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by

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LOW SPEED PULL-UP MANOEUVRES FOR A SLENDER WING TRANSPORT AIRCRAFT WITH STABILITY AND CONTROL AUGMENTATION

by

Dorothy M. Holford

SUMMARY

Low speed pull-up manoeuvres for a slender wing transport aircraft are calculated. Two extremes of aircraft weight are considered, 385 000 lb and 180 000 lb. For each aircraft weight, two CG positions are considered. Stability augmentation, in the form of angle-of-incidence and/or rate-of-pitch feedback, and control augmentation are investigated as a means of improving the response of the aircraft in pull-up manoeuvres.

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* Replaces RAE Technical Report 70194 - ARC 33169

CONTENTS

1 INTRODUCTION

The purpose of this paper is to investigate the low speed pull-up manoeuvre for a slender wing transport aircraft of which the general arrangement is shown in Fig.1. In the pull-up manoeuvre the pilot operates the controls so as to achieve a rapid gain in height followed finally by a steady climb at about 1 g. The incremental normal acceleration reached during the manoeuvre must not be excessive.

Early work by Czaykowski at RAE showed that the response of this type of aircraft to elevator movement is such that after a sluggish initial behaviour, the response could build up rapidly and excessively high normal acceleration and incidence could be achieved. The existence of sluggish initial response means that the aircraft is also slow to respond to any corrective elevator application. Thus the pilot must apply corrective elevator before he would normally recognise its necessity. Stability augmentation was suggested by Czaykowski as a possible means of improving the situation.

Stability augmentation was found to be very beneficial in American tests of supersonic transport handling qualities using an in-flight simulator[1]; the use of pitch-rate and angle-of-incidence feedback in conjunction with increased elevator-to-column gearing reduced the Cooper pilot ratings from 5.4, for the unaugmented aircraft, to 2.9 in low-speed longitudinal manoeuvres. (This paper considers pitch-rate and/or angle-of-incidence feedback.)

'Manoeuvre boost' is considered here as a form of control augmentation. There is a limit to the amount of boosting that can take place, because there is an overall maximum rate of elevator movement. A noteworthy feature of a manoeuvre boost system is that it reduces the amount of checking required from the pilot by providing some checking elevator movement when he returns the control to the trim condition.

Due to adverse elevator lift, the initial height response is in the opposite direction to that actually required. One measure of the delay in response is the time taken to regain original height, $t_{h=0}$. Pinsker[2] discussed the effect of pitch damping and manoeuvre boosting on this time, and found that both these augmentation systems gave a small improvement in $t_{h=0}$; however, the height loss during this time was increased. We find here that in the pull-up manoeuvre the elevator time-history which produces a very short $t_{h=0}$ and minimum height loss during this time does not necessarily produce a good climb performance.

2 MATHEMATICAL MODEL

The representation of the aerodynamic characteristics of the aircraft is given in section 2.1. The computation of the pull-up manoeuvre comprises two parts:

(a) determination of the initial conditions (section 2.2), and

(b) calculation of the response in the manoeuvre, referred to these initial conditions (section 2.3).

All computations were performed using an ICL 1907 digital computer.

2.1 Representation of the aerodynamic characteristics

The following expressions were taken to represent the dependence of the aerodynamic force coefficients C_L and C_D on angle of incidence, α, and elevator angle, η:

$$C_L = A_1 \alpha + A_2 \eta + A_3$$

and

$$C_D = B_1 \alpha^2 + B_2 \alpha + B_3 \alpha \eta + B_4 \eta + B_5$$

where the A's and B's are constants.

The pitch moment coefficient, C_m, about a reference point is given by

$$C_m = C_1 \alpha^2 + C_2 \alpha + C_3 \alpha \eta + C_4 \eta + C_5$$

where the C's are constants. The pitch moment coefficient about a point a fraction b of c_o ahead of the reference point is given by

$$C_m = C_1 \alpha^2 + C_2 \alpha + C_3 \alpha \eta + C_4 \eta + C_5 + b\,(- C_L \cos \alpha - C_D \sin \alpha) .$$

2.2 Initial conditions

The initial motion of the aircraft is 1 g steady level flight as a given speed V_e. Referring the motion of the aircraft to flight path axes with the origin at the centre of gravity and denoting equilibrium values by a subscript e gives

$$\left.\begin{aligned} B\frac{dq}{dt} &= 0 = \tfrac{1}{2}\rho V_e^2 S c_o C_{m_e} + T_e d \\ m\frac{dw}{dt} &= 0 = -\tfrac{1}{2}\rho V_e^2 S C_{L_e} - T_e \sin(\alpha_e + \vartheta) + W \\ m\frac{du}{dt} &= 0 = -\tfrac{1}{2}\rho V_e^2 S C_{D_e} + T_e \cos(\alpha_e + \vartheta) \end{aligned}\right\} \qquad (1)$$

where ϑ is the inclination of the thrust axis to the body datum and d is the thrust moment arm about the CG of the aircraft. d is given by

$$d = d_o - b\, c_o \sin\vartheta$$

where d_o is the corresponding moment arm about the reference point and b is the same as in section 2.1.

The set of equations (1) are solved simultaneously for α_e, η_e and T_e by a generalised form of the Newton-Raphson iterative method.

2.3 Equations of motion

The equations of longitudinal motion for the rigid aircraft are referred to aerodynamic body axes, which in the datum condition coincide with the flight path axes of section 2.2.

We have

$$m\frac{du}{dt} = -mg\sin\theta - mwq + \tfrac{1}{2}\rho S(V_e + u)^2\left(C_L\frac{w}{V_e} - C_D\right) + \tfrac{1}{2}\rho V_e^2 S C_{D_e} + T\cos(\alpha_e + \vartheta)$$

$$m\frac{dw}{dt} = mg\cos\theta - mg + mq(V_e + u) - \tfrac{1}{2}\rho S(V_e + u)^2\left(C_L + C_D\frac{w}{V_e}\right) + \tfrac{1}{2}\rho S V_e^2 C_{L_e} - T\sin(\alpha_e + \vartheta)$$

$$m k_B^2\frac{dq}{dt} = \tfrac{1}{2}\rho S c_o (V_e + u)^2\left[C_m + \frac{\partial C_m}{\partial \dot{w}}\frac{dw}{dt} + \frac{\partial C_m}{\partial q} q\right] - \tfrac{1}{2}\rho S c_o V_e^2 C_{m_e} + T d$$

$$\frac{d\theta}{dt} = q$$

$$\frac{dh}{dt} = (V_e + u)\sin\theta - w\cos\theta$$

$$\frac{dR}{dt} = (V_e + u)\cos\theta + w\sin\theta\ .$$

In the datum condition $u = w = q = \theta = h = R = 0$.

At the start of the manoeuvre an incremental thrust T_o may be demanded: the applied incremental thrust T is represented by an exponential rise to this value, i.e.

$$T = T_o (1 - e^{-kt}) \quad .$$

In these equations terms of the second order in u/V_e and w/V_e have been neglected so that the forward speed V is $(V_e + u)$ and the incremental angle of incidence, arctan (w/V), is approximated by w/V_e. Also cos (w/V_e) is taken as unity and sin (w/V_e) as w/V_e. C_L, C_D and C_m are functions of total angle of incidence and elevator angle and C_m is adjusted for the CG position under consideration as in section 2.1. The equations given above were non-dimensionalised for the purposes of computation.

3 CONTROL SURFACE MOVEMENT

The movement of the elevator control surface is assumed to be the algebraic sum of autostabiliser output and pilot-induced movement.

$$\eta = \eta_A + \eta_C \quad .$$

No attempt is made to incorporate the dynamics of the power control, which is assumed to be capable of moving the elevator at rates up to about 40^o/sec.

3.1 Stability augmentation

The autostabiliser produces an elevator deflection which is a function of angle of incidence and/or rate of pitch.

$$\eta_A = \eta_\alpha + \eta_q \quad .$$

The type and position of the sensors is not considered; the response quantities used are assumed to be available. η_A is not limited.

The law governing η_α is

$$\eta_\alpha = \frac{D}{k_\alpha + D} G_\alpha \alpha \; ,$$

where G_α and k_α are constants and D is the differential operator. This control law has the effect of a high-pass filter so that at low frequencies feedback is suppressed.

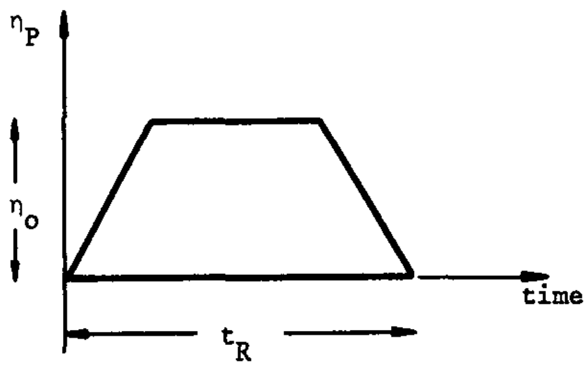
The law governing η_q is

$$\eta_q = \left(\frac{K_q + D}{k_q + D}\right) G_q q$$

where G_q, K_q and k_q are constants. When $K_q = 0$ the law is reduced to the same form as that for η_α and is that of a washed-out pitch damper. For $K_q = k_q$ the law governing η_q is that of simple pitch damper. For $K_q > k_q$ a stabilising component is added to the simple pitch damper and conversely.

3.2 Pilot's demands and control augmentation

The general form of the pilot's demand η_P is shown below



The maximum value of $\frac{d\eta_P}{dt}$ is taken to be 40°/sec. Thus for the unaugmented aircraft the maximum rate of control surface movement is 40°/sec.

The pilot's demand η_P is passed through a 'manoeuvre boost' system or 'stick filter' having a law of the form

$$\eta_C = \frac{1 + KD}{1 + D} \eta_P$$

where K is a constant. If the rate of pilot's demand $(d\eta_P/dt)$ changes by a certain amount, the instantaneous change in the rate of output of the stick filter $(d\eta_C/dt)$ is K times that amount.

In the absence of control augmentation

$$\eta_C = \eta_P .$$

4 AERODYNAMIC DATA

Wind tunnel data for C_m, C_L and C_D were fitted to the forms of section 2.1 and the numerical values of the various coefficients obtained are given in the Appendix. These data apply for the most part to the aircraft in an approach configuration with the nose drooped 17.5° and the undercarriage down. The reference CG position is 50% c_o. A comparison between the wind tunnel data and the fitted curves is shown in Figs.2,3 and 4. The representation was considered very good over the range of incidence and elevator angle that is of interest here.

Wind tunnel results ($\eta = 0$) for the aircraft configuration with the nose drooped 5° and the undercarriage up are also shown in Figs.2,3 and 4. There were no data for the elevator power in this configuration, and so the results quoted in this paper are for the approach configuration.

For positive elevator angles the wind tunnel results show that violent pitch-up occurs at about 24° angle of incidence (not shown in Fig.2). The tendency is just noticeable at 25° angle of incidence and zero elevator angle as shown in Fig.2. For negative elevator angles pitch-up occurs less violently at about 25° angle of incidence. No attempt was made to simulate these 'pitch-up' characteristics, and so if during the manoeuvre the angle of incidence exceeds about 25° the calculation becomes unrepresentative of the aircraft.

Fig.5 shows the variation of C_m with angle of incidence and CG position. The reference CG position of 50% c_o is included for completeness. For a CG position 53.5% c_o the slope of the curve is in the unstable sense for the range of α considered, while for a CG position of 51.5% c_o the slope is in the stable sense up to about $\alpha = 15°$.

5 CALCULATION OF THE PULL-UP MANOEUVRE

The fitted curves of C_m, C_L and C_D as given in the Appendix were used to calculate the 1 g trim conditions C_{L_e}, α_e, η_e and T_e, by the method of section 2.2, for various combinations of forward speed and aircraft weight. The results for C_{L_e}, α_e and η_e are shown graphically in Figs.6,7 and 8.

In the following response calculations two extremes of aircraft weight, 180 000 lb and 385 000 lb, each in association with two CG positions, 51.5% c_o and 53.5% c_o, are considered. The quoted results are for a trimmed forward speed of 200 knots: the trim conditions are given in Table 1. The maximum thrust available is assumed to be about 120 000 lb and reference to Table 1 shows that, after trimming, the amounts of incremental thrust available for aircraft weights of 385 000 lb and 180 000 lb are 25000 lb and 85000 lb respectively. Other relevant aircraft data are given in the Appendix.

The response of the aircraft was calculated with and without stability augmentation and the results are discussed below. A brief summary of the results obtained is given in Table 2 (W = 385 000 lb) and Table 3 (W = 180 000 lb).

5.1 The unaugmented aircraft

The response of the heavier aircraft (W = 385 000 lb) to a pilot elevator input of -2° is shown in Fig.9. The response of the aircraft with CG position 53.5% c_o is greater than that for one with CG position at 51.5% c_o. At trimmed incidence, the slope of the C_m v α curve is positive (i.e. in the unstable sense) for a CG position of 53.5% c_o while it is almost zero for one of 51.5% c_o, becoming positive at a slightly higher incidence. The aircraft is initially sluggish in response to the elevator. After some 2 seconds the response builds up very quickly. The initial height loss, due to adverse elevator lift, is small, being of the order of ½ ft, but approximately 1.7 seconds elapse from the start of the manoeuvre before the aircraft regains its original height. Removal of the elevator is not sufficient to check the manoeuvre. In terms of height gained the response is poor - approximately 55 ft after 5 seconds for the aircraft with CG at 53.5% c_o.

The response of the lighter aircraft (W = 180 000 lb) to a pilot elevator input of -1° is shown in Fig.10. The responses for the two CG positions 51.5% c_o and 53.5% c_o are different in character and reference to Fig.5 shows that for the aft CG position, 53.5% c_o, the slope of the C_m v α curve for the trimmed incidence of 8.05° is in the unstable sense whilst for the forward CG position, 51.5% c_o, and trimmed incidence of 8.44°, the slope is in the stable sense. For a CG position of 51.5% c_o, the removal of the elevator is sufficient to check the manoeuvre provided that the incidence reached during the application of the elevator is not too high. For either CG position the height loss is negligible and $t_{h=0}$ is 1.15 seconds.

In general the response of this lighter aircraft is much crisper than that of its heavier counterpart.

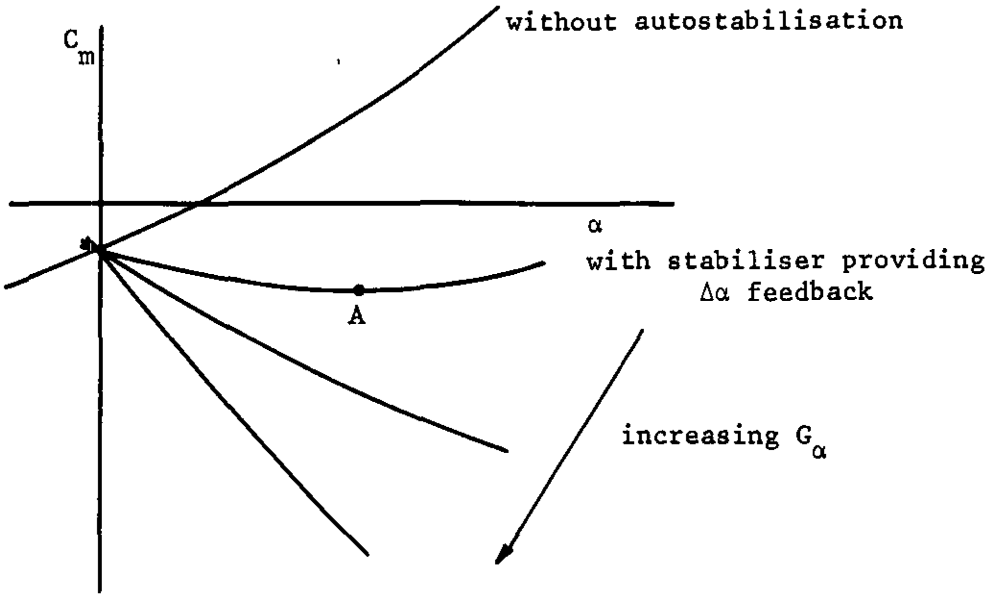
5.2 Effects of autostabilisation

If an elevator input is supplied by the pilot to the aircraft with some autostabilisation then the surface movement will not be the same as that demanded by the pilot. When the pilot's input of section 5.1 is applied then most of the initial ramp part of this input is transmitted to the control surface, but then as the pilot holds η_p constant less and less of this demand is actually applied until eventually opposite elevator may be applied at the surface: in such a case the time at which η becomes zero is denoted by t_z. The rapidity with which this corrective elevator angle is applied depends on the type of autostabilisation present - α and/or q feedback - and the magnitude of the gains G_α, G_q and the constants k_α, k_q and K_q (see section 3.1). It can be inferred that, because the pilot's input is reduced by the action of the autostabiliser, in order to pull the same maximum g during the manoeuvre the pilot's input for the augmented aircraft must be greater than that for the unaugmented aircraft.

5.2.1 Response of the aircraft with stability augmentation (W = 385 000 lb)

Fig.11 shows the responses of the aircraft (CG position 53.5% c_o) with an autostabiliser providing respectively α feedback, q feedback, and α and q feedback together - in the last two cases $K_q = 0$. When the difference in maximum normal acceleration reached during the manoeuvre is taken into account, there is very little difference in the climb performances in the three cases shown. The height response is better than that of the unaugmented aircraft but $t_{h=0}$ is only slightly reduced. The maximum normal acceleration reached during the manoeuvre, though reduced, is now reached much earlier and hence the distances to incremental altitudes of 35 ft and 50 ft are much reduced (see Table 2). The attitude θ reached during the manoeuvre is still large and increases more rapidly during later stages of the manoeuvre (not shown in the figure). Similar results are obtained for the aircraft with the CG position at 51.5% c_o.

Although autostabilisation, and in particular α-feedback, improves the performance of the aircraft, a closer inspection of Fig.11 reveals an undesirable feature. From the trace of incremental angle of incidence, it can be seen that α first increases quite sharply and then flattens and finally starts to increase again. The picture is made more complicated by the time constant k_α. The changes in slope of the α time history can be understood by considering the behaviour of the autostabiliser in the unpractical case where the time constant, k_α, is zero. If an elevator angle is held constant by the pilot of the unaugmented aircraft, then a possible C_m v α curve is shown below.



Suppose now that the pilot applies the same elevator angle to an aircraft with an autostabiliser providing α feedback; then a change in α would cause a corresponding change in elevator angle. Thus the C_m v α curve for the augmented aircraft is one where the elevator angle varies along it. The slope of the C_m v α curve for the augmented aircraft depends on the value of G_α, the modulus of the slope increasing as G_α increases. The slope is in the stable sense, throughout the range of α, for a high enough value of G_α. For low values of G_α, it can be seen that the slope of the C_m v α curve changes sign and if, during the manoeuvre, α exceeds that at point A the aircraft becomes unstable. Introduction of the time constant k_α reduces the amount of additional stability provided by the autostabiliser and there is an increase in the value of G_α at which this change of sign in the slope of the C_m v α curve occurs.

The final divergent nature of the α time-history of Fig.11 can be eliminated, and a good climb performance produced, by increasing G_α; however this results in a very high authority for the autostabiliser and a need for large control demands by the pilot.

The large values of θ obtained in these manoeuvres show the importance of the 'position' term K_q in the law for η_q (section 3.1). The effect of incorporating K_q in the control law for η_q can be seen by comparing the solid lines of Figs.11 and 12. The value of K_q in Fig.12 is 1.25. The peak normal acceleration, for the same pilot elevator input, is reduced with the introduction of K_q:- 1.45 g for $K_q = 1.25$ compared with 1.55 g for $K_q = 0$. Also the normal acceleration returns to about 1 g some 2 seconds after the removal of the pilot's elevator angle. After 5 seconds the height gained is therefore less for $K_q = 1.25$ than for $K_q = 0$; however, for the former value the aircraft has already settled into a fairly steady 3° climb. For $K_q = 1.25$ the drop in forward speed after 10 seconds is only 15 knots.

Reference to Fig.12 also shows that with K_q included in the control law for η_q, α-feedback may be dispensed with. (The assumed pilot's control deflection is reduced when α feedback is omitted in order that the peak normal accelerations shall be similar for the two cases.)

The value of K_q in Fig.12 may well be too high. A case similar to that presented as the solid line of Fig.12 but with $K_q = 0.8$ results in a peak normal acceleration of 1.48 g and a steeper final climb path of 4½°. The speed loss during the manoeuvre (20 knots after 10 seconds) is greater than that with $K_q = 1.25$ (see Table 2). The allowable climb angle depends on the amount of thrust available to maintain forward speed. Application of thrust in itself steepens the final climb path.

Fig.13 shows the response of the aircraft with an autostabiliser providing α and q feedback ($K_q = 1.25$) and incremental thrust ($T_o = 25000$ lb and $k = 0.5$) applied at the start of the manoeuvre according to the law of section 2.3. Two CG positions, 53.5% c_o and 51.5% c_o are considered and the pilot's elevator input has been adjusted to give a peak normal acceleration of about 1.6 g for both cases. The outcome of the manoeuvre shown in Fig.13 is a 5° climb, with a speed loss of 12 knots after 10 seconds compared with a 3½° climb, with a speed loss of 20 knots after 10 seconds without incremental thrust.

5.2.2 Response of the aircraft with stability augmentation (W = 180 000 lb)

Fig.14 shows the response of the aircraft with an autostabiliser providing α and q feedback ($K_q = 0$) for the two CG positions 51.5% c_o and 53.5% c_o. The behaviour is much the same for the two CG positions. $t_{h=0}$ is slightly reduced by the introduction of the autostabilisation. The position term K_q in the control law for η_q is again introduced and results for $K_q = 0.6$ are shown in Fig.15 for the two CG positions. The pilot's elevator input has been adjusted so that the peak normal acceleration reached during the manoeuvre is about 1.6 g for both CG positions. The aircraft settles into a 5^o climb after some 4 seconds. The required value of K_q for this aircraft is much smaller than for the heavier aircraft and a value in the range 0.4 to 0.6 would appear to be sufficient.

The effect of incremental thrust (T_o = 40000 lb, k = 0.5) applied at the start of the manoeuvre on the response of the aircraft with CG position 53.5% c_o is also shown in Fig.15. The peak normal acceleration is increased and approximately 1.1 g is pulled during the climb. The forward speed increases during the manoeuvre.

5.3 Control augmentation

Control augmentation or manoeuvre boost modifies the pilot's elevator demand in order to improve the aircraft's handling characteristics. It does not eliminate the need for stability augmentation though the provision of better controllability may lessen that need.

The first two time-histories of Fig.16 show the effect of control augmentation of the type discussed in section 3.2, with K = 2, on a particular pilot elevator input. The rate of elevator movement demanded by the pilot here is 20^o/sec so that the demanded rate of elevator movement 'downstream' of the control augmentation system is initially about 40^o/sec.

The remainder of Fig.16 shows the responses of the heavy aircraft with the CG position at 53.5% c_o. The autostabiliser provides both α and q feedback; two values of K_q are shown, 0 and 0.4. It can be seen that there is now less need for the position term since it would appear that the value of 0.4 is if anything too high, in contrast to the result that a value of about 0.8 was necessary in the absence of control augmentation. This can be attributed to the 'checking' action of the control augmentation when the pilot cancels his elevator demand.

Despite the action of control augmentation in making the aircraft's response crisper, the climb performance is only marginally better than that obtained previously. For the full benefit of control augmentation to be felt it is necessary to have a high rate of control movement available.

It is found that for the light aircraft, with control augmentation, acceptable characteristics are produced with an autostabiliser providing α and q feedback with the position term, K_q, zero.

6 CONCLUSIONS

If augmented at a weight of 385 000 lb the ~~unaugmented~~ slender wing transport aircraft would be statically unstable in level flight at low forward speeds and would be initially sluggish in response to the elevator. Some form of stability augmentation is necessary for long-term operation of the aircraft. Inclusion of a stability augmentation system comprising angle-of-incidence feedback and a pitch damper makes the aircraft statically stable over a part or the whole incidence range depending upon the gearing associated with the autostabiliser and the amount of control surface movement available. The response to elevator is improved but the pilot would still have to apply corrective elevator to produce a steady climb. Introduction of a pitch 'position' term in the autostabiliser further improves the situation and a steady climb is achieved after some 5 seconds with little or no corrective pilot activity. The height lost due to adverse elevator lift is small, about 1 to 1½ ft, and the time taken to regain the original height is of the order of 1.5 seconds.

Control augmentation may be used to improve the aircraft's response to pilot's control movements: its use does not eliminate the need for stability augmentation but the position term in the pitch autostabiliser law is then not so important.

The response to elevator of the aircraft of weight 180 000 lb is much crisper than that of its heavier counterpart. Stability augmentation is certainly necessary for a CG position of 53.5% c_o and is desirable for a CG position of 51.5% c_o since the aircraft is statically unstable above about 16^o of incidence. The height lost is of the order of ½ ft and the time taken to regain original height is about 1 second. For this aircraft the position term in the pitch autostabiliser is not as important as for the heavy aircraft. This term becomes relatively unimportant when control augmentation is employed.

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ERRATUM

The first sentence of section 6, page 14 should read:-

If unaugmented, at a weight of 385 000 lb the slender wing transport aircraft would be statically unstable in level flight at low forward speeds and would be initially sluggish in response to the elevator.

London, Her Majesty's Stationery Office, November 1972.

Appendix

DATA USED IN THE CALCULATIONS

General

Reference wing areas, $S = 3856$ sq ft

Reference wing chord, $c_o = 90.75$ ft

Radius of gyration in pitch, $k_B = 29.5$ ft

Moment arm of thrust contribution to pitching moment about CG at 50% c_o (reference CG position), $d_o = 2.26$ ft

Inclination of thrust axis to body datum, $\vartheta = 0.96^o$ nose up.

Aerodynamic

At the reference CG position of 50% c_o

$$C_L = 0.05866\ \alpha + 0.01288\ \eta - 0.14666$$

$$C_D = 0.001183\ \alpha^2 - 0.008355\ \alpha + 0.0001835\ \alpha\ \eta - 0.000069\ \eta + 0.054894$$

$$C_m = 0.00004114\ \alpha^2 - 0.0022067\ \alpha + 0.00001088\ \alpha\ \eta - 0.0040847\ \eta + 0.0041036$$

where α, η are in degrees.

$$m_{\dot{w}} = \frac{1}{2} \frac{\partial C_m}{\partial\left(\frac{\dot{w} c_o}{V_e^2}\right)} = -0.04$$

$$m_q = \frac{1}{2} \frac{\partial C_m}{\partial\left(\frac{q c_o}{V_e}\right)} = -0.08$$

Table 1

1 g TRIM CONDITIONS FOR THE AIRCRAFT FLYING AT 200 KNOTS FORWARD SPEED

Weight	CG position (% c_o)	α_e (deg)	η_e (deg)	T_e (lb)
385 000	53.5	13.68	2.77	91300
385 000	51.5	14.43	-0.99	96600
180 000	53.5	8.05	0.64	34500
180 000	51.5	8.44	-1.19	35500

Table 2

(W = 385 000 lb)

CG position % c_o	Stability augmentation					η_o (deg)	t_R (sec)	t_z (sec)	$t_{h=0}$ (sec)	Maximum height loss (ft)	Peak n (g units)	Time at which peak n occurs (sec)	Distance to h=35 ft (ft)	Distance to h=50 ft (ft)	h after 5 sec (ft)	Fig. No.
	α feedback		q feedback													
	G_α	k_α	G_q	K_q	k_q											
53.5	-	-	-	-	-	- 2	2.05	-	1.7	0.32	>1.56	>6	1455	1605	55	9
51.5	-	-	-	-	-	- 2	2.05	-	1.65	0.31	1.29	3.5	1515	1695	46	9
53.5	1	0.3	-	-	-	- 4	2.5	2.4	1.6	0.55	1.47	2.85	1300	1450	73	11
53.5	-	-	1	-	0.3	- 4	2.5	2.45	1.6	0.45	1.42	3.7	1360	1510	65	11
53.5	1	0.3	1	-	0.3	- 8	2.2	2.05	1.55	0.79	1.55	2.2	1200	1340	84	11
53.5	1	0.3	1	1.25	0.3	- 8	2.2	1.25	1.5	0.70	1.45	2.2	1310	1550	56	12
53.5	-	-	1	1.25	0.3	- 6	2.2	1.6	1.55	0.59	1.41	2.2	1340	1550	57	12
53.5	1	0.3	1	0.8	0.3	- 8	2.2	1.45	1.55	0.73	1.48	2.2	1265	1455	65	-
53.5	1	0.3	1	1.25	0.3	-10	2.25	1.25	1.45	0.82	1.56	2.3	1170	1350	76 †	13
51.5	1	0.3	1	1.25	0.3	-12	2.35	1.35	1.5	0.92	1.59	2.3	1140	1320	78 †	13
53.5	1	0.3	1	-	0.3	-10	1.3	0.95	1.55	1.43	1.56	1.3	1165	1350	74*	16
53.5	1	0.3	1	0.4	0.3	-10	1.3	0.95	1.5	1.38	1.56	1.3	1210	1450	63*	16

† Thrust applied at start of manoeuvre T_o = 25000 lb, k = 0.5

* Control augmented K = 2

Table 3

(W = 180 000 lb)

CG position % c_o	Stability augmentation: α feedback G_α	α feedback k_α	q feedback G_q	q feedback K_q	q feedback K_q	η_o (deg)	t_R (sec)	t_z (sec)	$t_{h=0}$ (sec)	Maximum height loss (ft)	Peak n (g units)	Time at which peak n occurs (sec)	Distance to h=35 ft (ft)	Distance to h=50 ft (ft)	h after 5 sec (ft)	Fig. No.
53.5	-	-	-	-	-	-1	2.025	-	1.15	0.16	1.39	3.4	1295	1450	75	10
51.5	-	-	-	-	-	-1	2.025	-	1.15	0.15	1.3	2.3	1380	1590	56	10
53.5	1	0.3	1	-	0.3	-4	2.1	2	1.05	0.34	1.51	2.1	1150	1325	82	14
51.5	1	0.3	1	-	0.3	-4	2.1	2	1.05	0.32	1.46	2.1	1200	1405	70	14
53.5	1	0.3	1	0.6	0.3	-6	2.15	1.5	1.05	0.47	1.58	2.2	1060	1230	87	15
51.5	1	0.3	1	0.6	0.3	-7	2.175	2.05	1.05	0.52	1.61	2.2	1070	1210	87	15
53.5	1	0.3	1	0.6	0.3	-6	2.15	1.25	1.00	0.44	1.65	2.2	1010	1160	107*	15

* Thrust applied at start of manoeuvre T_o = 40000 lb, k = 0.5

SYMBOLS

A_1,A_2,A_3	coefficients in the analytic representation of C_L (section 2.1)
B_1,B_2,B_3,B_4,B_5	coefficients in the analytic representation of C_D (section 2.1)
C_1,C_2,C_3,C_4,C_5	coefficients in the analytic representation of C_m (section 2.1)
C_m	pitching moment coefficient
C_L	lift coefficient
C_D	drag coefficient
D	differential operator
G_α, G_q	gearings in autostabiliser laws (section 3.1)
K	constant in control augmentation law (section 3.2)
K_q	constant in control law η_q (section 3.1)
R	horizontal distance travelled
S	reference wing area
T_o	incremental thrust demanded at start of the manoeuvre
T	thrust
V	forward speed
W	weight of aircraft
c_o	reference wing chord
d	moment arm of the thrust contribution to pitching moment about CG position
d_o	moment arm of the thrust contribution to pitching moment about the reference CG position
g	acceleration due to gravity
h	incremental altitude
k	constant in thrust equation (section 2.3)
k_α,k_q	constants in autostabiliser laws (section 3.1)
k_B	radius of gyration in pitch
m	mass of aircraft
n	normal acceleration at the CG
n_P	normal acceleration at the pilot's position
q	rate of pitch
t	time

SYMBOLS (Contd.)

$t_{h=0}$	time to regain original height
t_R	duration of pilot's elevator demand (section 3.2)
t_z	time when elevator angle becomes zero
u,w	incremental velocity components in the x,z directions
α	angle of incidence
ρ	air density
η	elevator angle
η_o	pilot's maximum elevator demand (section 3.2)
η_A, η_C	components of η (section 3)
η_P	pilot's elevator demand (section 3.2)
η_α,η_q	components of η_A (section 3.1)
ϑ	inclination of thrust axis to body datum
θ	attitude angle

Subscript

e	trim condition

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2	W.J.G. Pinsker	The landing flare of large transport aircraft. RAE Technical Report 67297 (ARC R & M 3602) (1967)

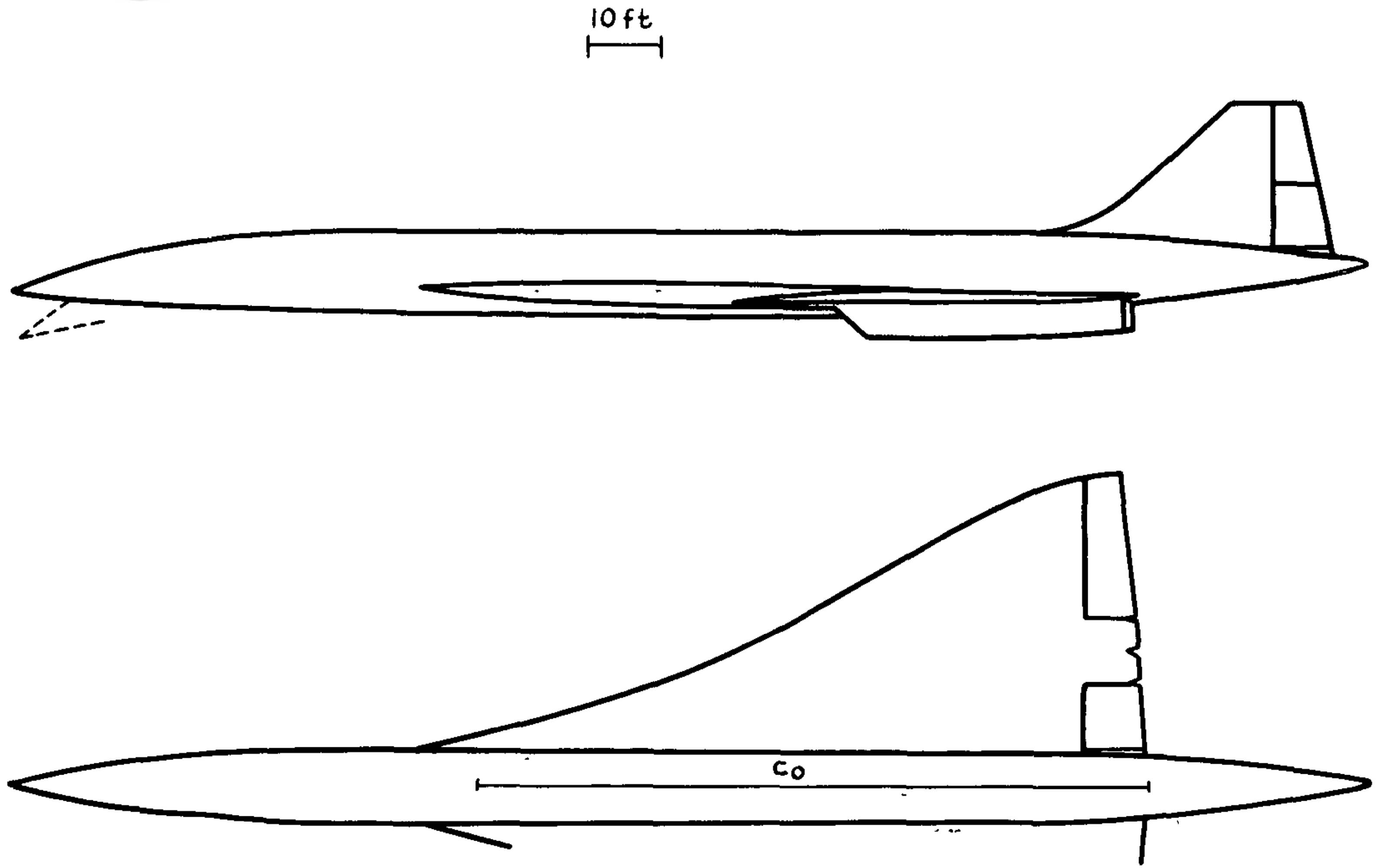

Fig. 1 General arrangement of aircraft

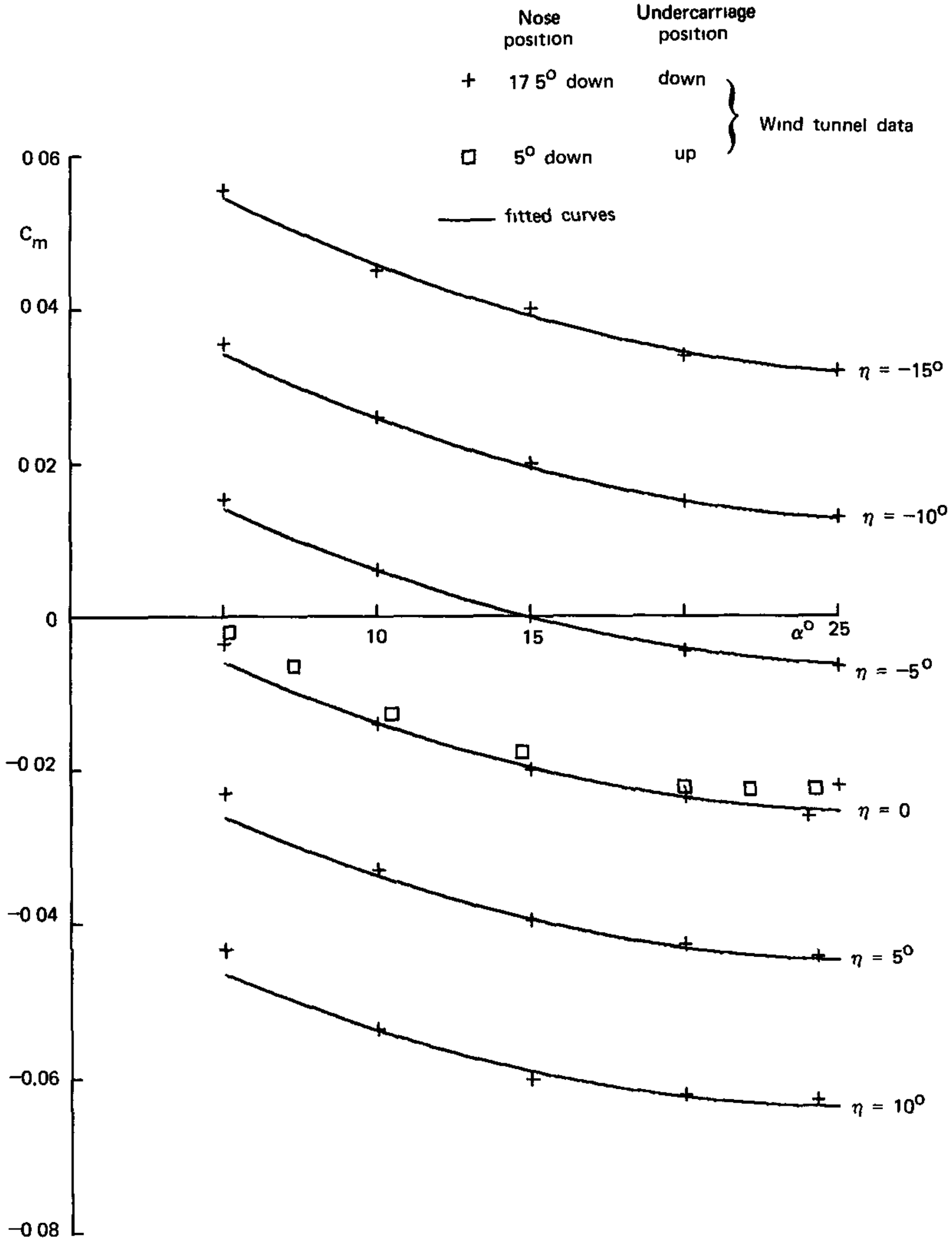

Fig.2 Comparison of wind tunnel data and fitted curves . C_m v. α for various η's
CG position 50% c_o

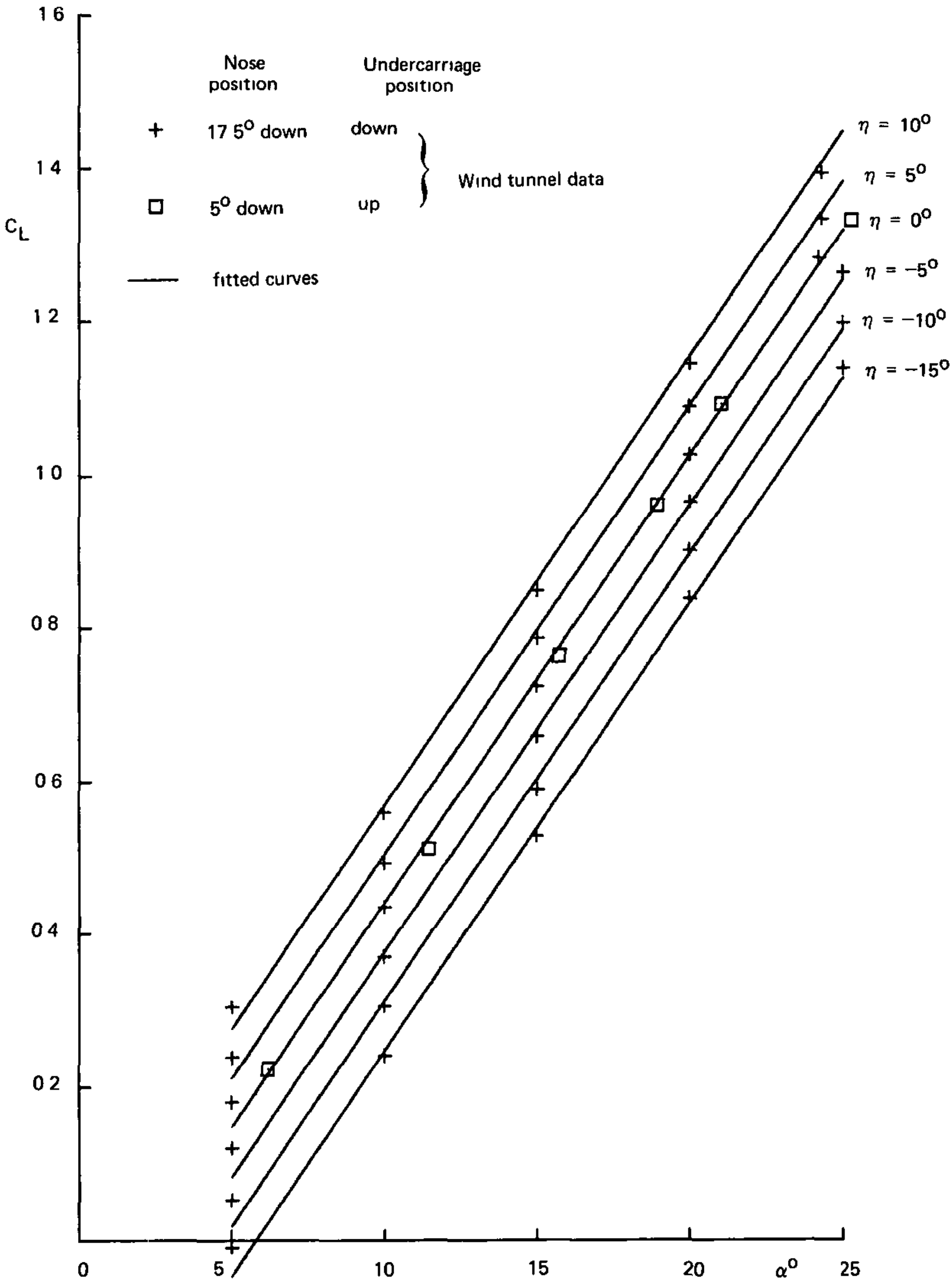

Fig 3 Comparison of wind tunnel data and fitted curves C_L v α for various η's

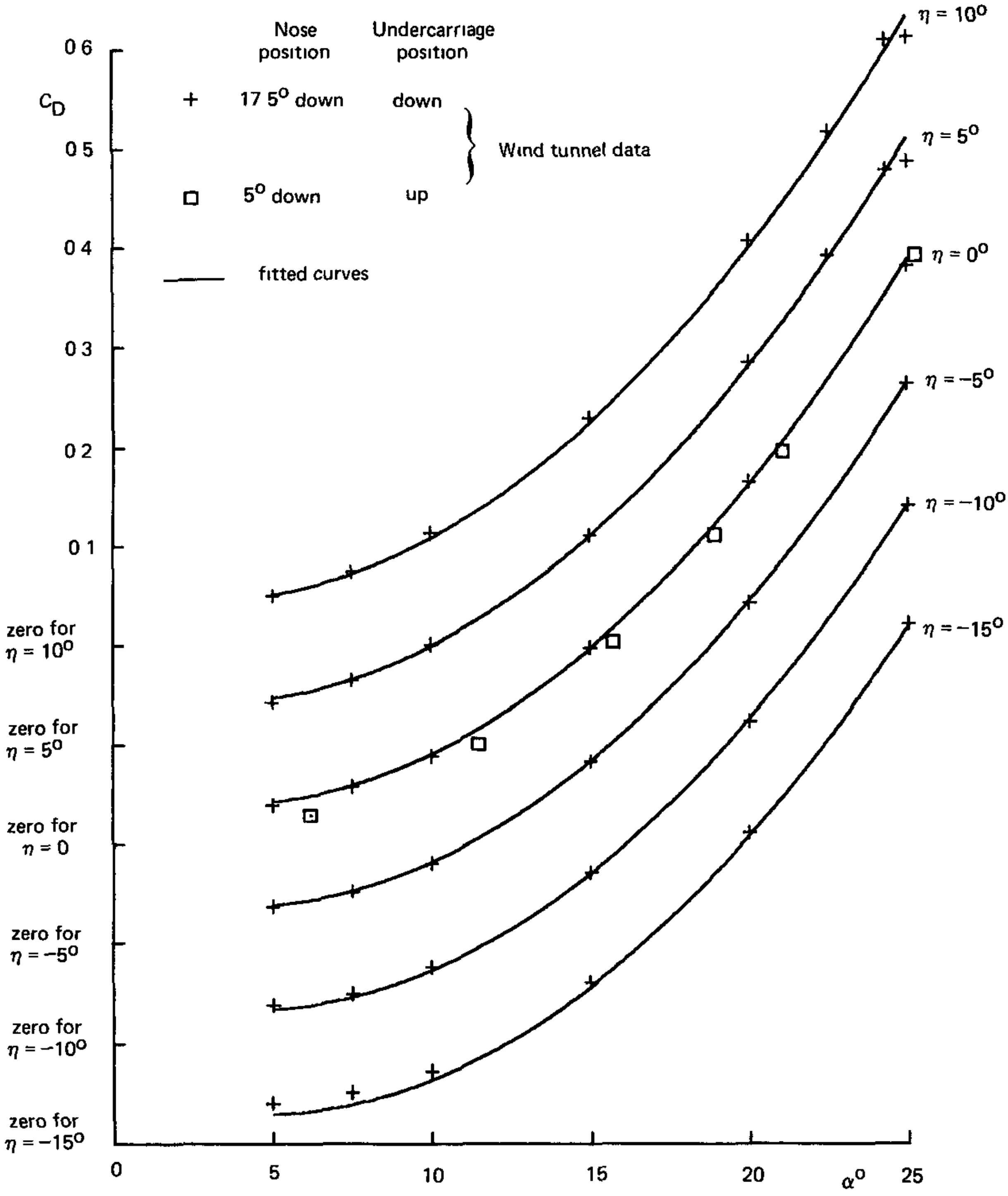

Fig 4 Comparison of wind tunnel data and fitted curves C_D v α for various η's

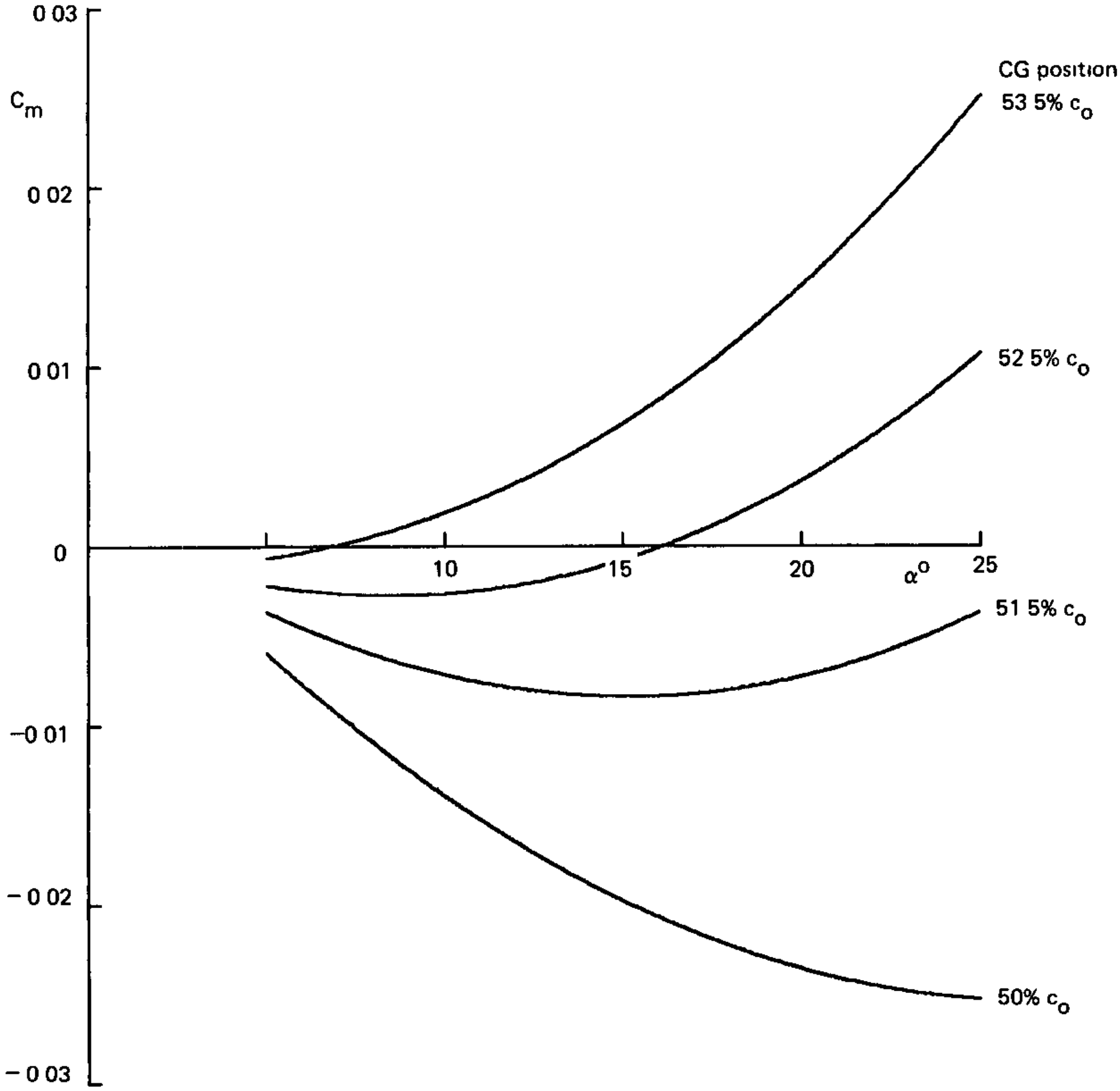

Fig.5 C_m v α for various CG positions $\eta = 0$

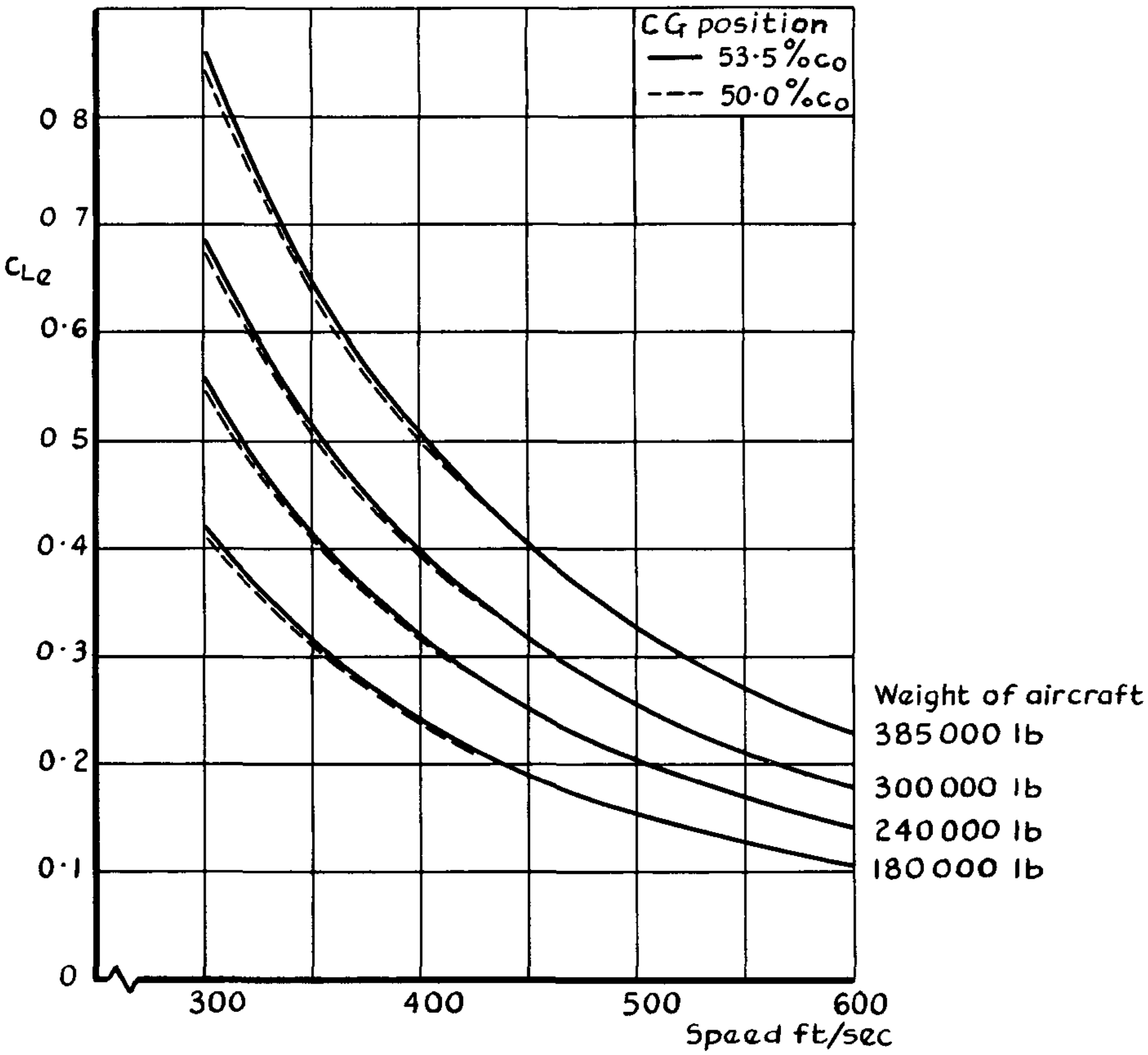

Fig. 6 1g trim conditions C_{L_e} vs V for various weights of the aircraft

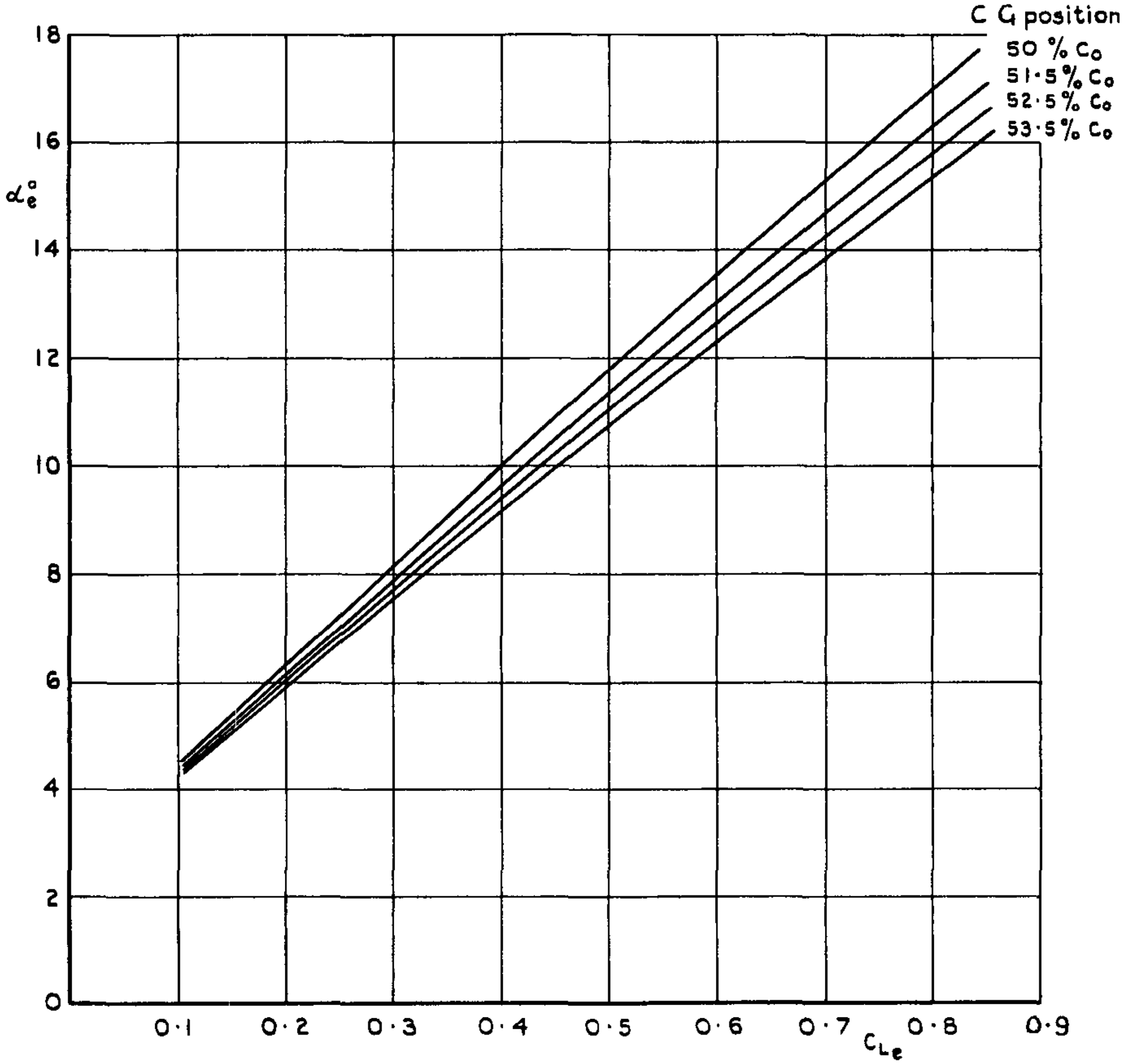

Fig. 7 Ig trim conditions α_e vs C_{L_e} for various CG positions

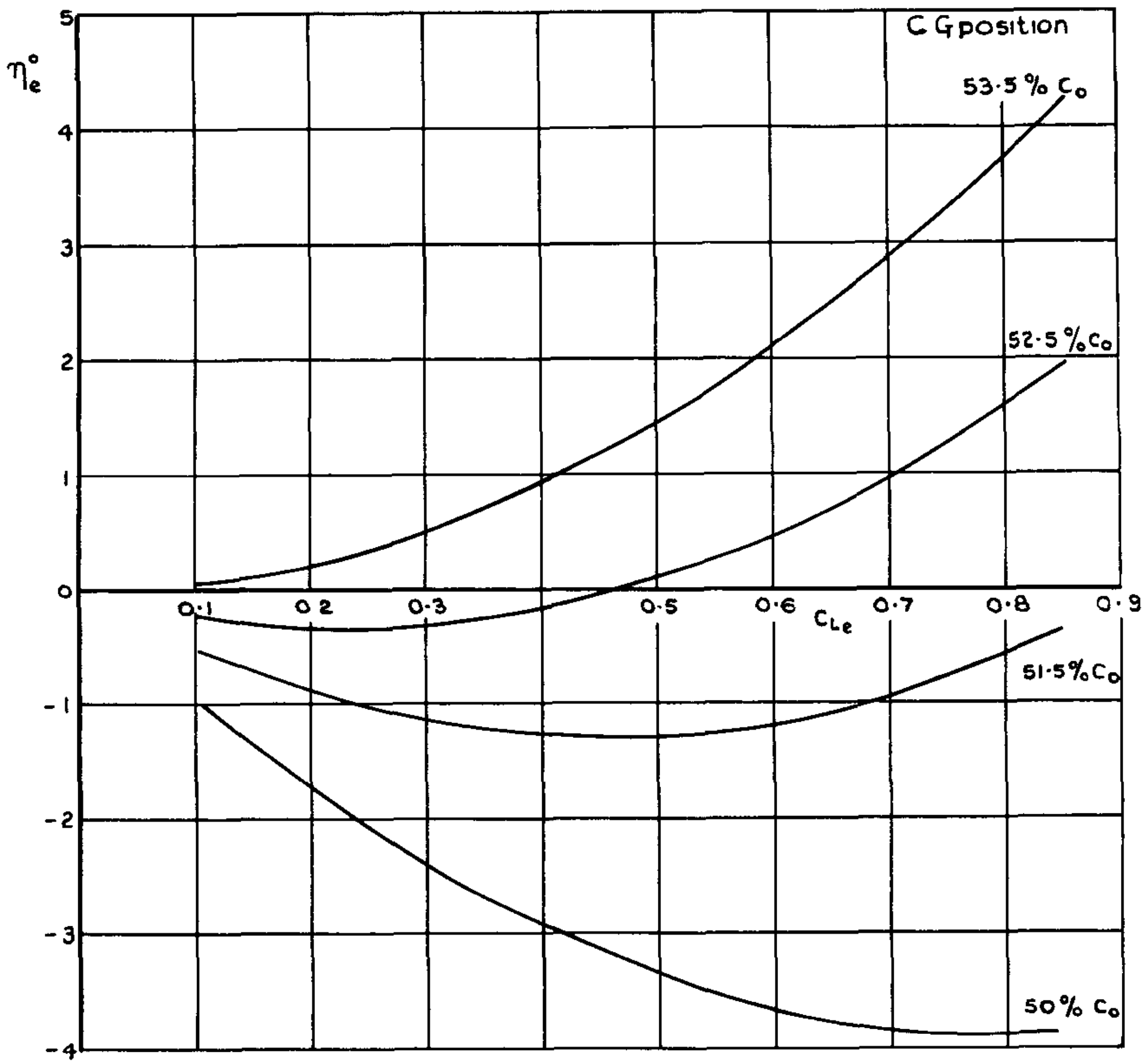

Fig. 8 Ig trim conditions η_e vs C_{L_e} for various CG positions

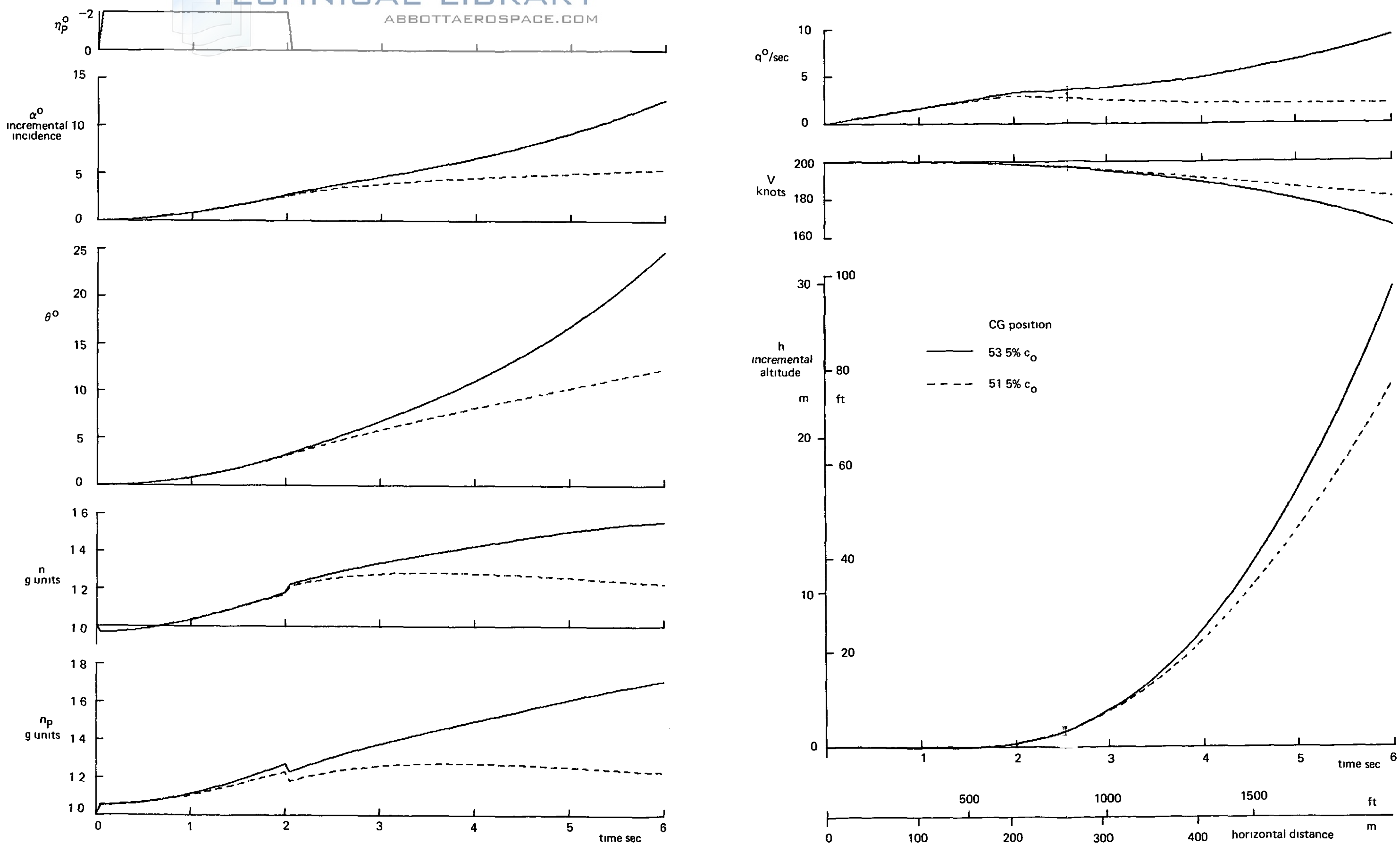

Fig 9 Response of the unaugmented aircraft . W = 385 000 lb

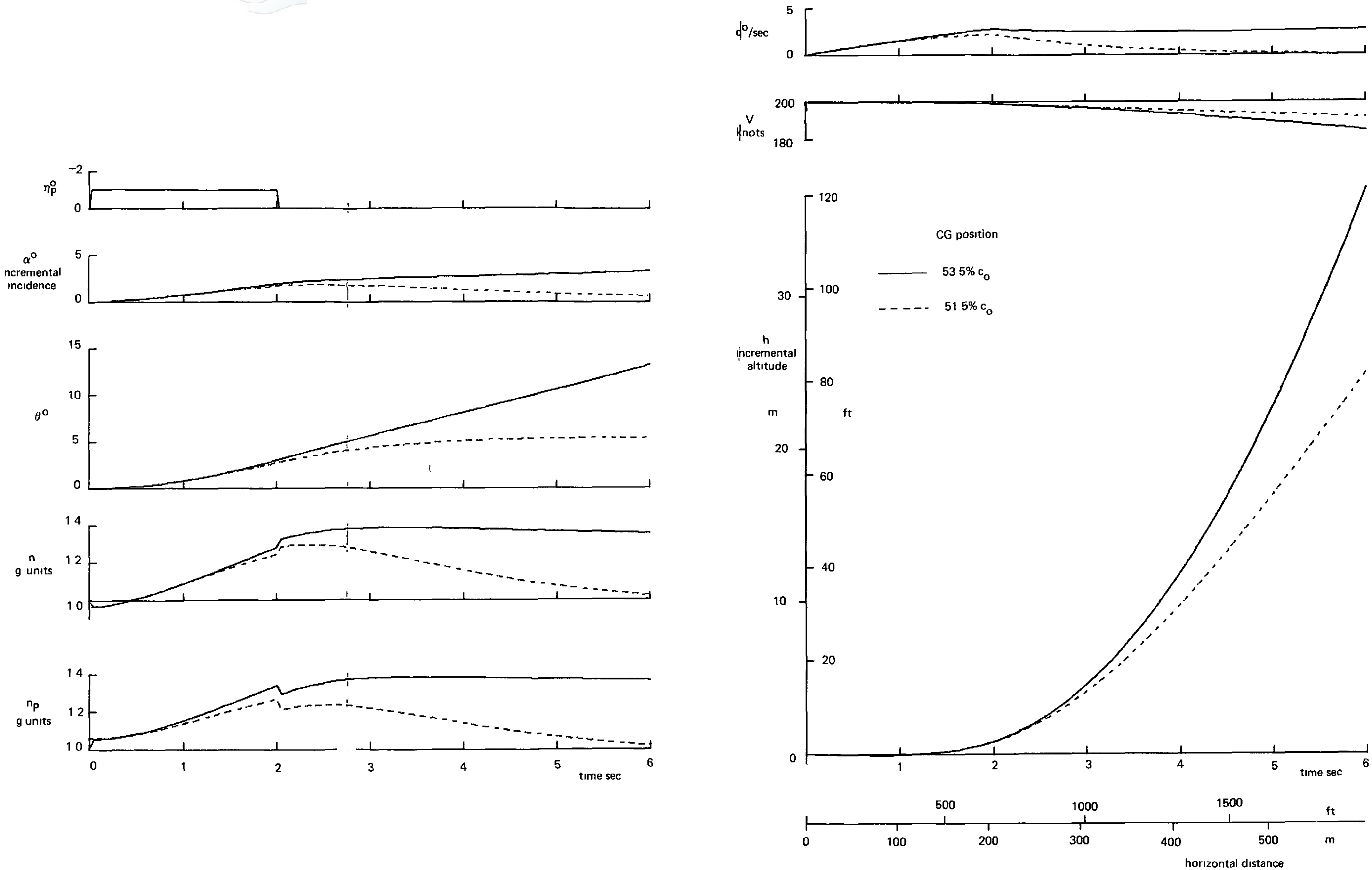

Fig 10 Response of the unaugmented aircraft W = 180 000 lb

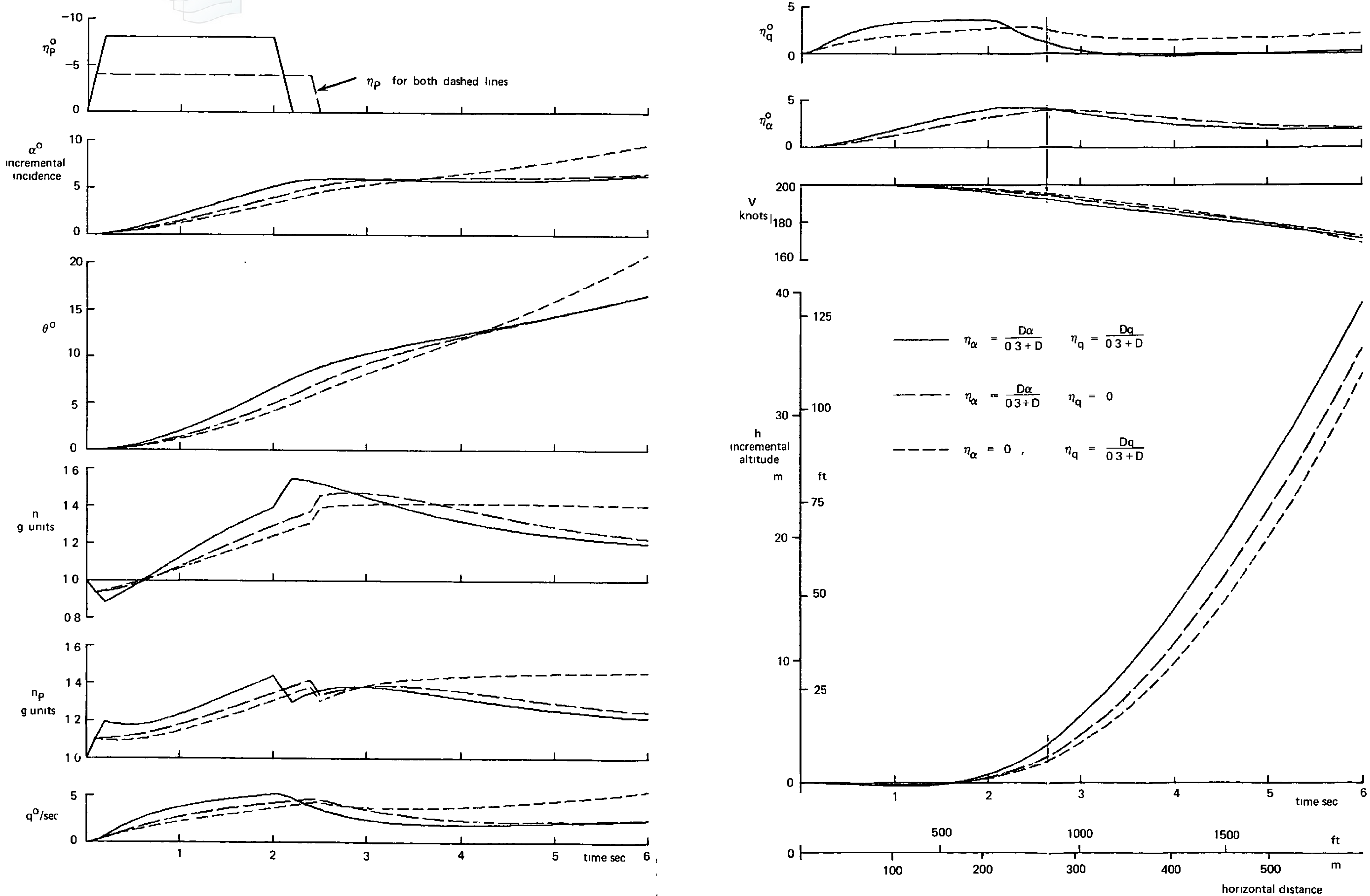

Fig 11 Response of the aircraft W = 385 000 lb with CG located at 53.5% c_o with an autostabiliser providing α feedback, q feedback and α and q feedback

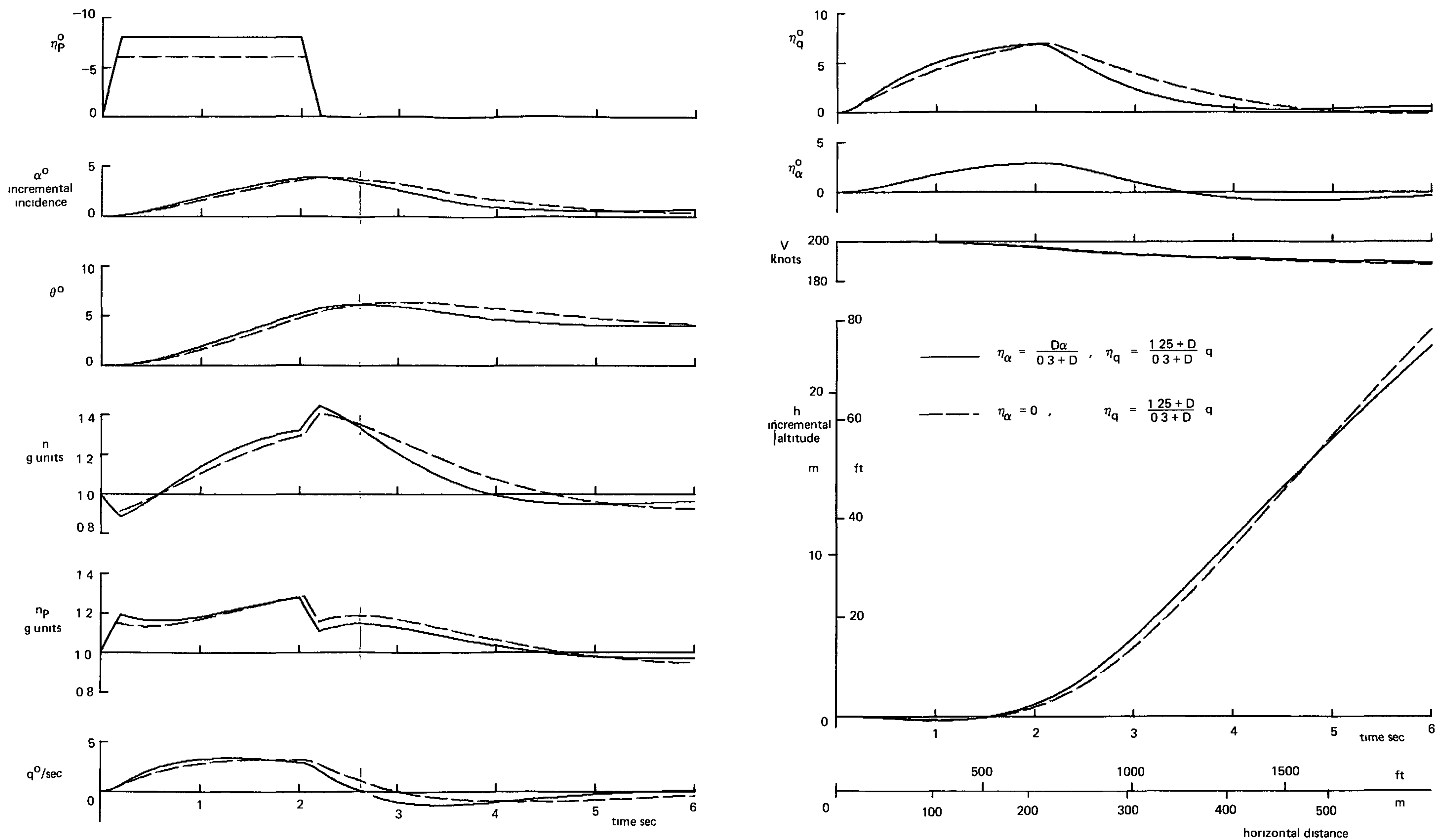

Fig 12 Response of the aircraft W = 385 000 lb with CG located at 53.5% c_o with position term included in the pitch autostabiliser

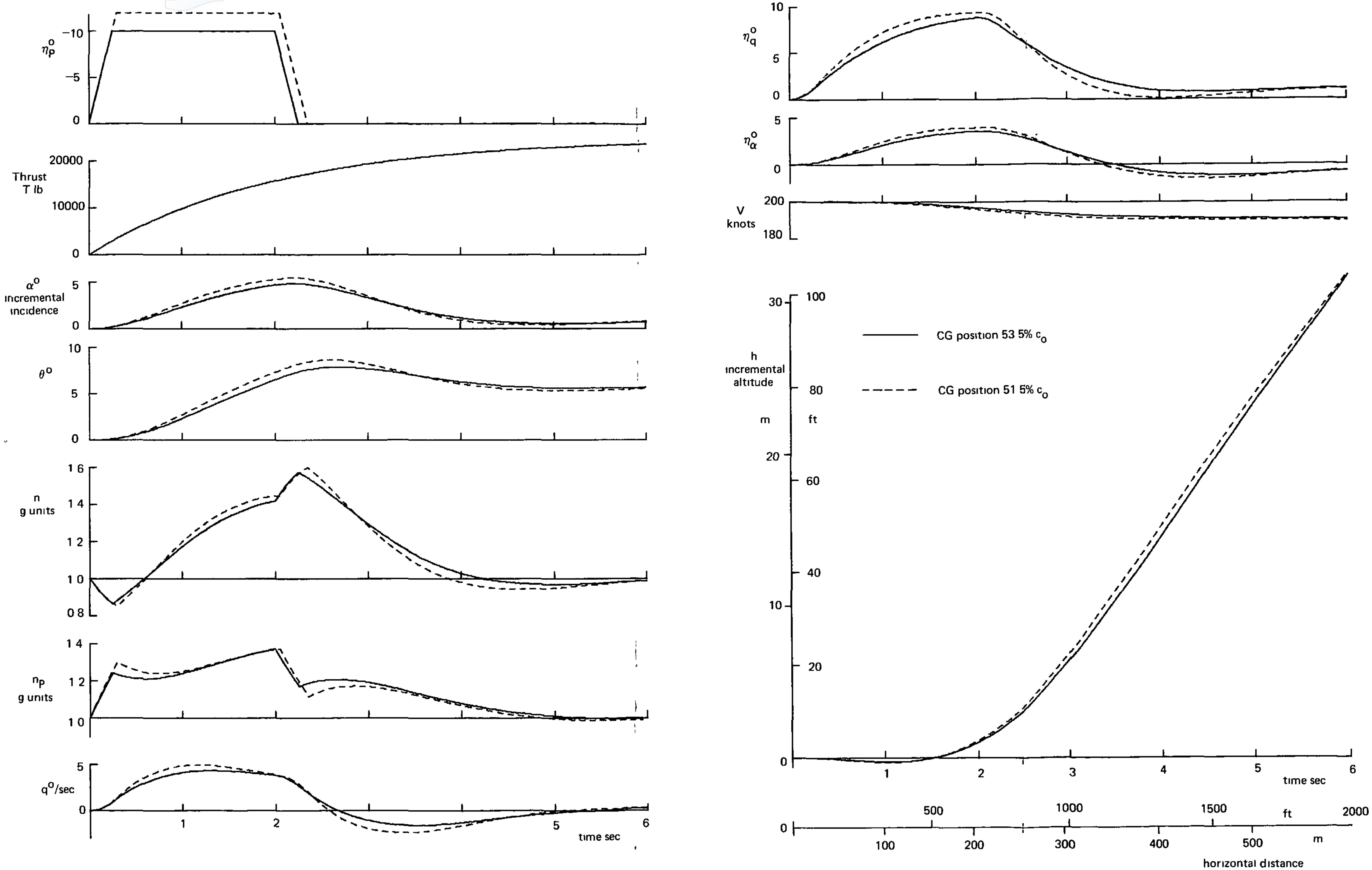

Fig 13 Response of the aircraft W = 385 000 lb with thrust applied at start of manoeuvre and autostabiliser providing α and q feedback

T_o = 25000 lb, k = 0.5, $\eta_\alpha = \frac{D\alpha}{0.3+D}$, $\eta_q = \frac{1.25+D}{0.3+D}\, q$

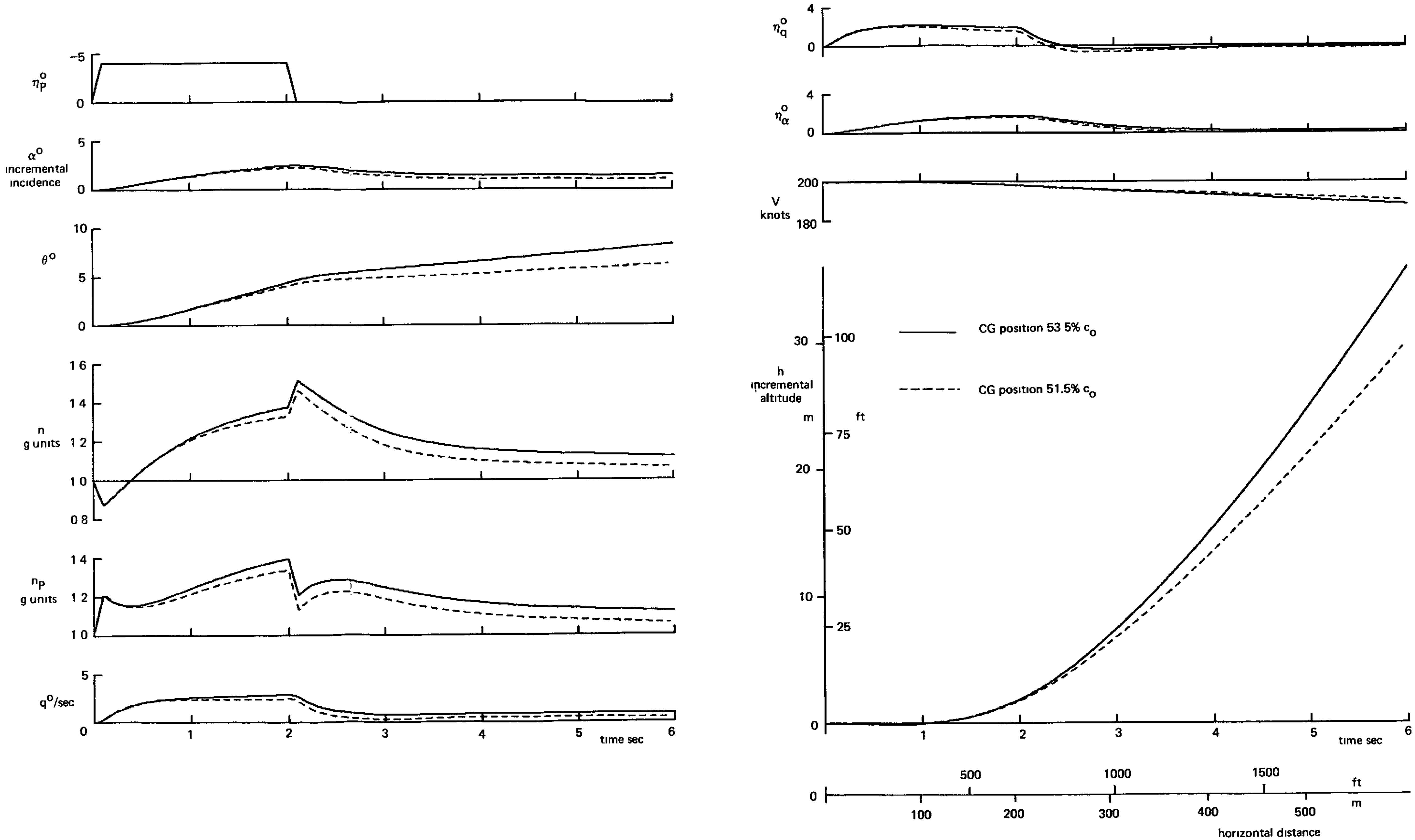

Fig.14 Response of the aircraft W = 180 000 lb with an autostabiliser providing α and q feedback

$$\eta_\alpha = \frac{D\alpha}{0\,3 + D}, \quad \eta_q = \frac{Dq}{0\,3 + D}$$

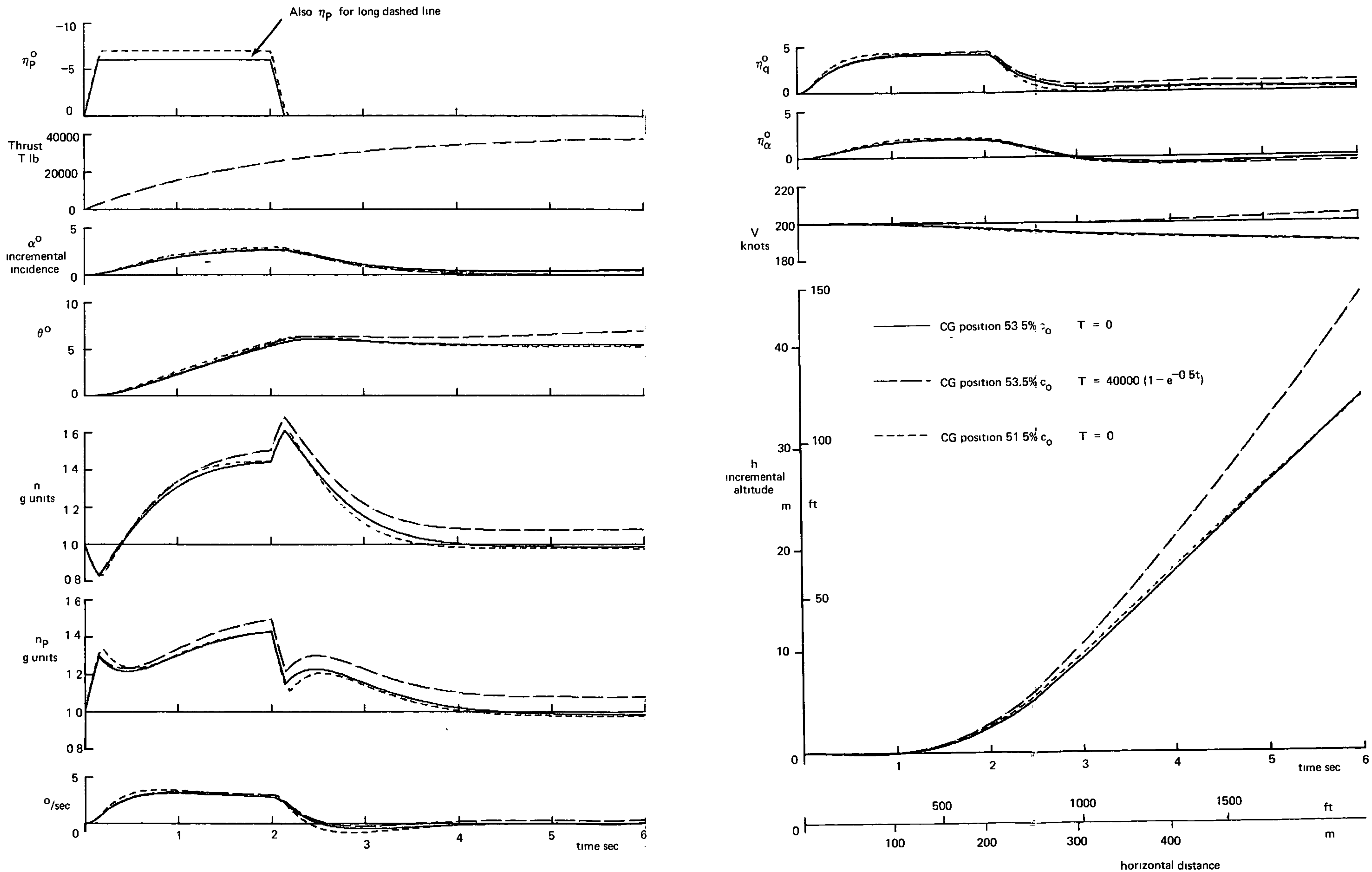

Fig.15 Response of the aircraft W = 180 000 lb with position term included in pitch autostabiliser and the effect of thrust applied at start of manoeuvre

$\left\{ T_o = 40000 \text{ lb}, \ k = 5 \right\} \ \eta_\alpha = \frac{D\alpha}{0\ 3 + D}, \ \eta_q = \frac{0\ 6 + D}{0\ 3 + D}\ q$

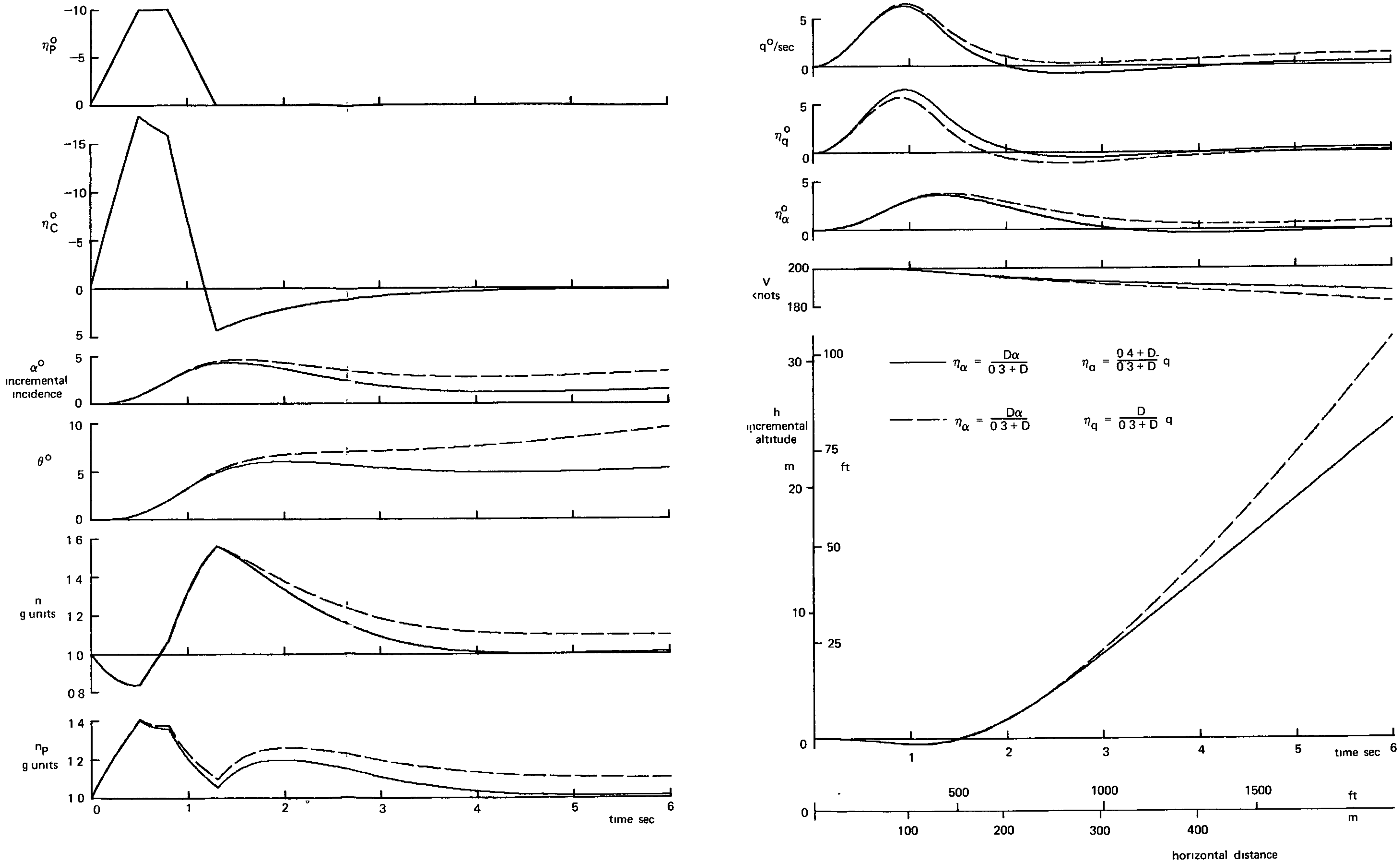

Fig.16 Response of the aircraft W = 385 000 lb with CG located at 53.5% c_o with an autostabiliser providing α and q feedback and control augmentation
$\eta_c = \frac{1+2D}{1+D}\eta_P$

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October 1970

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LOW SPEED PULL-UP MANOEUVRES FOR A SLENDER WING TRANSPORT AIRCRAFT WITH STABILITY AND CONTROL AUGMENTATION

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533.6.013.644
533.6.013.47
629.137.1
533.693.3
629.13.014
533.6.013.4
621–52

Low speed pull-up manoeuvres for a slender wing transport aircraft are calculated. Two extremes of aircraft weight are considered, 385 000 lb and 180 000 lb For each aircraft weight, two CG positions are considered Stability augmentation, in the form of angle-of-incidence and/or rate-of-pitch feedback, and control augmentation are investigated as a means of improving the response of the aircraft in pull-up manoeuvres.

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