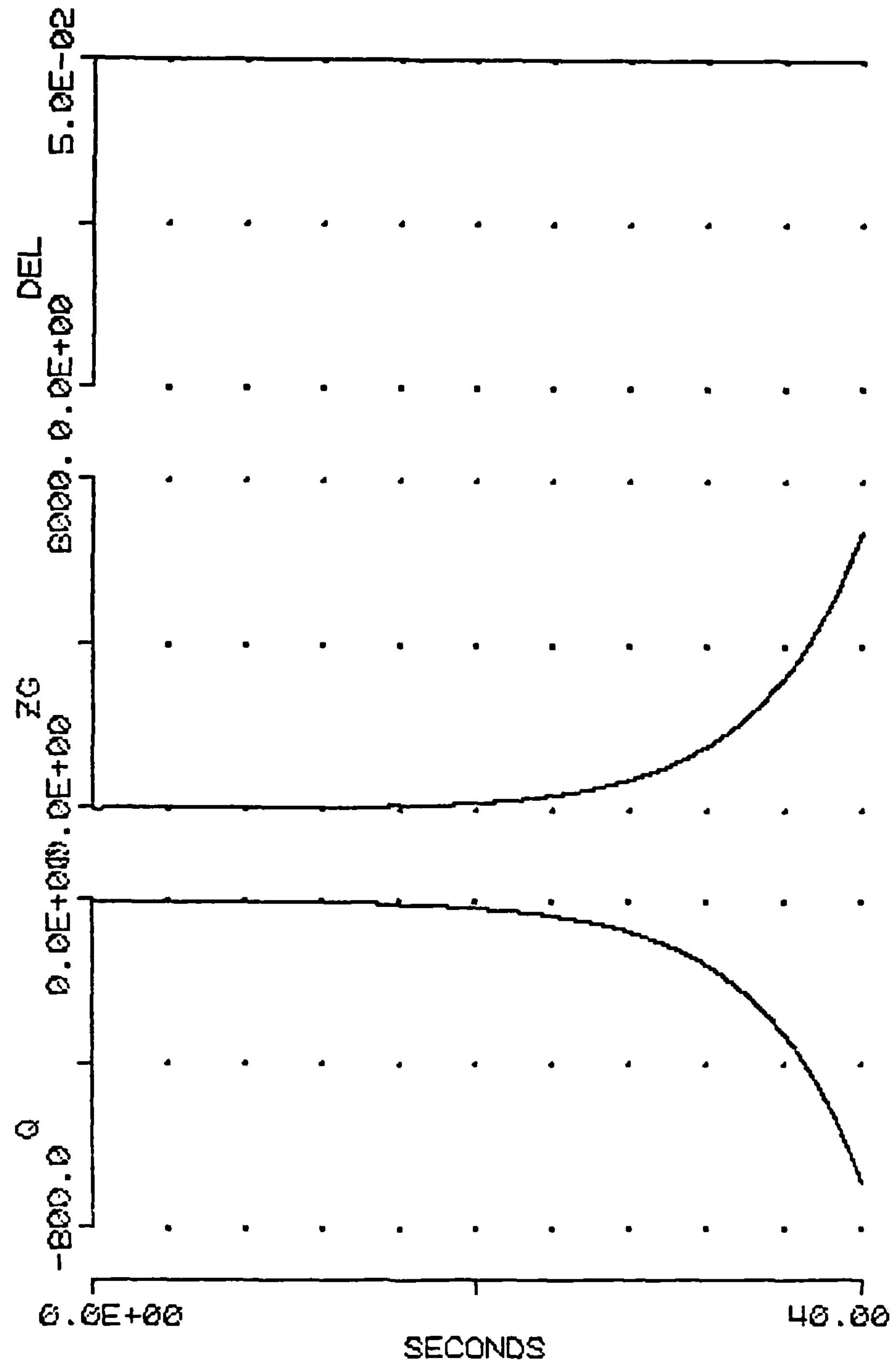


$X_{SPR} = 0.08$m

$X_i = 1.6$m

$V = 35$ m/sec

FIG. 9 TIME RESPONSE - BASIC A/C

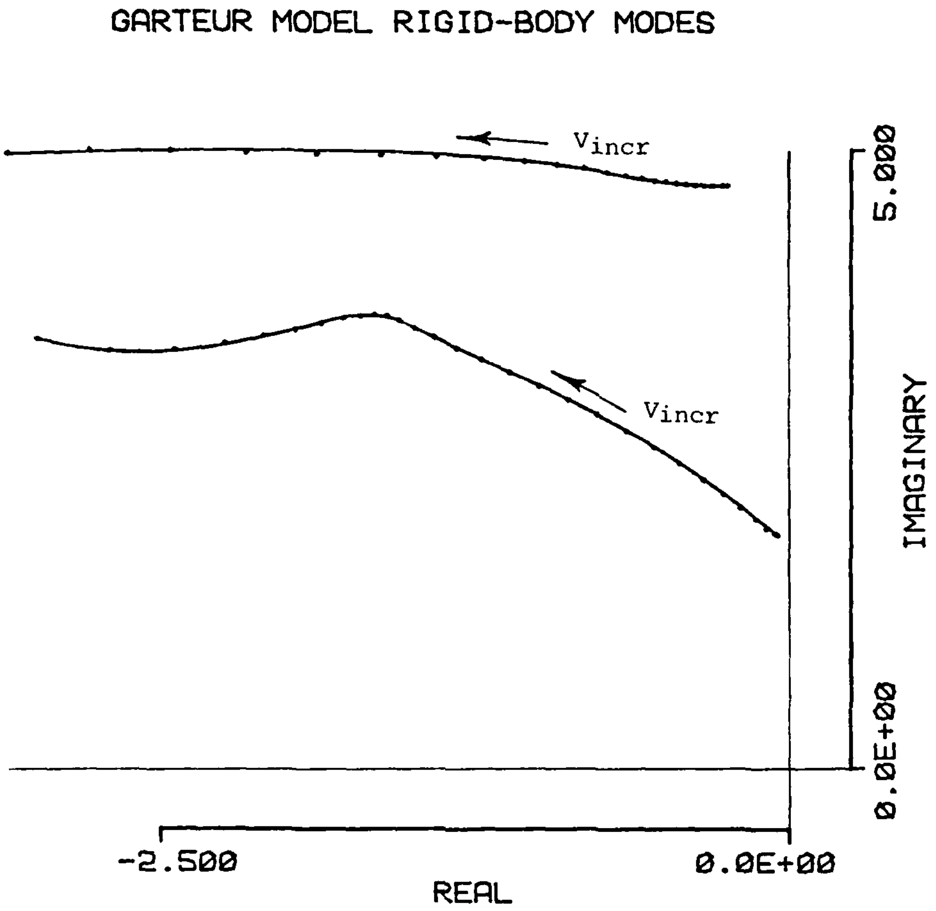


X_{SPR} = 0.09m

X_G = 1.6m

V = 35 m/sec

FIG. 10 TIME RESPONSE - BASIC A/C



CONTROL LAW:

$\delta = 0.4q + 1.0\theta + 0.2Z_G$

X_{SPR} = 0.07m, X_G = 1.6m

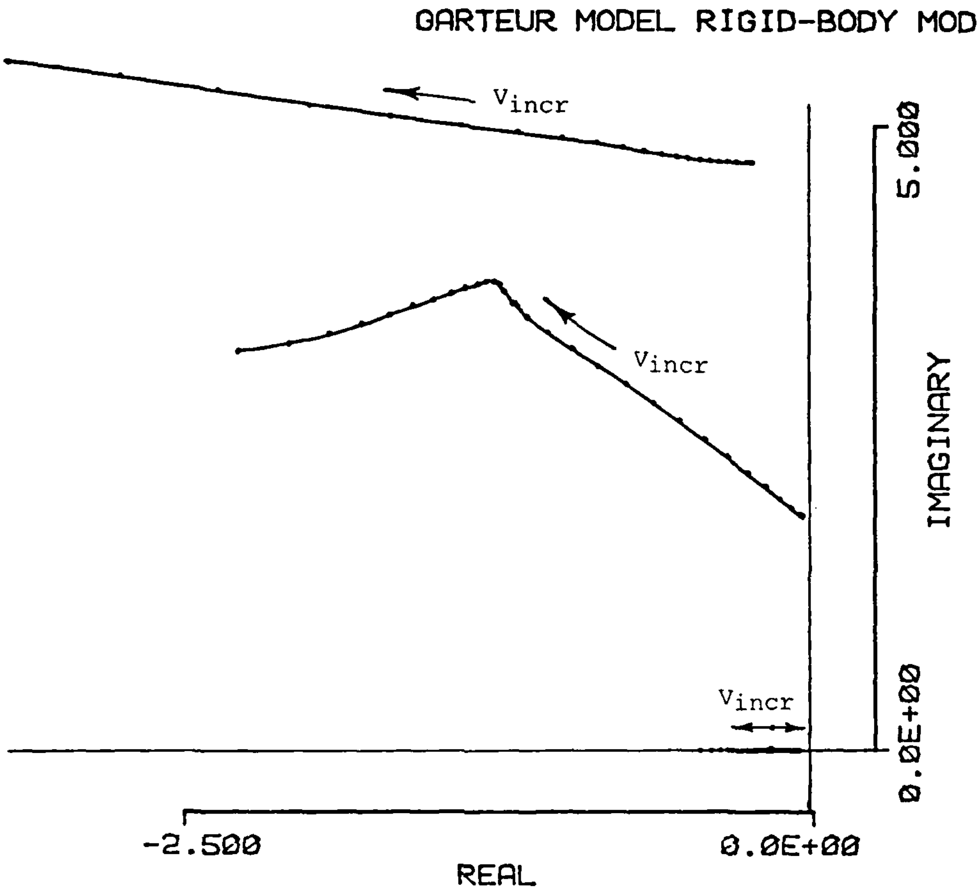
K1 = 0.4, K2 = 1.0

K3 = 0.2, K4 = 0.0

TAUACT = 0.01 sec, KACT = 1.0

$0.01 < V < 60$ in steps of 2.0 m/sec

FIG. 11(A) ROOT LOCUS - AIRCRAFT WITH ACTUAL RBM STATE FEEDBACK (DOMINANT ROOTS)



CONTROL LAW:

$$\hat{\delta} = 0.4\hat{q} + 1.0\hat{\theta} + 0.2\ \hat{z}_G$$

X_{SPR} = 0.07m, X_G = 1.6m

K1 = 0.4 K2 = 1.0

K3 = 0.2 K4 = 0.0 $0.01 < V < 60.0$ in steps of 2.0 m/sec

TAUACT = 0.01 KACT = 1.0

TAUNF = 0.03 TAUINT = 6.0

FIG. 11(B) ROOT LOCUS - AIRCRAFT WITH ESTIMATED RBM STATE FEEDBACK (DOMINANT ROOTS)

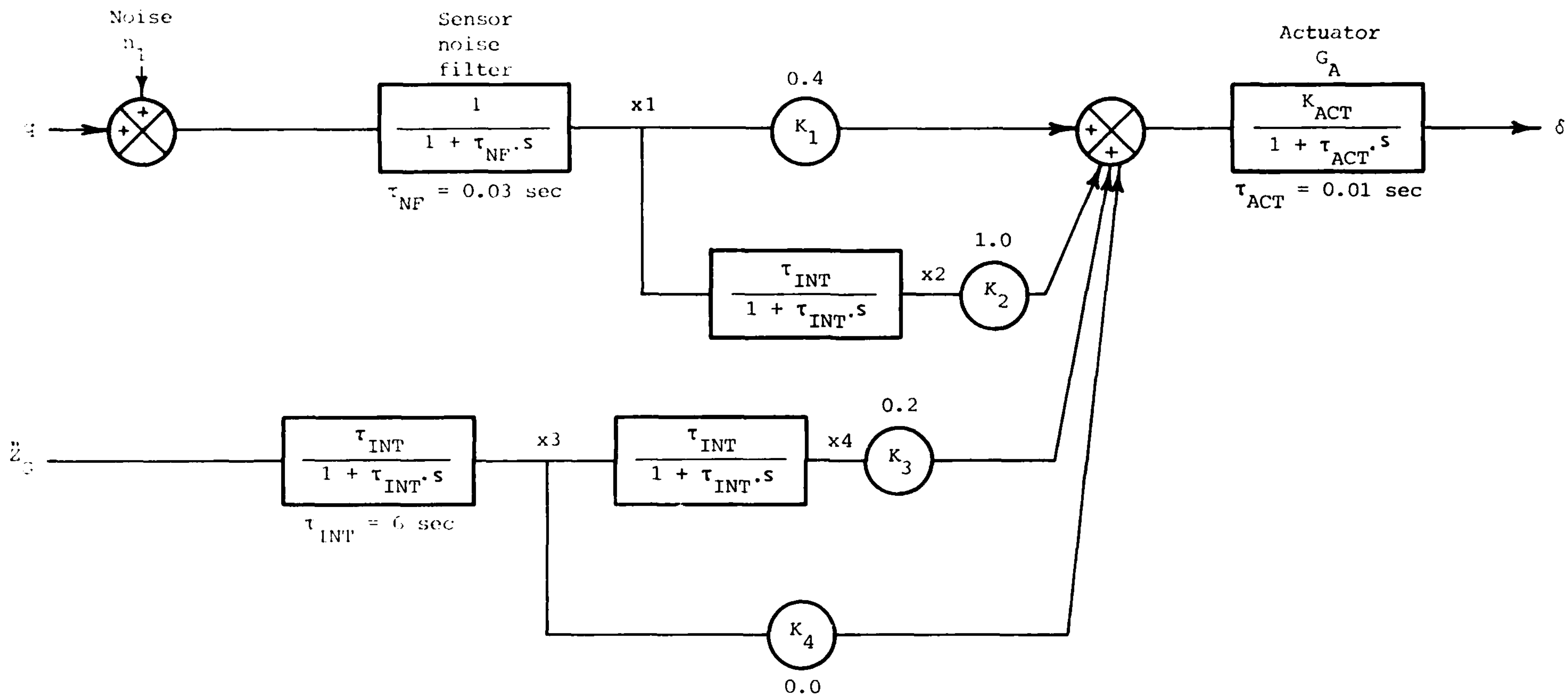
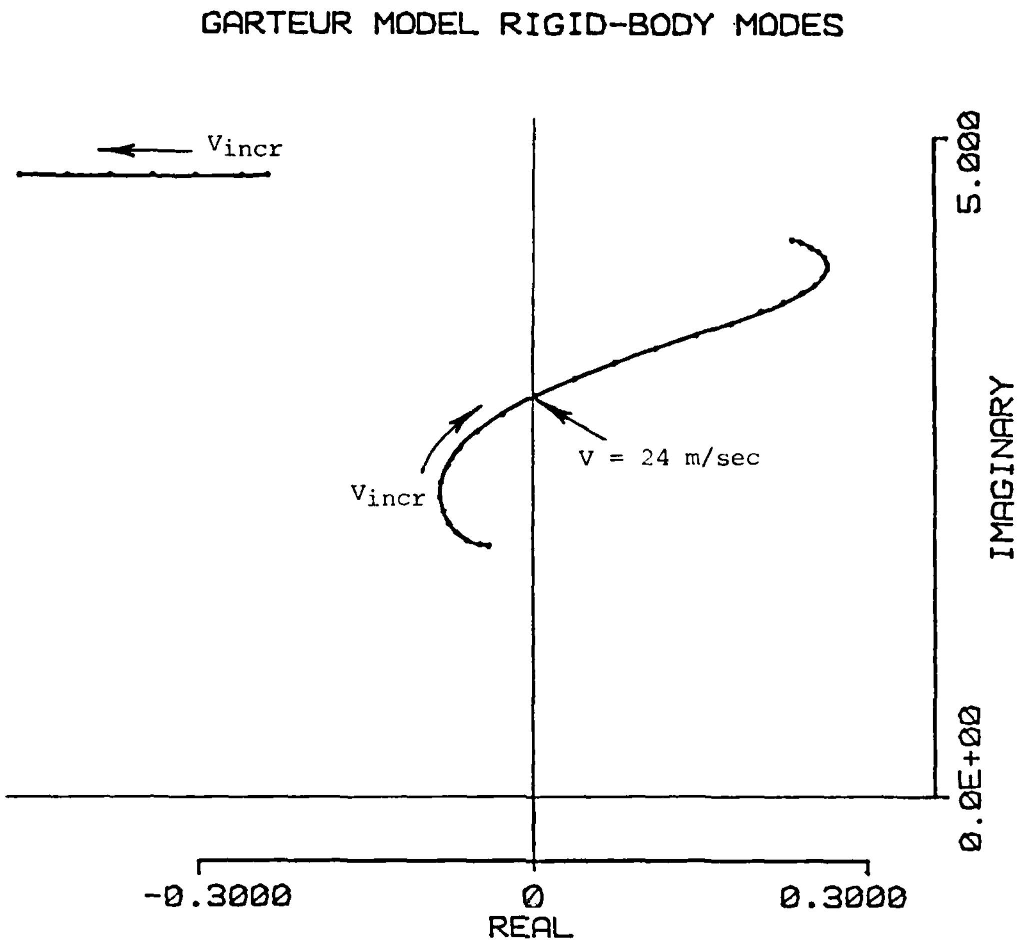


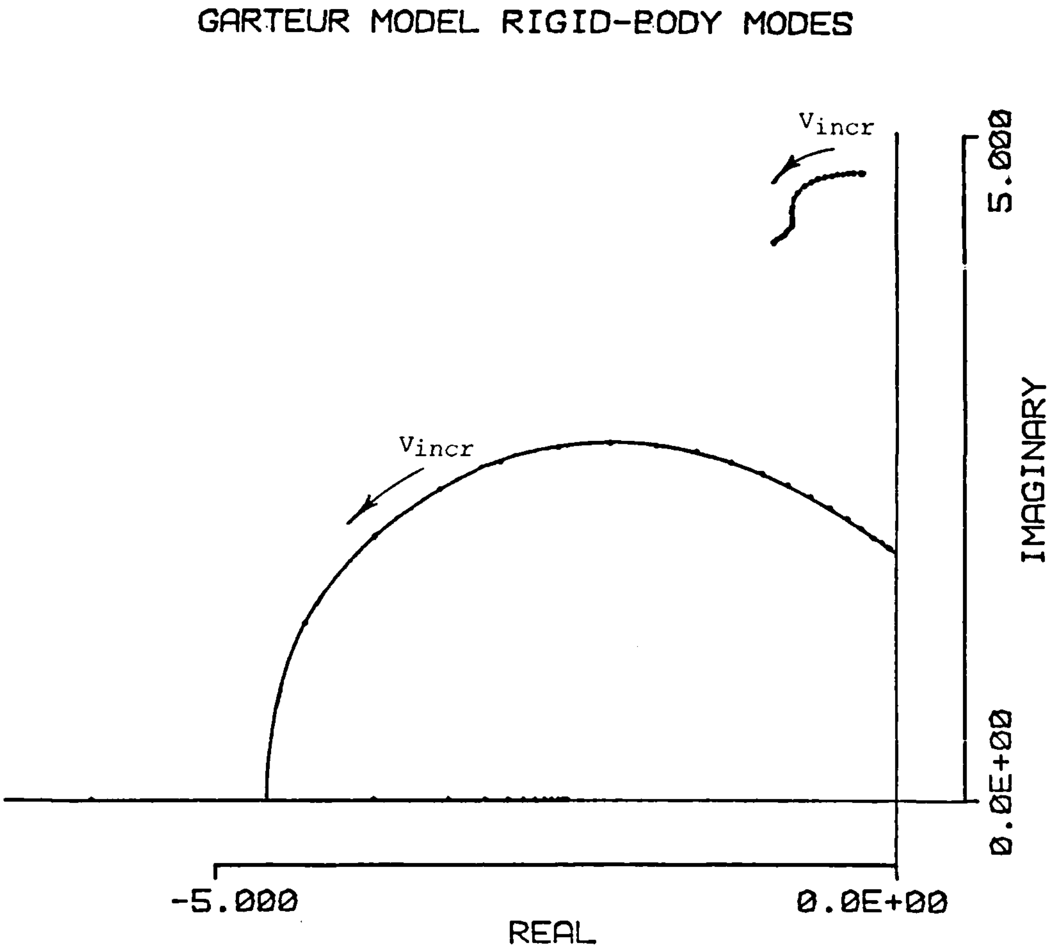
FIG. 12 CONTROL LAW BLOCK DIAGRAM



$X_{SPR} = 0.007$m; $X_G = 1.543$m; $0.01 \leq V \leq 60.0$ in steps of 2m/sec

$M = 62.8$ kg; $I_Y = 21.2$ kg.m²

FIG. 13 BASIC A/C ROOT LOCUS (MODIFIED AIRCRAFT)



CONTROL LAW $\delta = 0.5\ q$

$X_{SPR} = 0.007m$; $X_G = 1.543m$;

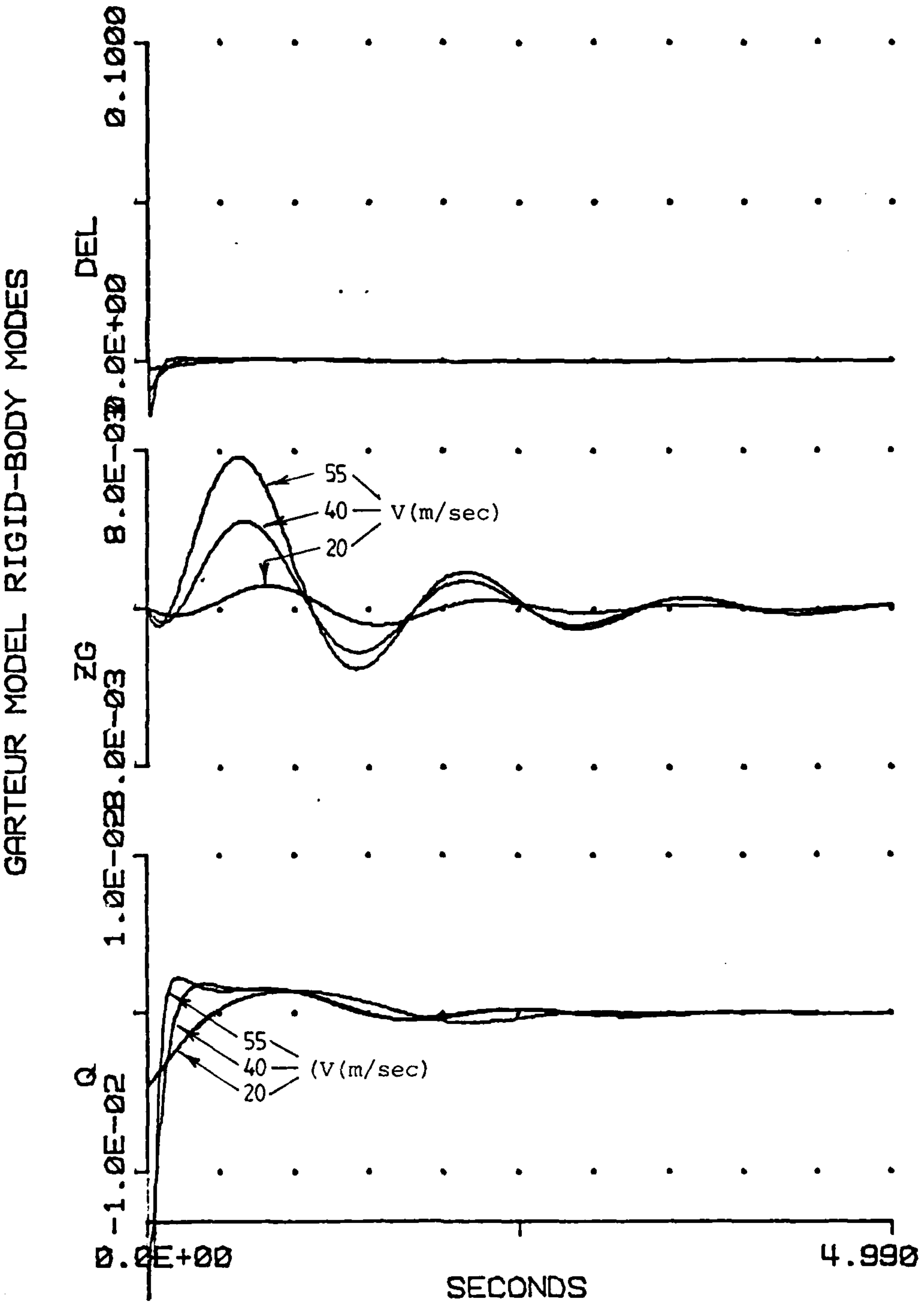
$K_1 = 0.5$

$K_2 = K_3 = K_4 = 0$

$M = 62.8$ kg; $I_Y = 21.2$ kg.m²

$0.01 \leq V \leq 60.0$ in steps of 2 m/sec

FIG. 14 MODIFIED AIRCRAFT WITH 'q' FEEDBACK ONLY



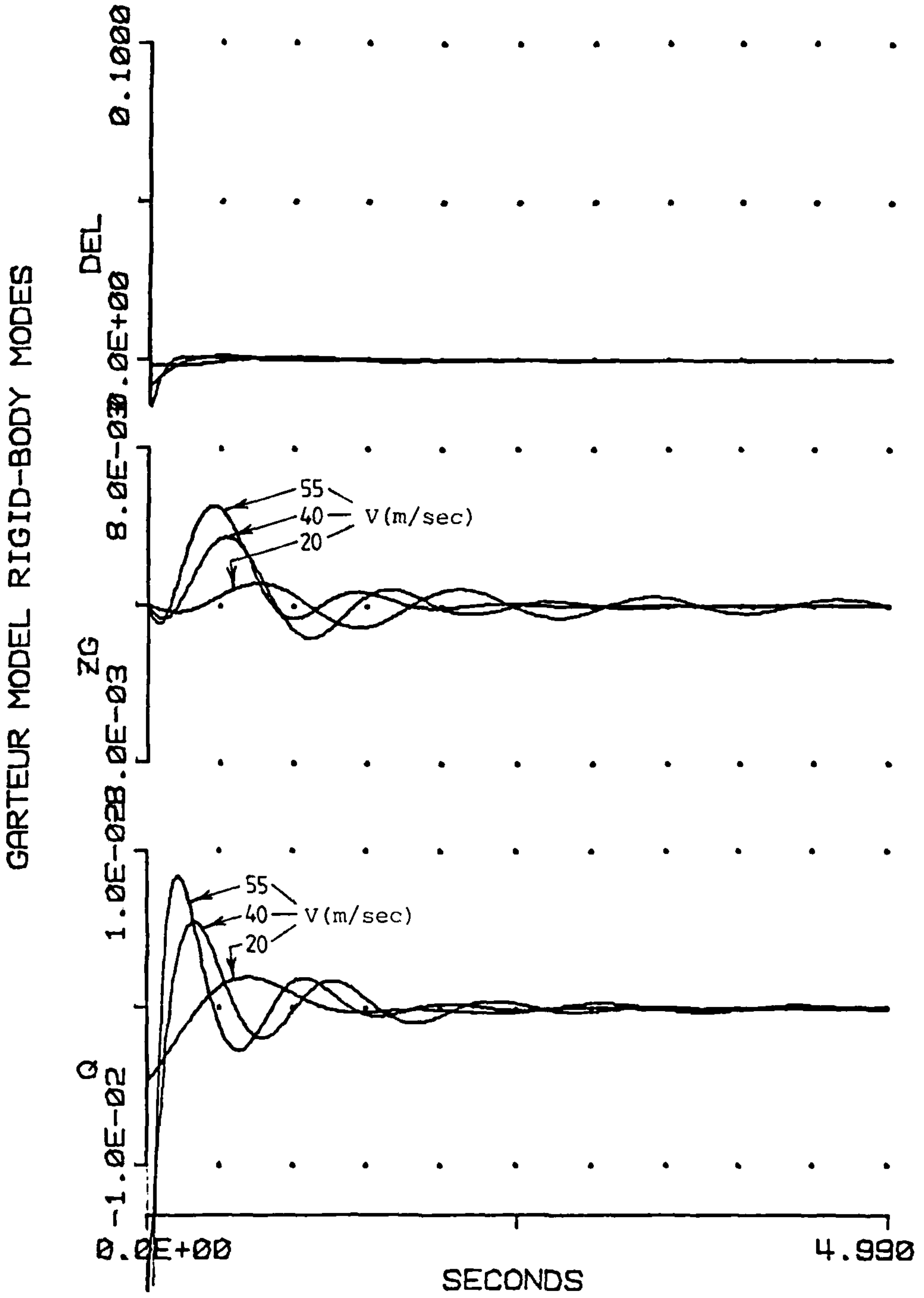
$K_1 = 0.5$

$K_2 = K_3 = K_4 = 0$

$X_{SPR} = 0.007\text{m}$; $X_G = 1.543\text{m}$

$M = 62.8$ kg; $I_Y = 21.2$ kg.m²

FIG. 15 TIME RESPONSE - MODIFIED AIRCRAFT WITH q FEEDBACK ONLY



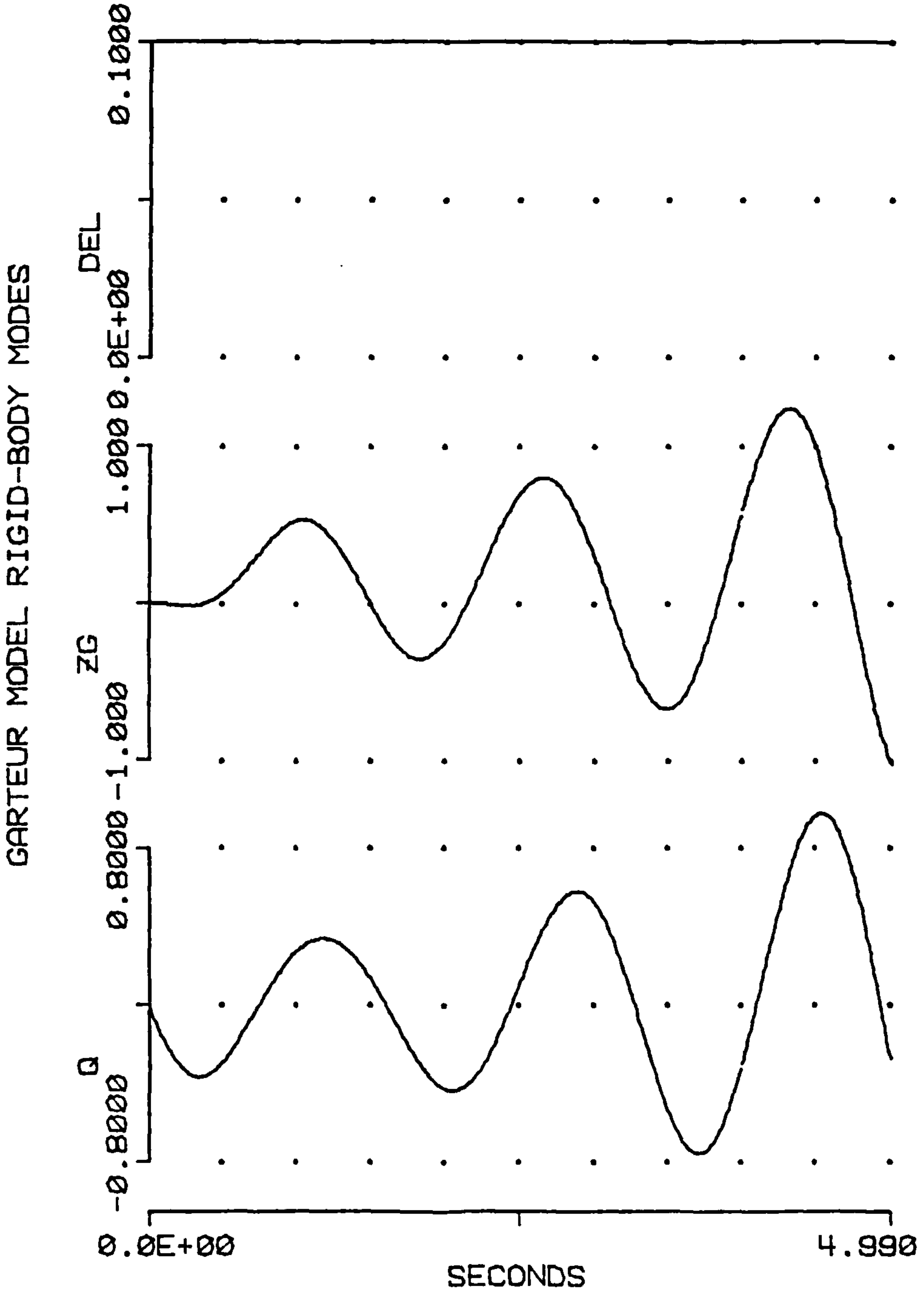
$K_1 = 0.5$; $K_2 = 1.0$

$K_3 = 0.1$; $K_4 = -0.1$

$X_{SPR} = 0.007m$; $X_G = 1.543m$

$M = 62.8$ kg; $I_Y = 21.2$ kg.m^2

FIG. 16 TIME RESPONSE - MODIFIED AIRCRAFT WITH ACTUAL RBM STATE FEEDBACK



$K_1 = K_2 = K_3 = K_4 = 0$

$X_{SPR} = 0.007m; \quad X_G = 1.543m$

$M = 62.8$ kg; $I_Y = 21.2$ kg.m²

FIG. 17 TIME RESPONSE, MODIFIED 'BASIC AIRCRAFT' FOR V = 35 m/sec

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16. Abstract

This report describes a simulation study undertaken by the author while attached to the RAE, Farnborough UK on a STUDS visit. The study was made in preparation for wind tunnel trials of flutter suppression control laws at DFVLR, Göttingen, West Germany.

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