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The Influence of Drag Characteristics on the Choice of Landing Approach Speeds

by

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THE INFLUENCE OF DRAG CHARACTERISTICS ON THE CHOICE OF LANDING APPROACH SPEEDS

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Paper for presentation to the AGARD Flight Test Panel, London, May, 1957

SUMMARY

A study has been made of the lift-drag characteristics of 19 jet aircraft which have minimum comfortable approach airspeeds that are fairly well established. On about half these aircraft, drag effects, which determine speed stability, were stated to limit the approach airspeeds, though there was naturally some lack of unanimity as to the level of the effect which could be classed as tolerable. Limiting values of speed stability are proposed for the three types of landing approach used for carrier, airfield and instrument landings. The speeds corresponding to these stability levels are in fair agreement with those actually used, even for those aircraft whose approach speeds were not primarily limited by drag effects. It is considered that drag effects must be taken into account in estimating approach airspeeds, and that this process materially improves the prediction of approach speeds for the latest types of aircraft.

Further data should be collected and analysed in order to confirm or modify the form or level of the proposed limits.

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1 INTRODUCTION

It has been appreciated for some time that among the many factors which influence the pilot in his choice of approach airspeed, the problem of controlling speed and rate of descent can play a significant part. In common with most of these other factors, speed and altitude control deteriorate as speed is reduced, and may in some cases become the main reason for not willingly reducing the approach speed still further. When this is the case, it should be possible, from a study of the aerodynamic parameters governing the speed and altitude control problem, to assign limiting values to these parameters such that the limiting approach speed of a new aircraft might be predicted. This procedure has, so far, met with scant success, because the speed and height control problem is only one of several, any one of which might override it and demand a higher speed for acceptable safety on the approach. Among the more important of these contending factors are the risk of an inadvertent stall, due to a gust or while manoeuvring, the danger of loss of stability about any of the three axes, the restriction of visibility from the cockpit at high incidence, or deterioration in control response generally.

Prediction of approach airspeed must take account of all these factors, and several others and we cannot hope to produce a simple rule based on only one aspect of the approach problem.

Nevertheless, deterioration of speed and altitude control is given as the main reason for maintaining a chosen minimum airspeed in more cases than one would expect, considering the variety of other reasons that might be given, and it has therefore been thought worthwhile to examine this control problem, on a number of aircraft for which the relevant aerodynamic data is available, in a further attempt to define a minimum acceptable standard.

2 DEFINITIONS

The minimum comfortable approach speed is defined as that airspeed below which the pilot will not, from choice, allow the aircraft to decelerate during the approach, up to the point where he reduces thrust and starts the flare-out. It can, therefore, be the speed which he tries to maintain throughout the whole approach, from the completion of the turn off the crosswind leg, or it can be the speed down to which he allows the aircraft to decelerate as it crosses the threshold of the landing area. In either case, it is known as the speed "over the hedge", as distinct from a possibly higher speed at the start of the approach.

Two distinct types of approach path must be recognised. The first is the so-called "carrier type" approach, made from a fairly tight, low altitude circuit as used by the Navy. The final straight descent path is joined at an altitude of 2-400 feet, so that steady conditions on the final approach are held for less than one minute, possibly for as little as 30 seconds. This relatively short time is significant.

The instrument approach, however, starts at an altitude of between 1000 and 2000 feet, and thus lasts of the order of 5 times as long as the first. Consequently, the pilot, while having more time to establish the desired steady conditions, is more conscious of extraneous effects which tend to disturb those conditions. Generally, therefore, a higher approach speed will be chosen for this type of approach, if, as is usual, a steady speed is to be maintained.

It is a characteristic of this second type of approach that the pilot is attempting to follow a prescribed glide path and it is under just these conditions that speed and altitude control problems are most acute. The same is true of the now-standard Naval approach for carrierl andings, using the mirror landing aid, though the time involved is much shorter.

Normal airfield landings are made along much less precisely defined paths, and in the limit the speed control problem can disappear if no attempt is made to follow a fixed path, or to aim for a fixed touch-down point, i.e. if there is no altitude control problem.

Generally, however, some attempt is made to follow a set path, and to touch-down somewhere near the down-wind end of the runway, and the closer this control of altitude becomes, the more aware will the pilot be of any deficiency in speed control.

It is therefore clear that a pilot's opinion of a minimum approach airspeed is difficult to interpret unless it is known what type of path he was attempting to follow.

A further difficulty arises from the inevitable divergence of opinion as to what constitutes a minimum comfortable speed for a particular type of approach. Pilots may well agree on the speed which they normally use as a safe minimum on a particular aircraft, but there is a lack of information on how far this speed could be reduced if the need arose, while keeping the standard of control of speed and altitude above an agreed minimum. The difficulty is, of course, to define this minimum standard of control.

3 SOME THEORETICAL RELATIONSHIPS

A comprehensive survey by Neumark[1] has established the criteria which define the regimes in which flight "under restraint" is stable or unstable. By "restraint" we mean the suppression, by the pilot using his elevator, of altitude disturbances, either relative to a horizontal or, more generally, to any rectilinear path. Stability therefore refers to the behaviour of the airspeed, following some initial disturbance, rather than to the more usual longitudinal mode.

The stability criterion, or condition for subsidence of the disturbance is given as

$$C_D + C_{AS} - \frac{C_L \cdot dC_D/dC_L}{1 + C_D/a} > 0 \qquad (1)$$

where C_L and C_D are respectively the lift and drag coefficients, a is the lift-curve slope $dC_L/d\alpha$, and C_{AS} is the equivalent drag coefficient of the propulsion system. C_{AS} thus depends on speed and throttle position, and is defined as

$$C_{AS} = -\frac{1}{S} \cdot \frac{\partial T}{\partial q} \qquad (2)$$

T being the thrust, q the dynamic pressure, and S the wing area.

The lift coefficient at which the left-hand-side of the inequality (1) becomes zero defines the airspeed below which, if the pilot tries to maintain a steady straight path, without varying the engine thrust, any disturbance in speed will be divergent. The time constant of the divergence (or of the subsidence, if the speed is above the critical) is inversely proportional to the left-hand-side of this inequality.

The inequality may justifiably be simplified in most cases (except near to, or at the stall) by ignoring the term C_D/a in the denominator. We can combine C_{AS} with the total drag coefficient when the thrust varies simply as V^2, or is constant.

If C_D/a is negligible, the inequality can be written

$$\frac{C_D + C_{AS}}{C_L} - \frac{dC_D}{dC_L} > 0 \tag{3}$$

since C_L is always positive. The time constant, τ, of the subsidence or divergence is given by

$$\tau = V\bigg/2g\left(\frac{C_D + C_{AS}}{C_L} - \frac{dC_D}{dC_L}\right) \tag{4}$$

We can define the minimum drag speed as that at a lift coefficient which satisfies the equation

$$\frac{C_D}{C_L} - \frac{dC_D}{dC_L} = 0 \tag{5}$$

while the speed for minimum glide angle with power on is given by

$$\frac{C_D + C_{AS}}{C_L} - \frac{dC_D}{dC_L} = 0 \tag{6}$$

which differs from the minimum drag speed unless C_{AS} is zero, i.e. the thrust is constant.

The minimum rate of descent at a fixed throttle setting occurs at a speed which is lower than that for minimum drag, and which is given by

$$\frac{C_D + C_{AS}}{C_L} - \frac{dC_D}{dC_L} = -\frac{1}{2}\left(\frac{C_D}{C_L} - \frac{T}{W}\right) \tag{7}$$

where $\frac{T}{W}$ is the corresponding thrust/weight ratio.

The various expressions defining the speeds for minimum drag, minimum glide angle and minimum rate of descent, as well as that for the time constant of the divergence or subsidence of a disturbance in speed, all contain the function

$$\left(\frac{C_D}{C_L} - \frac{dC_D}{dC_L}\right)$$

where C_D contains the term C_{AS}. It is however, mainly because this function determines the time constant of the decay or divergence of a speed error that it is believed to be of importance in determining the standard of control of speed which the pilot considers to be tolerable.

4 DATA AVAILABLE

Flight measurements of lift and drag in the landing configuration have been collected and analysed for 19 different aircraft. Relevant details of these aircraft are given in Table 1.

The lift and drag data are presented in two forms. In Figs. 1, 2 and 3 the variation of C_D/C_L is shown as a function of airspeed. C_D/C_L is, to a first approximation, equal to the thrust/weight ratio required to maintain level flight. The thrust/weight ratio needed for an approach at a glide angle γ will be less than this by a constant amount equal to the glide angle in radians.

In **Figs.** 4, 5 and 6 the apparently - critical parameter $(C_D/C_L - dC_D/dC_L)$ is plotted against airspeed. It is related to the variation of thrust required for level flight (T_{RL}) by the equation

$$\frac{C_D}{C_L} - \frac{dC_D}{dC_L} = \frac{V}{2W} \cdot \frac{d\,T_{RL}}{dV} \quad . \tag{8}$$

In this analysis, all the aircraft are jet-propelled, and C_{AS} will usually be negligibly small in comparison with C_D. C_{AS} need not henceforth appear as a separate coefficient, and will be absorbed into C_D. This assumes that the thrust is either constant, or varies linearly with V^2.

On each curve the minimum comfortable approach speed is marked, where possible, both for a normal airfield approach and for either a carrier type approach or an instrument approach. It is obviously not possible to be dogmatic about these speeds, since only in a few cases has a systematic series of tests been made, aimed at reducing the approach speed to the minimum. Only in those cases can one expect different pilots to agree as to this minimum to nearer than, say, 5 knots. Where speeds are quoted more precisely than this, they have been obtained either as mean measured values from landings made by a number of pilots (carrier landings in particular) or as minima established by systematic test. Other figures may be termed "generally accepted values".

These results are summarised in Table 1 according to the type of approach to which they refer. This table also lists the speed corresponding to the maximum available lift coefficient $\left(V_{C_{L_{max}}}\right)$ and the minimum drag speed, V_{MD}. The former speed is quoted without reference to controllability at this speed, i.e. it may well be lower than the lowest speed for steady controlled flight. The latter speed, V_{MD}, is the speed at which the curves of Figs. 4, 5 and 6 intersect the speed axis, i.e. the speed at which $(C_D/C_L - dC_D/dC_L)$ becomes zero. The ratio of the approach speeds to $V_{C_{L_{max}}}$ and to V_{MD} are also given.

The data in Table 1 are split into three groups, which correspond to the graphical data in Figs. 1 and 4, 2 and 5, and 3 and 6 respectively. In the first group (Figs. 1 and 4) are those aircraft whose approach speeds are stated to be limited primarily by difficulties of control of speed and height, i.e. by drag effects. The second and third groups comprise those whose speeds are limited for other reasons, as indicated in Table 1, the third group being distinguished only because these aircraft were fitted with some form of boundary layer or circulation control to increase lift.

On each curve of Figs. 4, 5 and 6, the speed corresponding to the maximum available lift coefficient is indicated.

5 DISCUSSION

5.1 Suggested criterion for limiting approach speeds

Fig.4 summarises the available data on the aircraft which exhibit the phenomenon under investigation, and it is here that we should expect to see some significant trend in the value of $(C_D/C_L - dC_D/dC_L)$ at the chosen approach speeds.

It is immediately obvious, however, that there is a lack of unanimity of opinion as to what degree of speed stability defines a "minimum comfortable" speed and one is forced to the usual excuse that much more information is needed before any really sound conclusions can be drawn.

However, we can double the amount of data available if we include that for aircraft on which the pilots did not specifically complain of difficulty due to height and speed control. This additional information will, at least, show what degree of instability (in this sense) can be tolerated by some pilots when other problems are present. Fig.7 therefore is a précis of all the data, and shows the values of $(C_D/C_L - dC_D/dC_L)$ at the chosen approach speeds. Different diagrams refer to the three classes of landing approach - carrier, airfield and instrument types.

There is still considerable scatter in the values of the stability parameter $(C_D/C_L - dC_D/dC_L)$ actually used, and even the general trend is not clearly indicated. There is no point in quoting mean values of the parameter for the three types of approach since roughly half the aircraft would be approaching with worse stability than that corresponding to the mean. It is, of course, limiting values that we require.

An earlier study of the problem[2], based on specific tests on a few aircraft making carrier-type approaches, suggested that since the time constant of the divergence or subsidence is proportional to speed (equation 4), the pilot would be aware of the rate at which a speed error grows or diminishes in terms of distance travelled. Using equation 4, the variation of speed with time, t, and distance, s, may be written, for small variations,

$$u = u_o\, e^{-t/\tau} = u_o\, e^{-g\rho s \frac{C_L}{W/S}\left(\frac{C_D}{C_L} - \frac{dC_D}{dC_L}\right)} \qquad (9)$$

where the variable speed $V = V_A + u$. V_A is the trimmed speed and u_o is the initial error from the trimmed speed.

This early work produced the tentative conclusion that for carrier landings, the quantity

$$-\frac{C_L}{W/S}\left(\frac{C_D}{C_L} - \frac{dC_D}{dC_L}\right)$$

ought not to exceed +0.003 sq ft/lb. This criterion has since been interpreted by Neumark as meaning that the minimum comfortable speed (for a carrier-type approach) is such that any initial small speed error would be doubled after the aircraft has moved 1000 yards. We are thus concerned with very slow rates of divergence, yet there was real difficulty in keeping the aircraft on the glide path at speeds below this limit.

This criterion is shown graphically on the diagram for carrier approaches in Fig.7, by plotting the resulting variations of $(C_D/C_L - dC_D/dC_L)$ with speed. In its present form, however, the criterion is not dimensionless, and it has been suggested by Neumark[4] that it can be made non-dimensional by multiplying it by the ambient pressure, p, so that, denoting the criterion by F, we have

$$F = -\frac{pC_L}{W/S}\left(\frac{C_D}{C_L} - \frac{dC_D}{dC_L}\right)$$

i.e.,

$$F = -\frac{2}{\gamma M^2}\left(\frac{C_D}{C_L} - \frac{dC_D}{dC_L}\right) \tag{10}$$

where $\gamma = 1.4$ is the adiabatic constant, and M is the Mach number. The limiting value of F for carrier-type approaches is then approximately 6, which is, incidentally a more convenient numerical value.

The variation of speed error with distance travelled may then be written

$$u = u_o e^{g \rho s F/p} \tag{11}$$

or, with numerical values appropriate to standard sea-level conditions,

$$u = u_o e^{3.616 \times 10^{-5} s F} .$$

In Table 1, values of F for the individual aircraft are recorded, and these are plotted in Fig.8 against speed, which is now expressed as the ratio $V_A/V_{C_{L_{max}}}$. This diagram shows that the additional data now available does not justify any change in the value of F for carrier approaches, and a limiting line is drawn at $F = 6$. In spite of the considerable scatter, one may tentatively suggest corresponding limiting values of $F = 2$ for airfield approaches and $F = -2$ for instrument approaches. These additional proposed limiting values are also shown on the remaining two diagrams in Fig.7, and all appear on the basic diagrams in Figs.4, 5 and 6.

On Fig.8, the symbols referring to aircraft whose approach speeds were stated to be limited by drag effects are distinguishable from those limited by other factors. It is apparent that this former group of aircraft do not exhibit markedly worse speed stability characteristics than the others. In all cases, this standard of speed stability was considered by the pilots to be tolerable, whether or not it was the limiting factor, and one is therefore justified in grouping all the data together in arriving at these limits.

The simple interpretation of these new limits is similar to that suggested for the carrier approach case, namely, that for airfield approaches, a small initial speed error can double itself in 3000 yards, while on an instrument approach stability is satisfactory if a speed error decays to half its initial value in 3000 yards.

5.2 Effect of type of landing approach on approach speed used

It may perhaps be inferred from the above that an airfield approach must be made at a speed higher than that which appears acceptable on a carrier. This, of course, is not the case. Rather is it that the naval pilot is under much greater pressure to bring his speed down to the absolute minimum than is the case when an airfield approach is being made. This, plus the fact that his approach is probably a shorter one than is usual ashore, makes it possible for him to be more tolerant of this form of instability. The naval pilot would, of course, welcome an improvement in speed stability as much as anyone, but the benefit has to be weighed against the severe penalty of the higher approach speed involved. Conversely, the shore-based pilot, of equal skill, would be able to cope with a lower standard of speed stability than appears to be usual, but the advantages of the lower approach speed (a shorter landing distance, for example) are seldom sufficiently important to make the exchange worthwhile.

On an instrument approach, however, the pilot is very much pre-occupied with the manoeuvring of the aircraft in three dimensions, and any tendency for airspeed to wander is a distraction which he will wish to avoid. Positive stability of airspeed is a reasonable requirement in such a case, and has already been postulated, in a different form, by Prescott[3]. In this alternative form, the approach speed should be such that the change in pitch attitude per unit change in glide angle does not exceed +2.7. This ratio, n, may be written, with $a = dC_L/d\alpha$,

$$n \quad = \quad d\theta/d\gamma \quad = \quad 1 + \frac{C_L}{a\left(\frac{C_D}{C_L} - \frac{dC_D}{dC_L}\right)} \qquad (12)$$

from which it is seen that n increases as the speed is reduced towards the minimum drag speed, and is infinite at V_{MD}.

The two criteria are related, according to equations (10) and (12), so that

$$n \quad = \quad 1 - 2\, C_L/a\gamma\ M^2 F\ . \qquad (13)$$

The speeds at which n and F have their proposed limiting values of 2.7 and -2.0, respectively, for instrument approaches are compared in the following table for aircraft J, H and O, for which actual instrument approach speeds are also available.

Aircraft	Speed for n = 2.7, Knots	Speed for F = -2.0, Knots	Actual speed, Knots
J	134	103	115
H	155	158	145
O	132	119	130

Prescott's criterion gives excellent agreement with fact on aircraft O, upon which aircraft the criterion was mainly based. For the others, there is little to choose between the two methods. Overall, Prescott's criterion produces speeds which average 10 knots higher than those actually used on the above three aircraft, while the present method produces speeds which average about 3 knots lower than practice. The scatter is about the same with the two methods, so far as this can be judged from three results.

5.3 Comparison between actual and limiting approach speeds

In making these comparisons, it must be emphasised at the start that the limiting approach speeds have been derived solely from considerations of speed stability, with no reference to the many other aspects of the overall handling problem which might demand some higher approach speed. Thus, where the suggested limiting speed is less than that actually used, it cannot be suggested that the approach has been made unnecessarily fast.

Across each of Figs.4, 5 and 6, three lines are drawn which represent the suggested variation of the limiting values of the function $(C_D/C_L - dC_D/dC_L)$ with speed for each of the three types of approach. The intersections of these lines with the curves giving the measured variation of this function with speed for the individual aircraft give the above limiting approach speeds. These speeds are listed in Table 2 and also plotted on Fig.7.

Table 2 shows that the speeds corresponding to the proposed minimum standards of speed stability (i.e. the "limiting" speeds) were generally lower than those actually used. This is only to be expected since the proposed standards define lower limits.

For the aircraft whose approach speeds were limited by drag effects, the limiting carrier and airfield approach speeds average only 2-3 knots lower than those actually used. For the remaining aircraft, the mean difference is still only about 4 knots for carrier type approaches, but about 10 knots for airfield or instrument approaches.

There is no significant difference in the scatter, represented by the root-mean-square deviation, between aircraft which were limited by drag effects and those which were not.

The fact that the RMS deviation is significantly smaller for carrier type approaches than for the others, is probably indicative of the tighter limits controlling this type of approach.

Fig.9 is a straight graphical comparison between the actual approach speeds and those corresponding to the proposed minimum standard of speed stability. This figure shows that the differences are roughly constant over the range 80-160 knots.

Fig.9 also indicates that consideration of drag effects can materially improve the prediction of approach speeds in the higher range as compared with estimates based, for example, on the margin over $V_{C_{L_{max}}}$ (see section 5.4 below).

The following table, extracted from Table 2, summarises the differences between the actual approach speeds and the speeds at which speed stability would have fallen to the proposed limiting values. Differences are positive when the actual speed exceeds the limiting value.

		Carrier approach	Airfield approach	Instrument approach
Aircraft limited by speed stability on approach	Mean difference, knots R.M.S. difference, knots	+2.7 6.2	+2.0 9.1	- -
Aircraft limited by other factors on approach	Mean difference, knots R.M.S. difference, knots	+4.2 5.7	+9.2 10.5	- -
All 19 aircraft together	Mean difference, knots R.M.S. difference, knots	+3.6 5.9	+5.9 9.9	+3.3 12.0

The probable difference in any single example (roughly, $^2/_3$ of the R.M.S. difference) is therefore ±4 knots for a carrier approach, ±6 knots for an airfield approach or ±8 knots for an instrument approach. These data are based on 8 examples of carrier approach, 15 airfield approaches, and only 3 instrument approaches.

5.4 Additional data on speed margin over $V_{C_{L_{max}}}$

Although strictly outside the scope of the present paper, the data available permit some observations on margins of speed over $V_{C_{L_{max}}}$. Fig.8 is drawn on a scale of $V_A/V_{C_{L_{max}}}$ and the points representing approach speeds which were stated to be limited by proximity to the stall (Table 1) are shown by appropriate symbols. From this it can be suggested that carrier type approaches should be at or above 1.15 $V_{C_{L_{max}}}$ while the corresponding factors for airfield and instrument approaches should be 1.20 and 1.30. This last figure is, however, not well-founded.

Accepting these limits, we can then show on Fig.9 to what extent the reductions in approach speed, which may have been feasible on the basis of drag effects alone, may have been achieved, so far as danger of stalling is concerned. Each point on this figure has been linked to the corresponding limiting speed in terms of $V_{C_{L_{max}}}$. It appears that in some cases, reductions in speed towards the limiting values determined solely from consideration of drag effects might have been possible had it not been for factors other than drag effects or stalling limitations which must also be considered.

5.5 Effect of boundary layer or circulation control

No special problems are apparent, so far as speed stability is concerned, on those few aircraft examined which were fitted with some form of boundary layer or circulation control system. Fig.3 shows the effect of these high lift devices on drag/lift ratio and Fig.6 shows their effect on the parameter $(C_D/C_L - dC_D/dC_L)$. On aircraft I, application of the high lift system (resulting in aircraft R) produces a more rapid deterioration in speed stability below minimum drag speed, rather similar to that shown for the conventional aircraft M (Fig.5). These two aircraft were tested at different establishments, and possibly this accounts for some of the difference in the pilot's estimate of tolerable approach speeds. The stalling speed of aircraft R is much higher than that of aircraft M, however, and the speed margins over $V_{C_{L_{max}}}$ for airfield approaches are not much different for the two aircraft.

5.6 Artificial means of improving speed stability

The obvious method of improving the speed stability is the use of an automatic throttle control system. In its simplest form, this produces changes in thrust proportional to changes in airspeed from some pre-selected datum.

Then, referring to equations 2 and 3, it will be seen that we can always produce positive speed stability (i.e. a subsidence of any speed error) if the inequality of equation 3 is satisfied by artificially increasing C_{AS} - the drag coefficient equivalent to the thrust/speed variation.

Suppose, on a certain aircraft, it is desired to reduce the approach speed to a value for which, in the absence of an automatic throttle, the parameter F would have a value of F_B, say. F_B will be positive if this new speed is below the normal minimum drag speed. The automatic throttle may be treated as a device designed to reduce F_B by an amount ΔF so as to provide speed stability, i.e. ΔF will be greater than F_B.

The new value of $F = F_B - \Delta F$ is achieved by increasing C_D to $(C_D + C_{AS})$ where

$$C_{AS} = \tfrac{1}{2} \gamma M^2 C_L \Delta F$$

$$= \frac{\Delta F}{P} W/S \tag{14}$$

The definition of C_{AS} in equation 2 may be re-written in terms of the actual variation of thrust with speed

$$C_{AS} = - \frac{1}{\rho V S} \frac{dT}{dV} \tag{15}$$

and the minimum required thrust/airspeed variation becomes

$$\frac{dT}{dV} = - \rho V W \Delta F/p . \tag{16}$$

A device of this type can in fact be used to produce a considerably better standard of speed stability than is indicated by the limiting value of F equal to -2.0, and a value nearer -20 is probably more appropriate. It has been found in practice that this higher thrust/airspeed variation is beneficial in quickly damping out the transient variations in airspeed resulting from corrections to the flight path.

It may be noted in passing that the variation of thrust (for level flight) with airspeed given in equation 8 is such as to produce only neutral speed stability. Equation 16 allows the stability to be improved to any desired level. For example, the value of -20 for the parameter F, which is readily achievable, means that a small speed disturbance would decay to half amplitude in 300 yards, i.e. in about 4-5 seconds.

6 OTHER EFFECTS

One possibly significant factor, not so far considered, is the thrust response to throttle movement. On a normal approach, the pilot will attempt to do what the automatic throttle control system is designed to do for him. His success in this direction must depend on thrust response, and one would expect him to be more tolerant of speed instability if, in fact, he could vary the thrust quickly and accurately. For example, the ratio of the thrust used on the approach to the maximum available thrust might have an optimum value. A low value of the ratio means that the engine is operating in a region where response may be poor, while a value approaching unity means that there is little in hand for correcting for a loss in airspeed.

For the present, no data are available on the possible significance or likely value of this ratio.

7 CONCLUSIONS

The minimum comfortable approach airspeed of an aircraft is determined from consideration by the pilot of the way in which a number of aspects of the behaviour of the aircraft deteriorate as the speed is reduced. Among the more important of these aspects is the phenomenon of speed stability, and in about half of the 19 cases examined in this paper, speed stability (or lack of it) is stated by pilots to be the factor which prevents the use of a lower approach speed.

The parameters determining speed stability have been examined and limiting values have been suggested which define speeds below which it is expected that speed stability would have been inadequate. For these aircraft whose approach speeds were, in fact, limited by marginal speed stability, these proposed limiting speeds average 2-3 knots lower than those actually used. Taking all 19 aircraft together, the average difference is about 5 knots, but the R.M.S. difference is between 6 and 12 knots, depending on the type of approach.

Automatic throttle control is in theory an effective method of improving speed stability to an acceptable level, particularly on instrument approaches.

To summarise, it is shown that speed stability can limit approach speeds, and that the level of stability must be taken into account in estimating these speeds. This process can materially improve the prediction of approach speeds for the latest types of aircraft.

The study of the relation between approach speed and lift-drag characteristics should continue, as more data become available. The importance of thrust response to throttle movements should also be examined.

LIST OF SYMBOLS

a	lift curve slope, C_L/radian
C_{AS}	drag coefficient equivalent of variable thrust (equation 2)
C_D	drag coefficient
C_L	lift coefficient
$C_{L_{max}}$	maximum available lift coefficient

F	speed stability parameter (equation 10)
ΔF	reduction in F produced by automatic throttle
g	gravity constant, ft/sec^2
M	Mach number
n	pitch attitude change per unit change in steady glide angle, $d\theta/d\gamma$
p	ambient pressure, lb/ft^2
q	dynamic pressure $\frac{1}{2}\rho V^2$, lb/ft^2
S	wing area, sq ft
s	distance along flight path, ft
T	engine thrust, lb
T_{RL}	thrust required for level flight, lb
t	time, sec
u	change in airspeed, ft/sec
u_o	initial value of u, ft/sec
V	airspeed, ft/sec or knots, as indicated
V_A	approach airspeed, ft/sec or knots, as indicated
$V_{C_{L_{max}}}$	airspeed in unaccelerated flight at $C_{L_{max}}$, ft/sec or knots
V_{MD}	minimum drag speed, knots
W	landing weight, lb
W/S	wing loading, lb/ft^2
γ	glide angle, radians
ρ	air density, $slugs/ft^3$
θ	pitch attitude of aircraft datum, radians
τ	time constant of divergence or subsidence, secs

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4	Neumark, S.	Private communication.

TABLE 1

Basic data on 19 aircraft

Aircraft type	Speed at max. lift $V_{C_{L_{max}}}$, kts	Min. drag speed V_{MD}, kts	$\frac{V_{MD}}{V_{C_{L_{max}}}}$	Min. comfortable approach speeds, V_A, for:- Carrier type landing V_{A_1} kts	Airfield approach V_{A_2} kts	Instrument approach V_{A_3} kts	$V_A/V_{C_{L_{max}}}$ (1)	(2)	(3)	V_A/V_{MD} (1)	(2)	(3)	$\left(\frac{C_D}{C_L} - \frac{dC_D}{dC_L}\right)$ at V_A (1)	(2)	(3)	$\frac{-2}{\gamma M^2}\left(\frac{C_D}{C_L} - \frac{dC_D}{dC_L}\right) = F$, at V_A (1)	(2)	(3)	Reasons for limiting approach speed
A	83	117	1.41		120			1.44			1.02			+0.030			-1.27		Drag effects
B	86	111	1.29		110			1.28			0.99			-0.011			+0.63		"
C	84	105	1.25	101	107		1.20	1.28		0.96	1.02		-0.030	+0.014		+ 1.90	-0.63		"
D	74	98	1.33	85	90		1.15	1.22		0.86	0.92		-0.127	-0.091		+11.00	+7.00		"
E	111	172	1.55		160			1.44			0.93			-0.089			+2.12		"
F	88	164	1.86		135			1.54			0.83			-0.141			+4.87		"
G	119	150	1.26	128			1.08			0.86			-0.079			+ 2.96			"
H	96	140	1.46			145			1.51			1.03			+0.031			-0.85	"
I	89	105	1.18		118			1.33			1.13			+0.046			-2.12		"
J	85	89	1.05			115			1.35			1.28			+0.065			-2.96	Control response
K	78	100	1.28	87	101		1.12	1.30		0.87	1.01		-0.043	+0.002		+ 3.60	0		Lateral control
L	93	109	1.17	104			1.12			0.96			-0.047			+ 2.75			Stalling
M	83	100	1.21	91	95		1.10	1.15		0.91	0.95		-0.100	-0.027		+ 7.60	+1.90		"
N	100	121	1.21	115	130		1.15	1.30		0.95	1.07		-0.100	+0.086		+ 4.65	-3.18		Lateral stability
O	86	106	1.23		105	130		1.22	1.51		0.99	1.23		-0.003	+0.094		+0.21	-4.00	Stalling
P	85	91	1.07		110			1.35			1.26			+0.070			-3.60		Attitude and trim
Q	85	114	1.34		109			1.28			0.96			-0.021			+1.06		Stalling and long. control
R	93	100	1.08		111			1.19			1.10			+0.053			-2.75		"
S	78	90	1.15	90	100		1.15	1.23		1.00	1.11		0	+0.052		0	-3.18		Attitude and trim

NOTE: Aircraft Q and R are modifications of Aircraft I } With boundary layer or circulation control.
Aircraft S is a modification of Aircraft P

TABLE 2

Comparison between actual and proposed limiting approach speeds

Aircraft type	Reasons for limiting approach speed	Min. comfortable approach speeds, V_A, for:- Carrier type landing V_{A_1}, knots	Airfield approach V_{A_2}, knots	Instrument approach V_{A_3}, knots	Limiting approach speeds, from speed stability considerations only V'_{A_1}	V'_{A_2}	V'_{A_3}	$V_A - V'_A$ (1)	(2)	(3)
A	Speed stability (i.e. drag effects)		120			114			+ 6	
B			110			107			+ 3	
C		101	107		96	101		+ 5	+ 6	
D		85	90		90	96		- 5	- 6	
E			160			161			- 1	
F			135			147			-12	
G		128			116			+ 8		
H				145			158			-13
I			118			100			+18	
		For aircraft limited by drag effects			Mean difference, knots R.M.S. difference, knots			+ 2.7 6.2	+ 2.0 9.1	-13 13.0
J	Factors other than drag effects			115			103			+12
K		87	101		81	91		+ 6	+10	
L		104			99			+ 5		
M		91	95		92	95		- 1	0	
N		115	130		114	118		+ 1	+12	
O			105	130		98	119		+ 7	+11
P			110			86			+14	
Q			109			106			+ 3	
R			111			98			+13	
S		90	100		80	85		+10	+15	
		For aircraft limited by factors other than drag effects			Mean difference, knots R.M.S. difference, knots			+ 4.2 5.7	+ 9.2 10.5	+11.5 11.5
		Results for all 19 aircraft together			Mean difference, knots R.M.S. difference, knots			+ 3.6 5.9	+ 5.9 9.9	+ 3.3 12.0

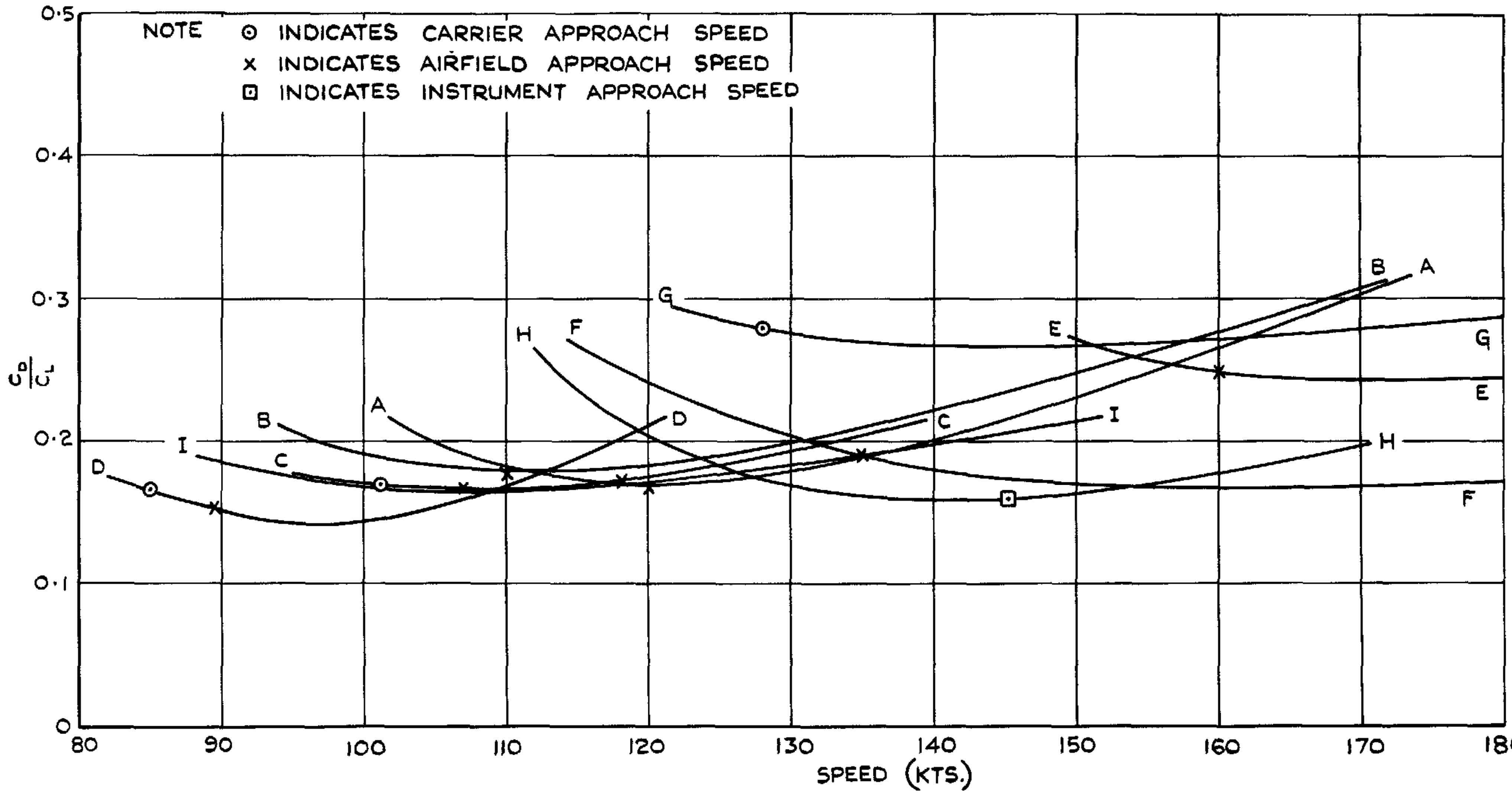


FIG. I. VARIATION OF DRAG / LIFT RATIO WITH SPEED FOR AIRCRAFT WHOSE APPROACH SPEEDS WERE LIMITED BY DRAG EFFECTS.

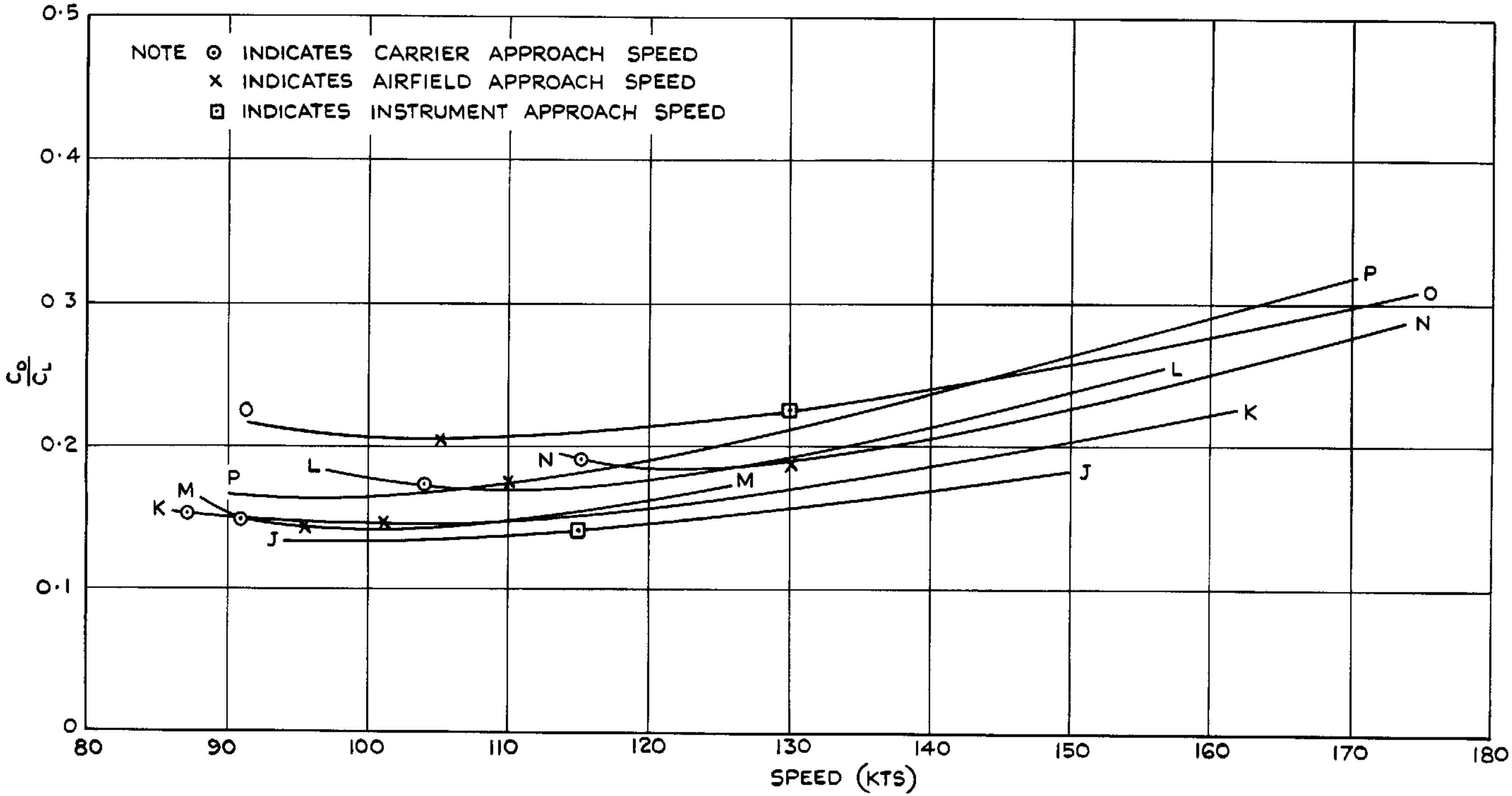


FIG. 2. VARIATION OF DRAG / LIFT RATIO WITH SPEED FOR AIRCRAFT NOT LIMITED BY DRAG EFFECTS ON THE APPROACH.

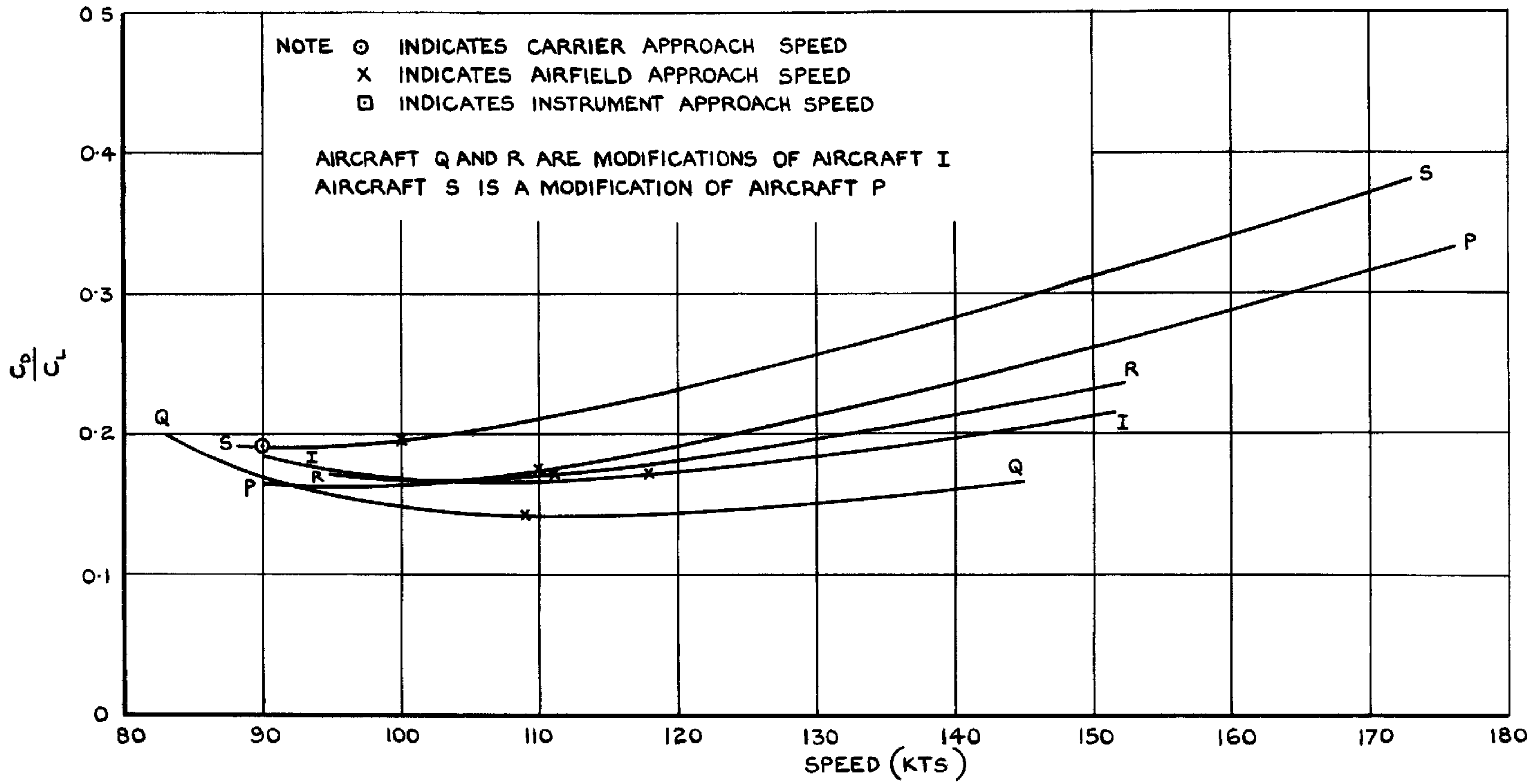


FIG. 3. VARIATION OF DRAG / LIFT RATIO WITH SPEED FOR AIRCRAFT WITH AND WITHOUT BOUNDARY LAYER OR CIRCULATION CONTROL SYSTEMS.

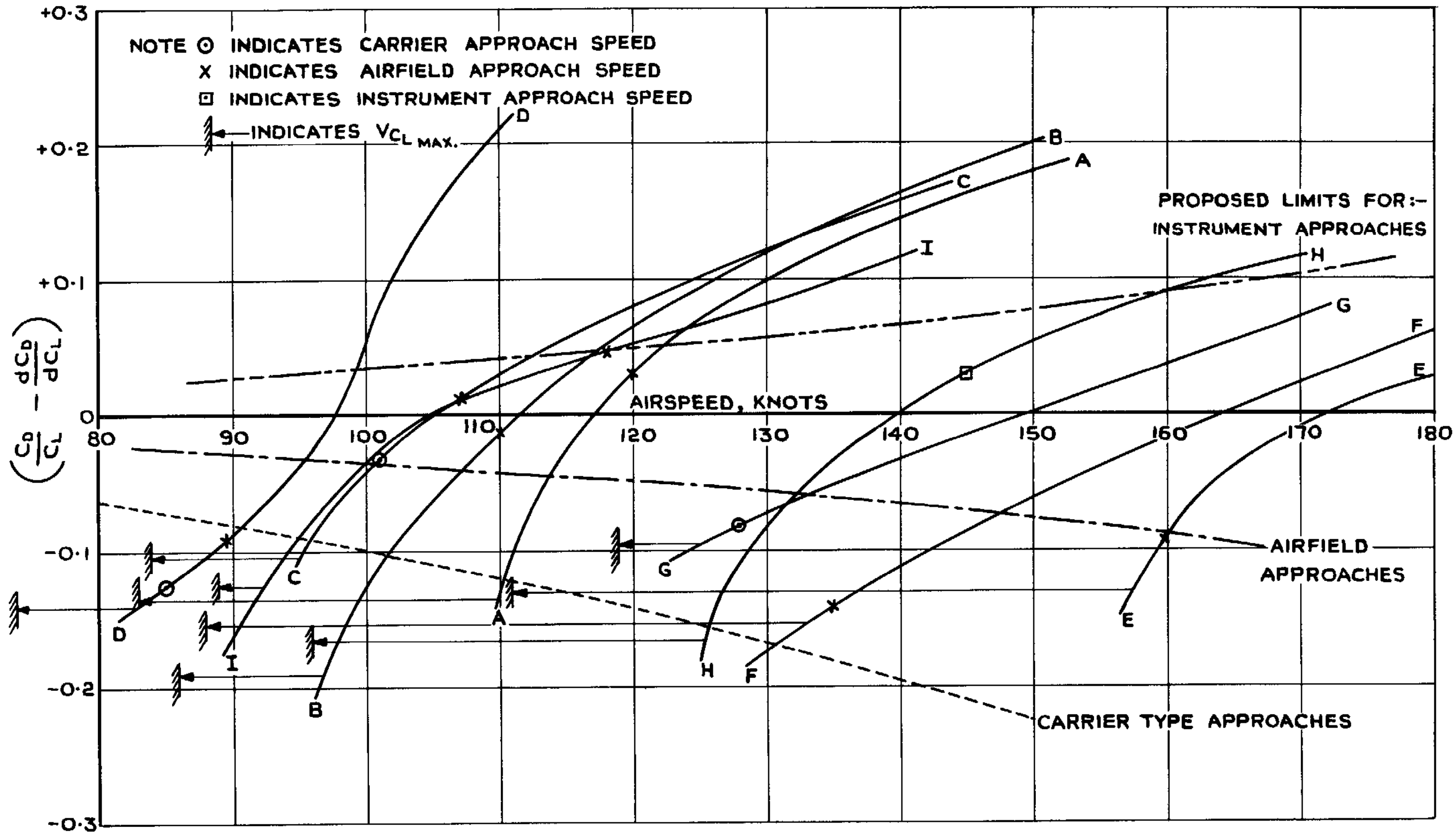


FIG. 4. VARIATION OF $(C_D/C_L - dC_D/dC_L)$ WITH SPEED FOR AIRCRAFT WHOSE APPROACH SPEEDS WERE LIMITED BY DRAG EFFECTS.

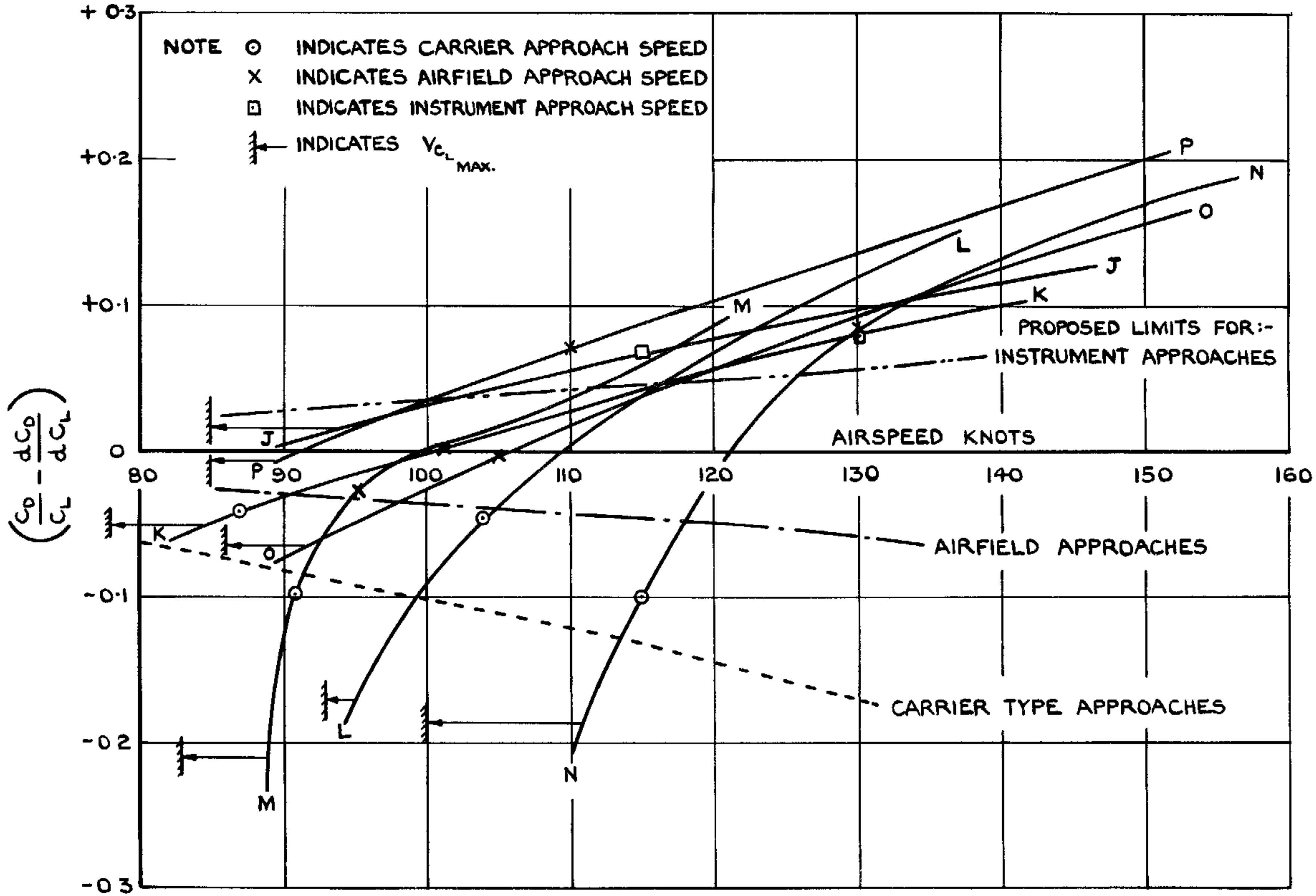


FIG.5. VARIATION OF $\left(C_D/C_L - dC_D/dC_L\right)$ WITH SPEED FOR AIRCRAFT NOT LIMITED BY DRAG EFFECTS ON THE APPROACH.

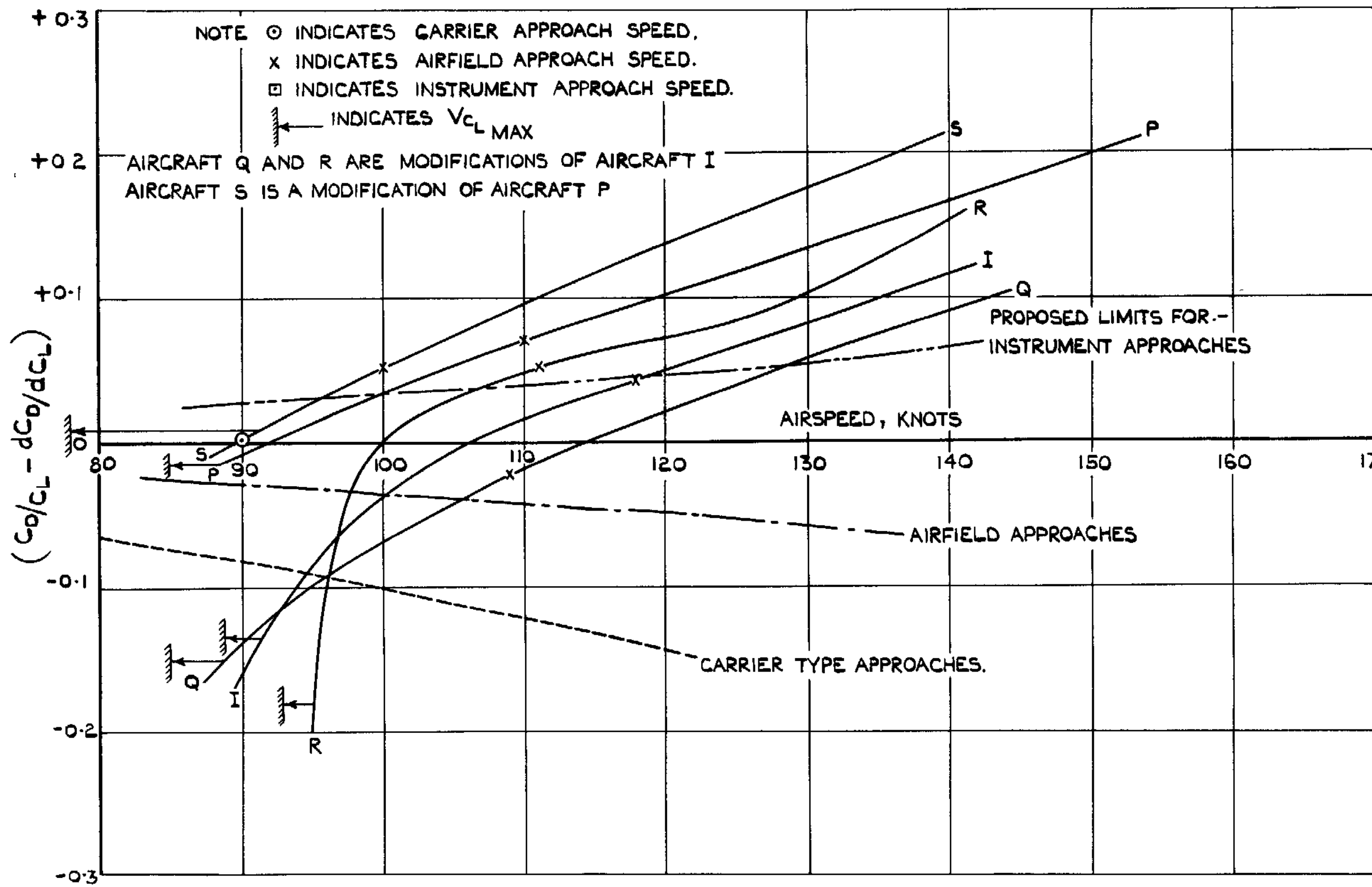
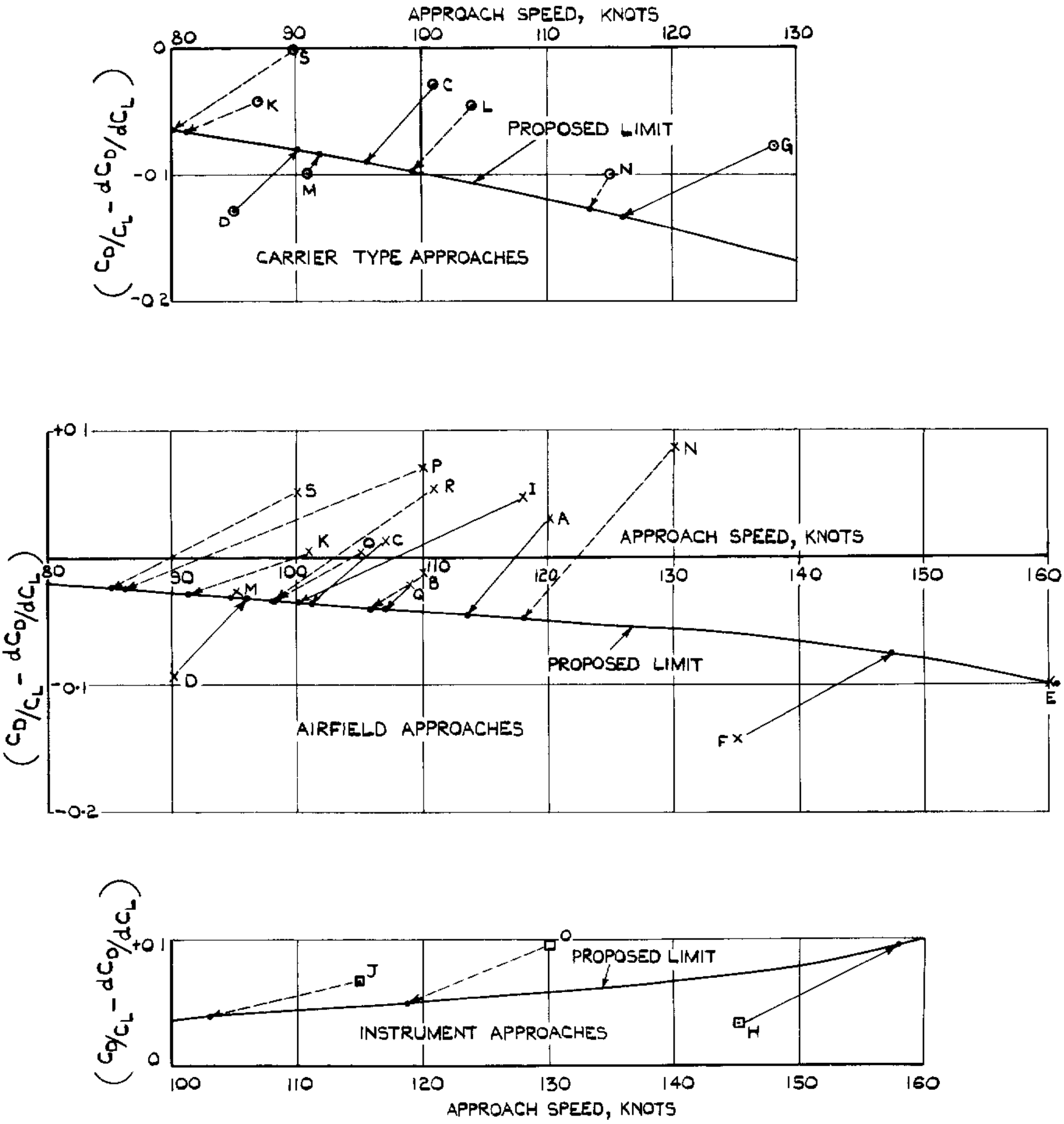


FIG.6. VARIATION OF $(C_D/C_L - dC_D/dC_L)$ WITH SPEED FOR AIRCRAFT WITH BOUNDARY LAYER OR CIRCULATION CONTROL SYSTEMS.



NOTE ① EACH POINT REPRESENTING AN ACTUAL APPROACH CONDITION IS LINKED TO THE CORRESPONDING LIMITING CONDITION BY A FULL LINE IF ACTUAL APPROACH SPEED WAS LIMITED BY DRAG EFFECTS OR BY A DOTTED LINE IN OTHER CASES

② LETTERS ALONGSIDE SYMBOLS REFER TO AIRCRAFT TYPE (TABLE I).

③ LIMITING SPEEDS DERIVED FROM FIGS 4, 5 & 6.

FIG. 7. ACTUAL AND LIMITING APPROACH SPEEDS, AND VALUE OF STABILITY PARAMETER $(C_D/C_L - dC_D/dC_L)$.

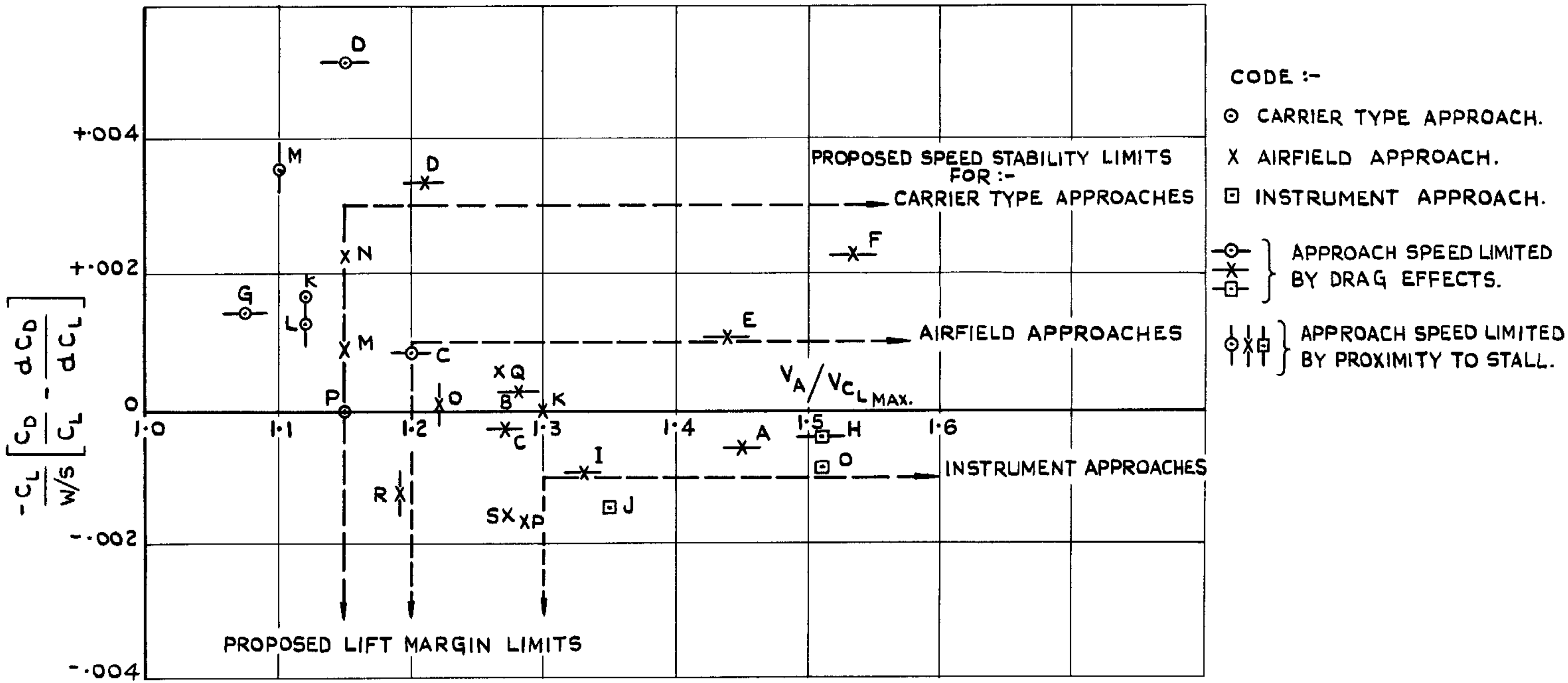


FIG.8. DERIVATION OF PROPOSED LIMITS DUE TO BOTH DRAG EFFECTS AND TO MARGIN OVER $V_{C_{L\,MAX}}$.

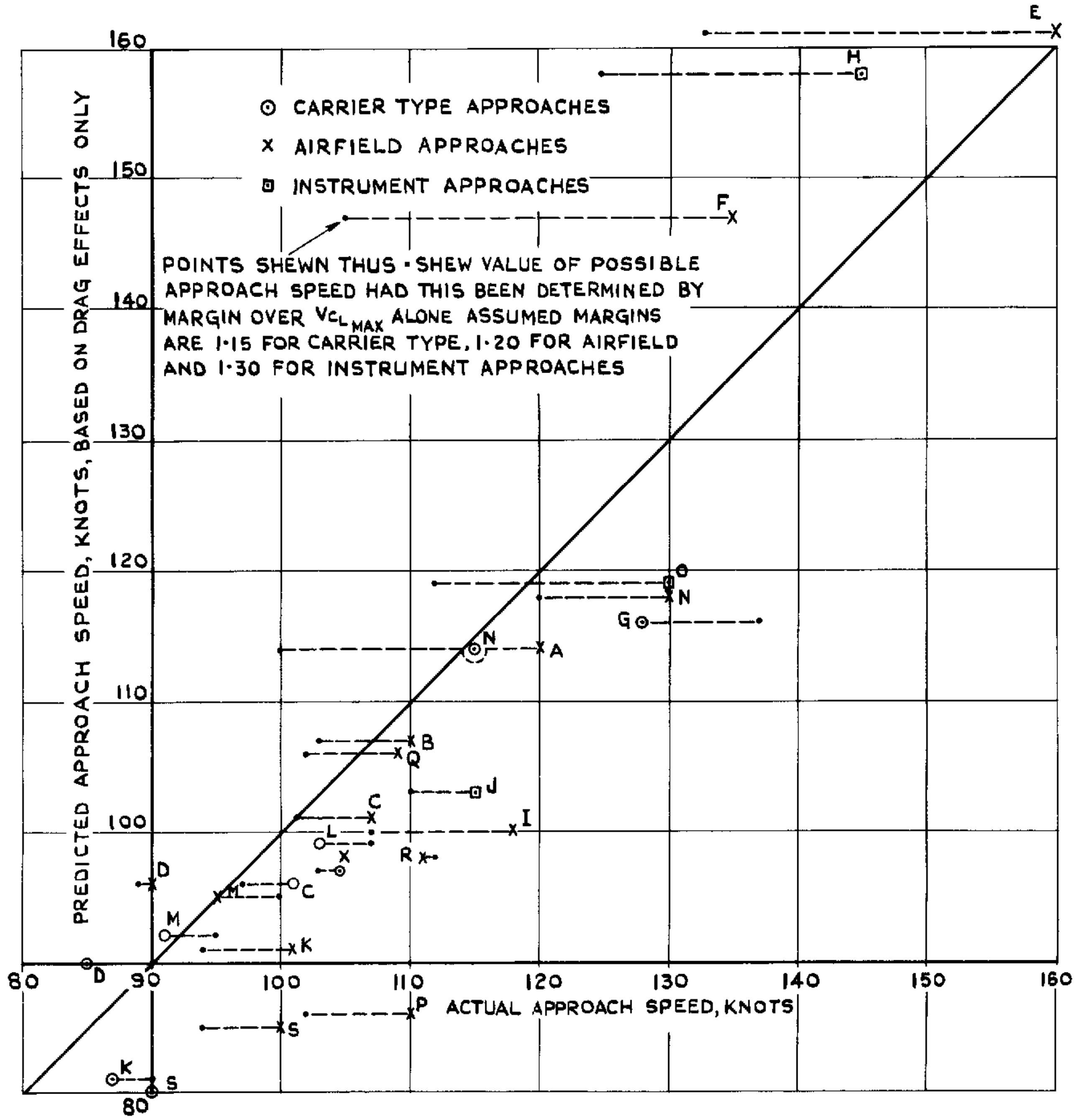


FIG.9. COMPARISON BETWEEN ACTUAL APPROACH SPEEDS AND SPEEDS PREDICTED FROM DRAG EFFECTS ONLY.

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ROYAL AIRCRAFT ESTABLISHMENT

The Aerodynamic Effects of Aspect Ratio on Control Surface Flutter

by

H. Hall
and
E. W. Chapple

SUMMARY

The report describes a series of low speed flutter tests to obtain a direct measurement of the aerodynamic effects of aspect ratio on wing-aileron flutter. The tests were made on rigid wings fitted with full span ailerons, the wings having root flexibilities in roll and pitch. Provision was made for massbalancing the ailerons. Some general conclusions are drawn concerning the effects of aspect ratio and massbalance on control surface flutter.

LIST OF CONTENTS

LIST OF ILLUSTRATIONS

1 Introduction

The tests described in this paper are the last of a series[1,2,3,4] using a technique, described in reference 1, whereby the aerodynamic effects of aspect ratio on flutter can be isolated. The procedure is to use wings that are virtually rigid, but have root flexibilities. By suitable adjustment of the inertia and elastic characteristics of the families of wings considered, it is possible to make any change in flutter speed between individual wings of these families dependent only on the aerodynamic effect of aspect ratio.

The technique was applied in this instance to a family of unswept, untapered wings, each wing having a full span aileron whose chord was 0.3 of that of the wing. The influence of massbalance on the control surface flutter characteristics was also investigated.

2 Experimental details

2.1 Description of the wings and mounting rig

All wings were of solid homogeneous construction, being made of spruce. Each wing was fitted with a full span aileron, whose chord was 0.3 of that of the wing and which operated on a plain bearing hinge. There was no stiffness between the wing and aileron. The wing section used throughout was R.A.E.101.

The rig allowed wing freedoms in modes of linear flexure (roll) and uniform pitch. A further degree of freedom was allowed, that of aileron rotation about its hinge line. The wing root was 0.075 span above the roll axis, and the pitching axis was 0.35 chord aft of the leading edge. Torsion bars of adjustable length provided the required stiffnesses, and sliding weights enabled the roll and pitch inertias to be adjusted.

The wing mounting was designed so that its product of inertia between roll and pitch was zero, but the mounting contributed to the direct inertias of the wings so that means of adjusting them were required. The moments of inertia of the rig (wing and mounting) about the axis of roll and pitch were adjusted by means of the sliding weights to vary as s^3 in roll and s in pitch, where s is the distance from roll axis to wing tip. Furthermore the wings were designed so that the products of inertia between roll and pitch and between roll and aileron rotation varied as s^2 and the product of inertia between pitch and aileron rotation and the moment of inertia of the aileron about its hinge line varied as s. The inertia values are given in the table accompanying Fig.1 together with the dimensions of the wings.

2.2 Massbalance arrangement

A massbalance rider was attached to a carrier arm at the outboard end of each aileron (Fig.2) and the massbalance contribution to the various inertias was such that the dependence of the inertias on the above functions of s (Section 2.1) was preserved. This was achieved by making all spanwise dimensions of the massbalance system vary as s, other dimensions being constant for all the wings. The carriers were made of steel and the riders of lead; the structural details of the massbalance system are given in the table accompanying Fig.2.

The massbalance system was effective in balancing out the dynamic cross inertia between wing roll and aileron rotation. When the c.g. of the rider was located 1.06" forward of the aileron hinge line the aileron was dynamically balanced in roll. However, when the rider was situated in its furthest forward position on the carrier, the rider c.g. then being 1.72" forward of the aileron hinge line, the cross inertia between wing pitch and aileron rotation was only reduced by 21% of its initial value.

The variation of massbalance conditions covered a range from 45% to 80% static balance of the aileron. The addition of mass balance had a pronounced effect on the pitching moment of inertia of the aileron. When the carrier only was added the inertia was increased by 15% of its basic value and when the rider was added at its furthest forward position it was increased by a further 73%.

2.3 Wind tunnel measurements

The tests were conducted in the 5 ft diameter open jet tunnel. All wings were mounted vertically above a reflector plate, to simulate the symmetric flow condition. The wing aspect ratios ranged from 2.0 to 6.0 being defined as 2 s/c where c, the chord of the wing, was constant for the whole series.

The wings were set up by adjusting the torsion bars so that for all the wings, the frequencies of the corresponding modes (with aileron fixed to the wing) were the same. The natural frequency of the wings in roll was 3.1 c.p.s. and in pitch 9.6 c.p.s. These frequencies were measured with the massbalance rider placed on the carrier with its c.g. 0.82" forward of the aileron hinge line.

For a particular massbalance condition i.e. aileron c.g. position, the various wings are so related that the flutter equations are identical apart from the aspect ratio effects on the aerodynamic coefficients. The fact that the natural frequencies of the wing in roll and pitch were measured at a particular massbalance condition does not imply that the relation holds only for this condition. The relation holds for all massbalance conditions but the equations for each massbalance condition will be different.

The tests were made with the aileron free and with various massbalance conditions. Readings were taken (1) with only the carrier fitted and (2) with the rider fitted on the carrier at various positions along the arm. For each of these conditions measurements were made of flutter characteristics for the binary types of flutter wing roll-aileron rotation and wing pitch aileron rotation and of the ternary wing roll-wing pitch-aileron rotation.

Flutter speeds and frequencies were measured for each wing, the speed being that at which the oscillation just died out as the tunnel speed was reduced. As some of the flutter speeds were unusually low and below the accurate calibrated value for the tunnel, measurements of all speeds were made using a Chattock gauge.

3 Results

The results of the wind tunnel tests are plotted in Figs 3-8. In Figs 3-6 flutter speed and frequency are plotted against the reciprocal of aspect ratio and in Figs 7 and 8 flutter speed is plotted against massbalance position for each of the wings in turn.

The investigation was divided into three distinct parts, depending on the degrees of freedom of the system that were allowed. These were

(1) Wing roll and aileron rotation

(2) Wing pitch and aileron rotation

(3) Wing roll and pitch and aileron rotation

3.1 Wing roll-aileron rotation

The variation of flutter speed and frequency with aspect ratio for this type of flutter is shown in Fig.3. It was found that the flutter was quite mild and could be allowed to continue right through its speed range so that an upper bound to the flutter was obtained. Flutter frequencies are only plotted for two massbalance conditions to avoid confusion in the figure.

It was found that the upper critical speeds were sensitive to damping in the aileron degree of freedom. The aileron amplitude near the upper critical speed is extremely small and the aileron inertia at similar amplitudes in the wind off condition is insufficient to overcome even the small amount of friction present in the aileron bearing. Too much significance should not therefore be attached to these upper critical speeds; the upper bounds are indicated by a broken line to indicate the uncertainty about the absolute values.

It can be seen that for certain massbalance conditions, as the aspect ratio increases there is a limit beyond which flutter of this type does not occur. The tests indicate that for increasing massbalance the limiting aspect ratio decreases. The limiting aspect ratio is slightly less than 4 when the aileron is 59% statically balanced decreasing to just greater than 3 as the balance rises to 64%. No nose to the flutter speed curve was found when the aileron static balance was less than 59%, the trend of the results indicates that limiting finite aspect ratios should exist but no values can be assigned to them.

It was considered that the extremely low Reynolds number at which the tests were conducted (between 3.5×10^4 and 24.5×10^4) could be producing some unwanted aerodynamic effect. The roll-aileron rotation flutter tests were accordingly repeated with transition wires fitted to the wing, this had the effect of increasing the width of the flutter band and making the flutter more violent. The general shape of the flutter speed curve is, however, unaltered and in particular for the higher values of mass balance a limiting aspect ratio exists above which flutter does not occur.

3.2 Wing pitch-aileron rotation

The variation of flutter speed and frequency with aspect ratio for this type of flutter is shown in Fig.4. Flutter speed increases linearly with decreasing aspect ratio; the frequency remains approximately constant as the aspect ratio increases from 2 to 4 but for larger values gradually decreases. In the range of aspect ratios examined the flutter speed can be expressed by a relation of the form $V = V_o\, f(A)$, where V_o is the extrapolated experimental value for the two-dimensional case* and $f(A)$ is a function of aspect ratio. The particular form to be assigned to the function f depends on the massbalance condition and several values are given in the figure. However, for the range of massbalance considered a reasonable average value of $f(A)$ is $f(A) = 1 + 4.25/A$.

Wing pitch-aileron rotation type flutter occurs at higher speeds than the other type for all the wings tested.

3.3 Wing roll-wing pitch-aileron rotation

When both of the wing degrees of freedom were allowed together with aileron rotation, two forms of flutter were obtained closely resembling the

---

* It should be noted that extrapolation of the results beyond those for the wings of highest aspect ratio tested may not be justified.

types of binary flutter considered above. The mainly wing roll-aileron rotation type flutter was excited first, and then at higher speeds whilst this type of oscillation persisted, a further disturbance would excite the wing pitch-aileron rotation type flutter, which then became dominant. The variations of flutter speed and frequency of these types of flutter with aspect ratio are shown in Fig.5 and 6.

When the rider was fitted so that its c.g. was 1.12" forward of the aileron hinge line an instability of the roll-aileron rotation type was obtained involving large wing amplitudes. It was impossible to ascertain whether this was flutter or not as the rig immediately "hammered" against the amplitude limit stops. This phenomenon was not noticed for the binary system. Otherwise the flutter speeds and frequencies for the wing roll-aileron rotation type flutter are practically the same as the corresponding binary ones. The same doubts exist about the accuracy of the upper critical speeds for this type of flutter as were mentioned in connection with the binary flutter.

The results for the wing pitch aileron rotation type flutter are shown in Fig.6 and are very similar to those for the corresponding binaries. Flutter speeds for the ternary are greater than those for the binary having the same massbalance conditions. The slopes of the lines representing the increase in flutter speed with decrease of aspect ratio, decrease as the massbalance is reduced and they are greater than those of the corresponding binary case.

The regions in which the two types of flutter are possible are overlapping for certain massbalance conditions, and it is possible to have both types occurring at a particular speed. For an aircraft, only the lower bound is, in general, significant and there will be a transition from one form of flutter to another, the transition point being at a particular massbalance condition (corresponding to the nose of the wing roll-aileron rotation type flutter (Figs 7 and 8) of the tests) which depends on the aspect ratio of the wing in question.

3.4 Comparison with theory

The fact that a decrease of aspect ratio could increase the danger of a mild aileron flutter has been noticed previously by Jordan[5] in some flutter calculations on a similar system to this. To a certain extent this is confirmed by these tests i.e. for certain massbalance conditions a limiting aspect ratio exists above which flutter will not occur. Flutter calculations for the roll-aileron rotation binary using two dimensional derivatives[6] do not give agreement with the trends indicated by these measured results. These calculations show that a flutter speed exists for the infinite aspect ratio wing for all massbalance conditions between that in which no rider is carried and that in which the rider is 0.82" forward of the aileron hinge line. For more forward massbalance positions no flutter speed exists for the two dimensional case.

Attempts to predict the flutter characteristics for the finite aspect ratio wings using two dimensional derivatives factored by the previously determined aspect ratio correction[2] for the main surface and the full values for the control surface, gave generally poor agreement for the lower critical speeds and the upper critical speeds were very much lower than the measured ones. (There is doubt about the accuracy of the measured upper critical speeds though). The upper and lower bounds of the flutter speed curve are roughly parallel with the aspect ratio axis. The theoretical results obtained for the no rider case are indicated in Fig.3.

Speculation arises as to what is the cause of the discrepancy between calculation and practice. The introduction of structural damping into the flutter equations will eventually eliminate flutter in the infinite aspect ratio case but the amount of damping required in the aileron degree of freedom to achieve this is prohibitively large and such an amount of damping is certainly not present in practice.

Flutter calculations for the binary pitch-aileron rotation type flutter gave very poor agreement with the extrapolated experimental values for the infinite aspect ratio case and the theoretical work was not continued further than this.

4 Conclusions

Two types of ternary control surface flutter were characteristic of the system considered here, one in which the main motion was roll of the wing and control surface rotation and a second in which the main surface motion was predominantly pitch. For a particular massbalance condition both types exhibit an increase of flutter speed with decreasing aspect ratio, the increase being slight for the first type. A linear increase was found for the second type of flutter, which could be expressed in the form $V = V_o f(A)$, V_o being the extrapolated value for the two dimensional speed and A the aspect ratio, which was valid over the range of aspect ratios tested.

Some confirmation is provided by these tests of an earlier theoretical conclusion[5] that a decrease of aspect ratio can increase the probability of encountering a region in which a mild aileron flutter occurs. The limiting aspect ratio below which flutter occurs depends on the amount of massbalance carried by the control. Increase of percentage static balance has quite a marked effect on the first type of flutter, the flutter eventually being eliminated; for the wing of aspect ratio 2 this occurs at 70% static balance whilst for that of aspect ratio 4 it occurs at 59%. The effect on the second type of flutter is a gradual increase in flutter speed with increasing massbalance.

It is reasonable to expect that the results obtained will be applicable qualitatively to control surface flutter in general, where the aileron will not be free as in this case.

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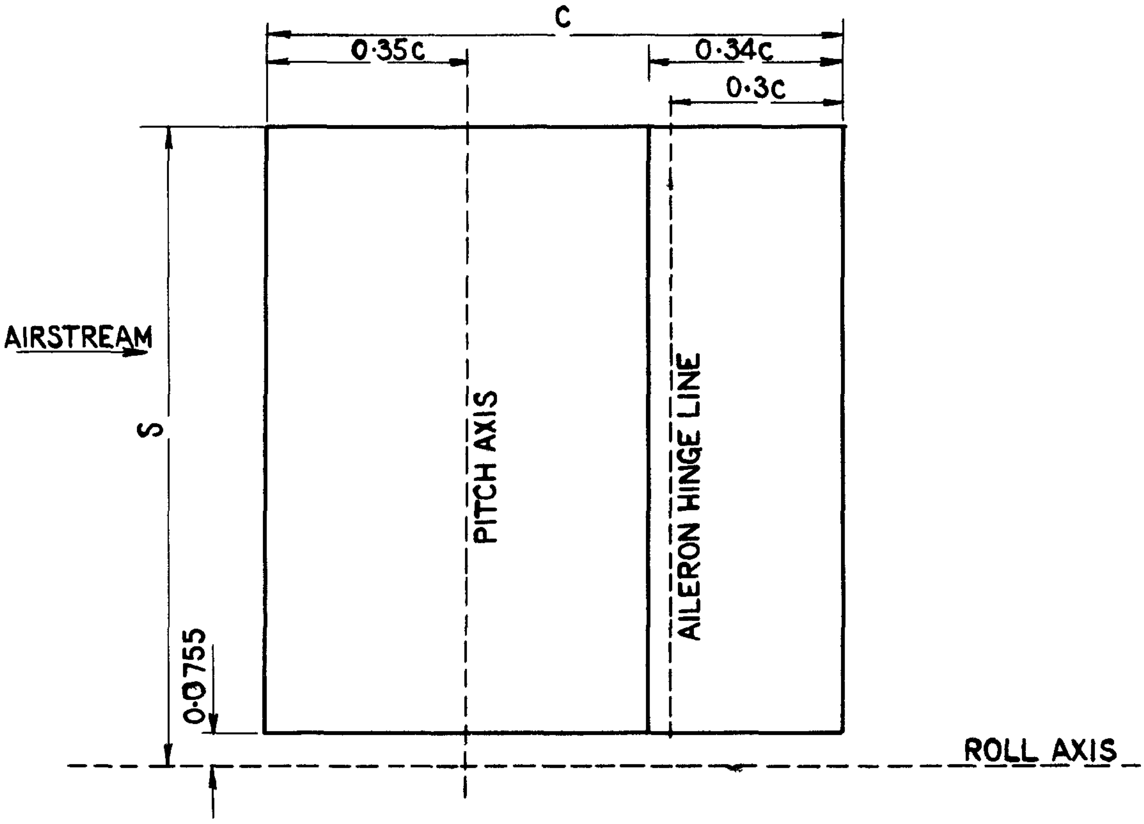
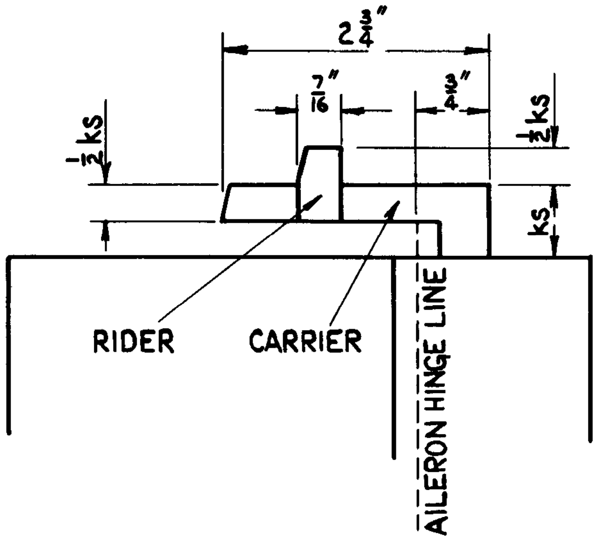


WING CHORD = 0·5 FT.
WING THICKNESS/CHORD = 0·15
WING SECTION = R.A.E. 101

WING Nº	WING SPANS INS.	ASPECT RATIO	INERTIAS WING & MOUNTING - LB. IN². ROLL INERTIA	ROLL-PITCH INERTIA	PITCH INERTIA	INERTIAS AILERON - LB. IN². ROLL-ROTATION INERTIA	PITCH-ROTATION INERTIA	ROTATIONAL INERTIA
1	6	2·00	7·90	0·570	0·794	0·0870	0·0812	0·0251
2	7	2·33	12·73	0·722	0·924	0·1136	0·0922	0·0294
3	8	2·67	18·99	1·172	1·060	0·1532	0·1076	0·0335
4	9	3·0	26·70	1·237	1·195	0·2088	0·1274	0·0376
5	10	3·33	37·68	1·547	1·326	0·2428	0·1357	0·0417
6	12	4·00	64·83	2·030	1·600	0·3610	0·1665	0·0501
7	15	5·00	122·7	3·064	1·996	0·5768	0·2115	0·0627
8	18	6·00	214·8	5·004	2·401	0·8074	0·2485	0·0749

FIG. 1. GEOMETRICAL AND STRUCTURAL DETAILS OF THE WINGS.



k = 0·02083

WING Nº	CARRIER WEIGHT x 10^2 LB.	RIDER WEIGHT x 10^2 LB.
1	1·777	0·617
2	2·090	0·672
3	2·372	0·787
4	2·650	0·974
5	2·972	1·080
6	3·574	1·226
7	4·460	1·565
8	5·368	1·918

FIG. 2. THE AILERON MASSBALANCE SYSTEM.

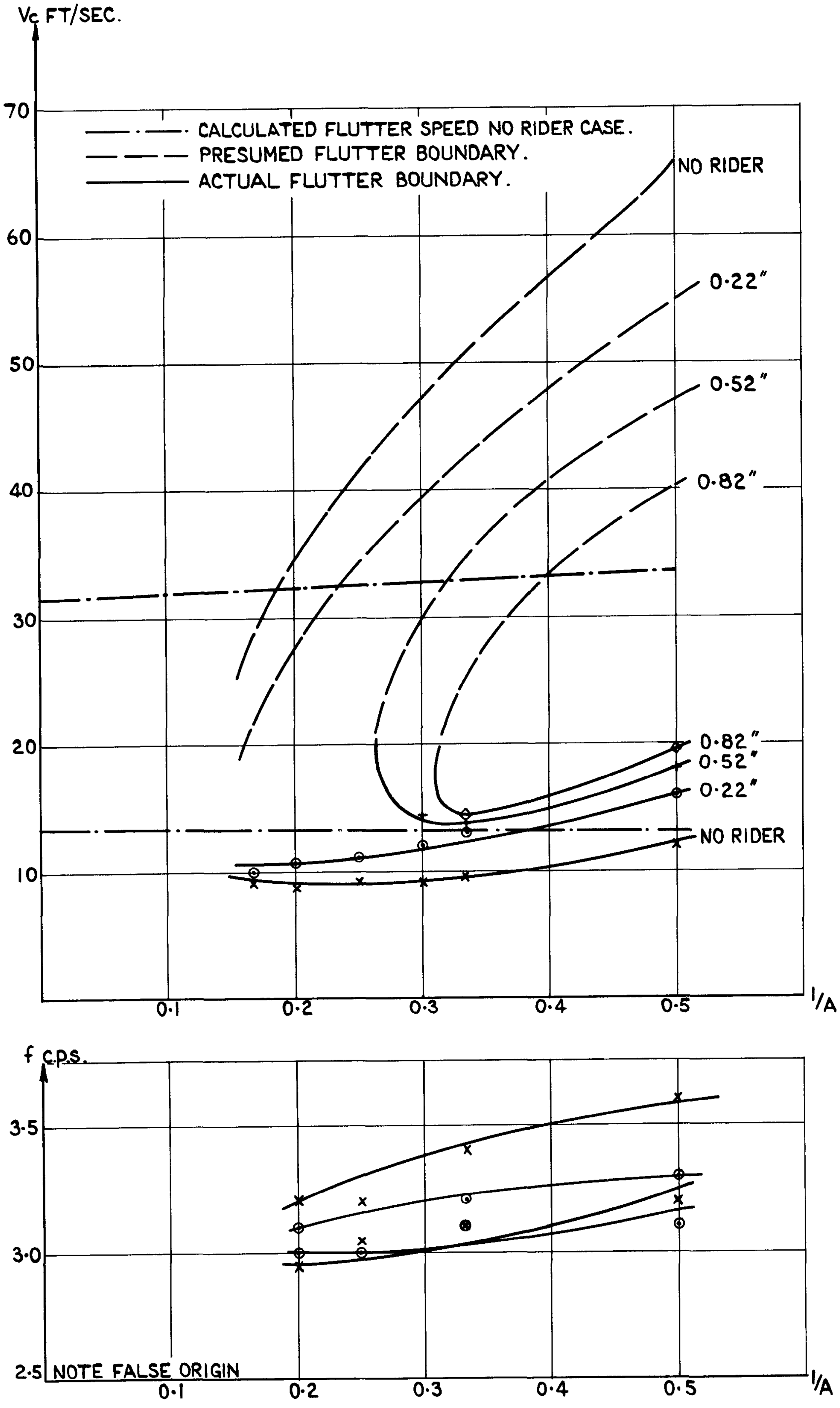
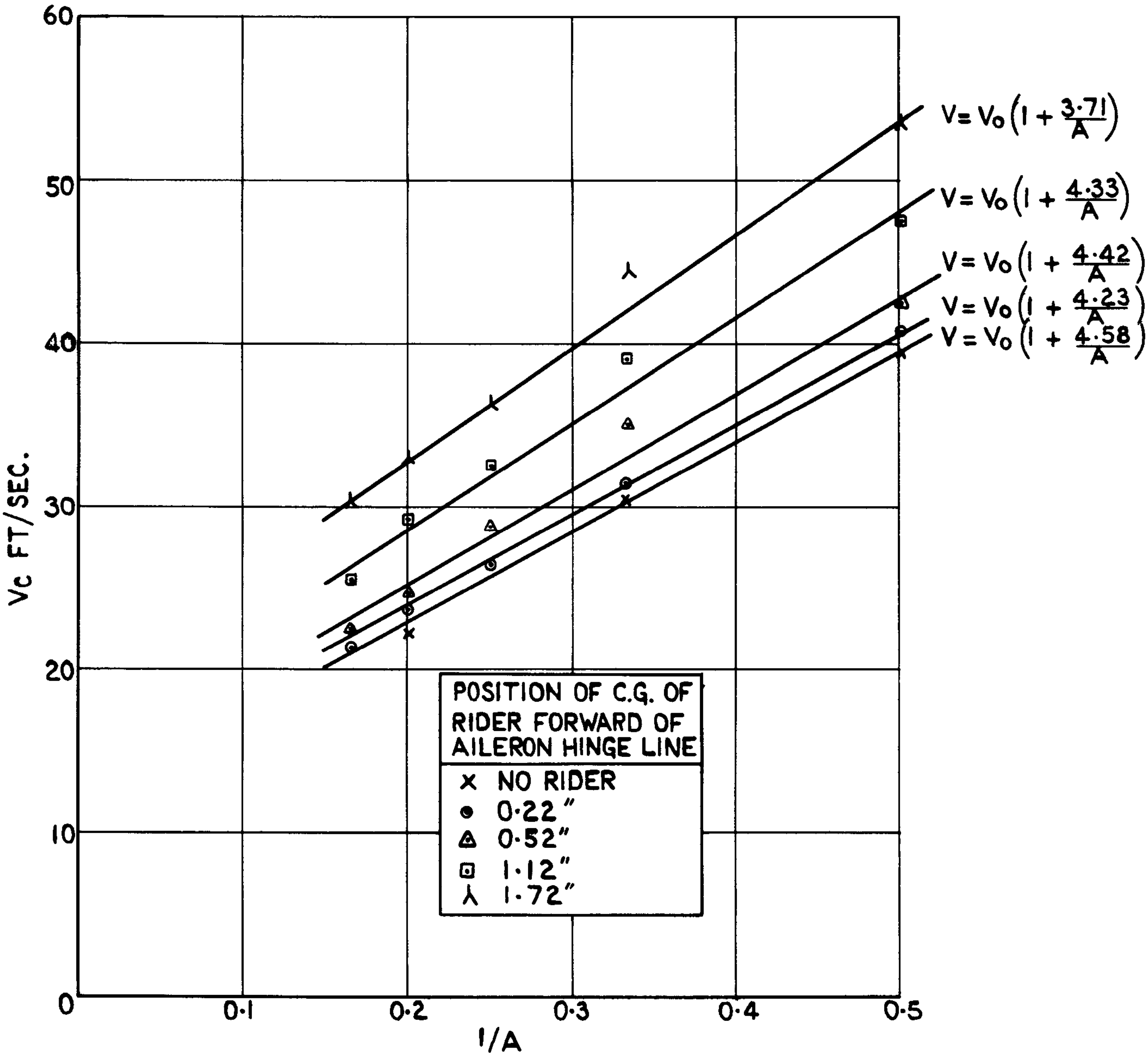


FIG. 3. THE VARIATION OF FLUTTER SPEED AND FREQUENCY WITH ASPECT RATIO FOR THE ROLL-AILERON ROTATION TYPE FLUTTER.



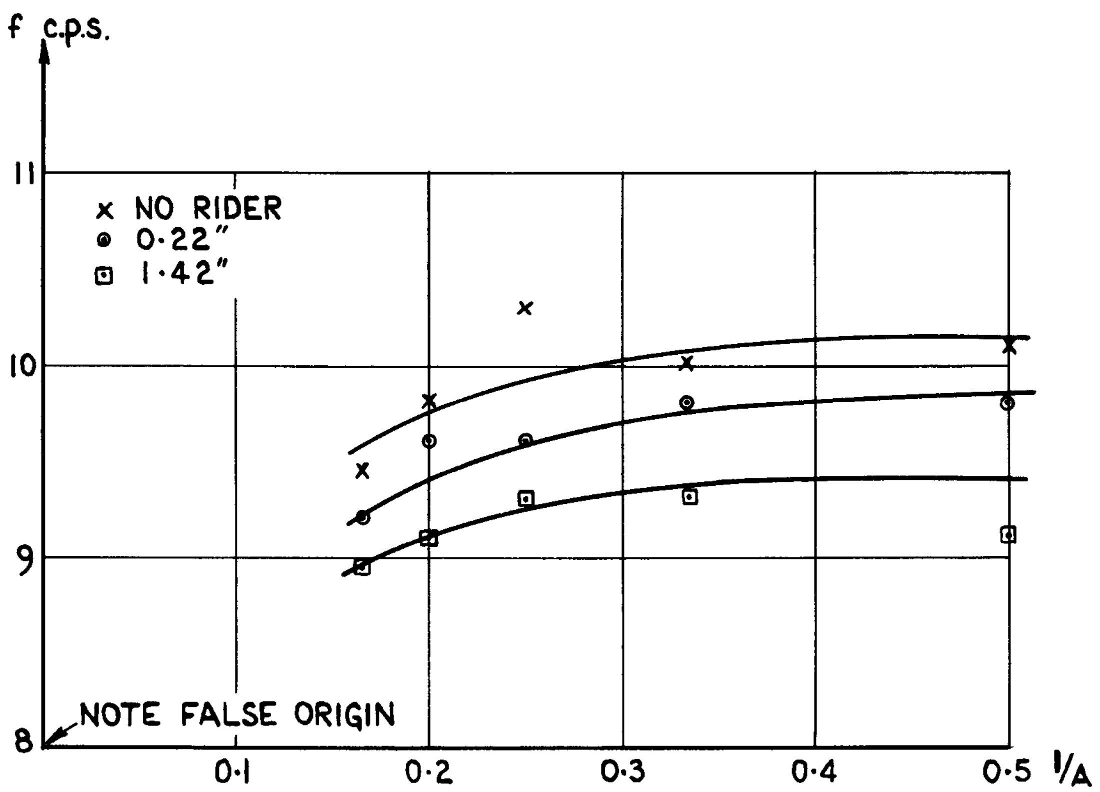


FIG.4. THE VARIATION OF FLUTTER SPEED AND FREQUENCY WITH ASPECT RATIO FOR THE PITCH-AILERON ROTATION TYPE FLUTTER.

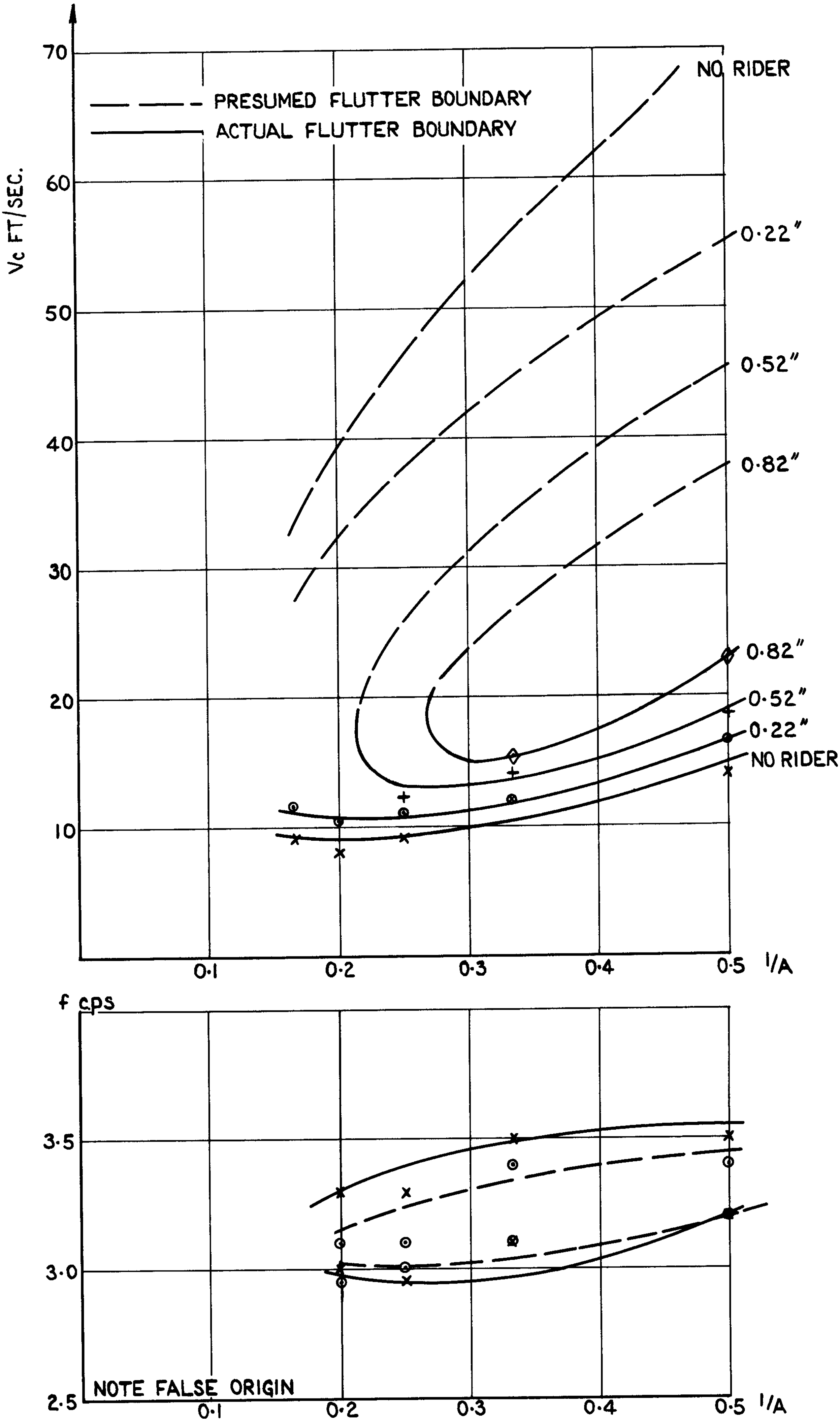


FIG.5. THE VARIATION OF FLUTTER SPEED AND FREQUENCY WITH ASPECT RATIO FOR THE TERNARY TYPE FLUTTER OF ROLL-AILERON ROTATION CHARACTER.

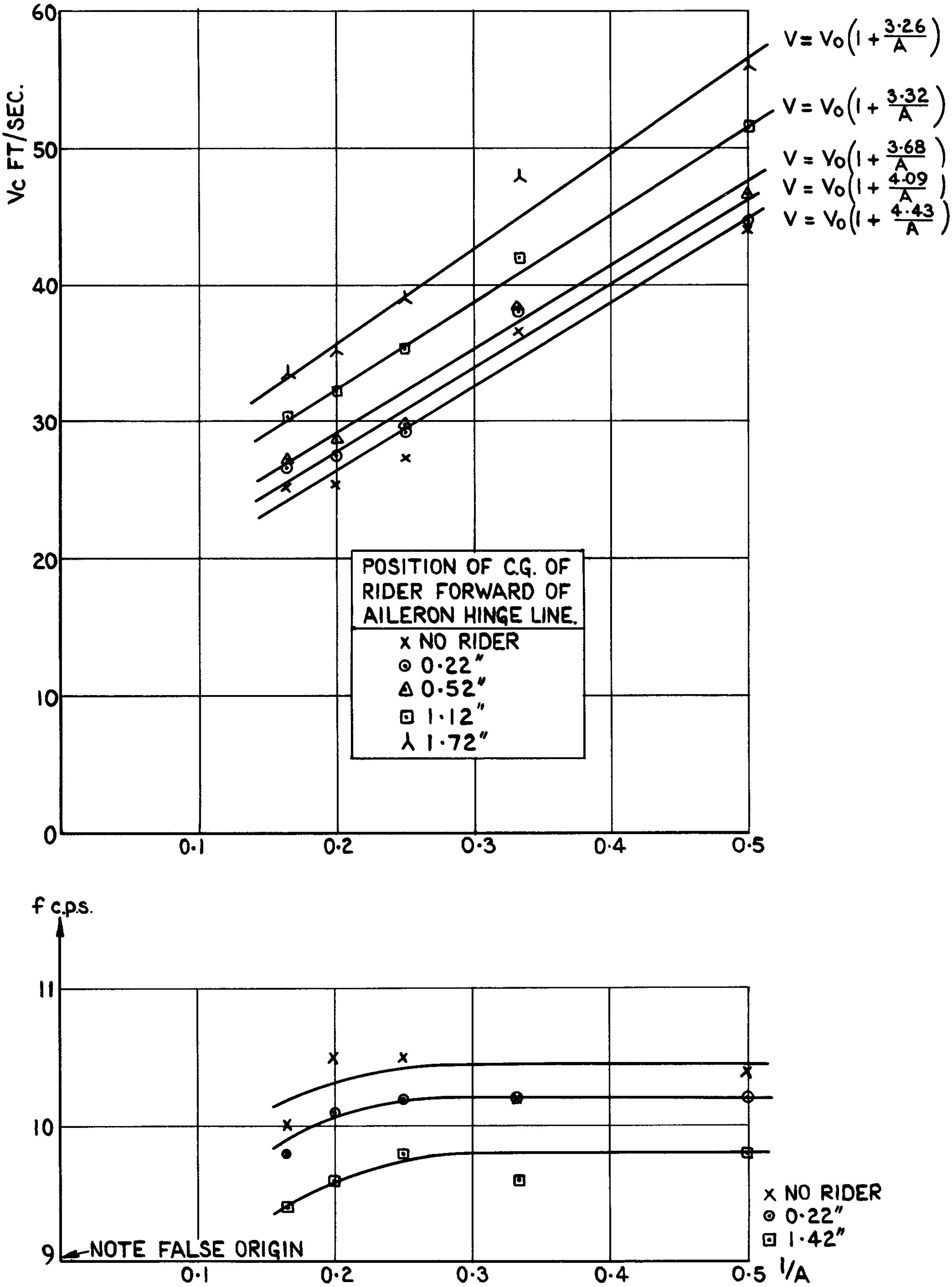


FIG. 6. THE VARIATION OF FLUTTER SPEED AND FREQUENCY WITH ASPECT RATIO FOR THE TERNARY TYPE FLUTTER OF PITCH-AILERON ROTATION CHARACTER.

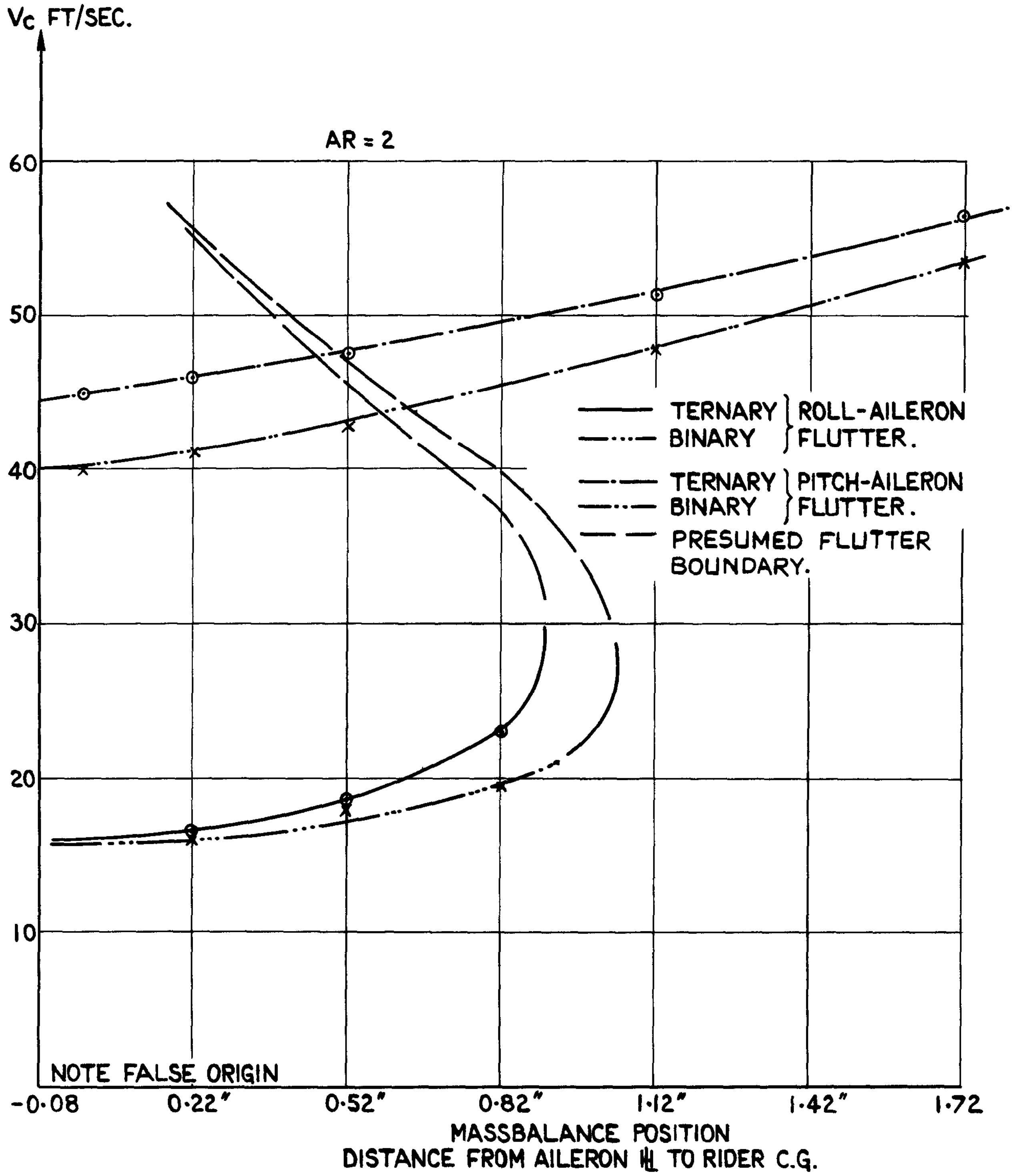
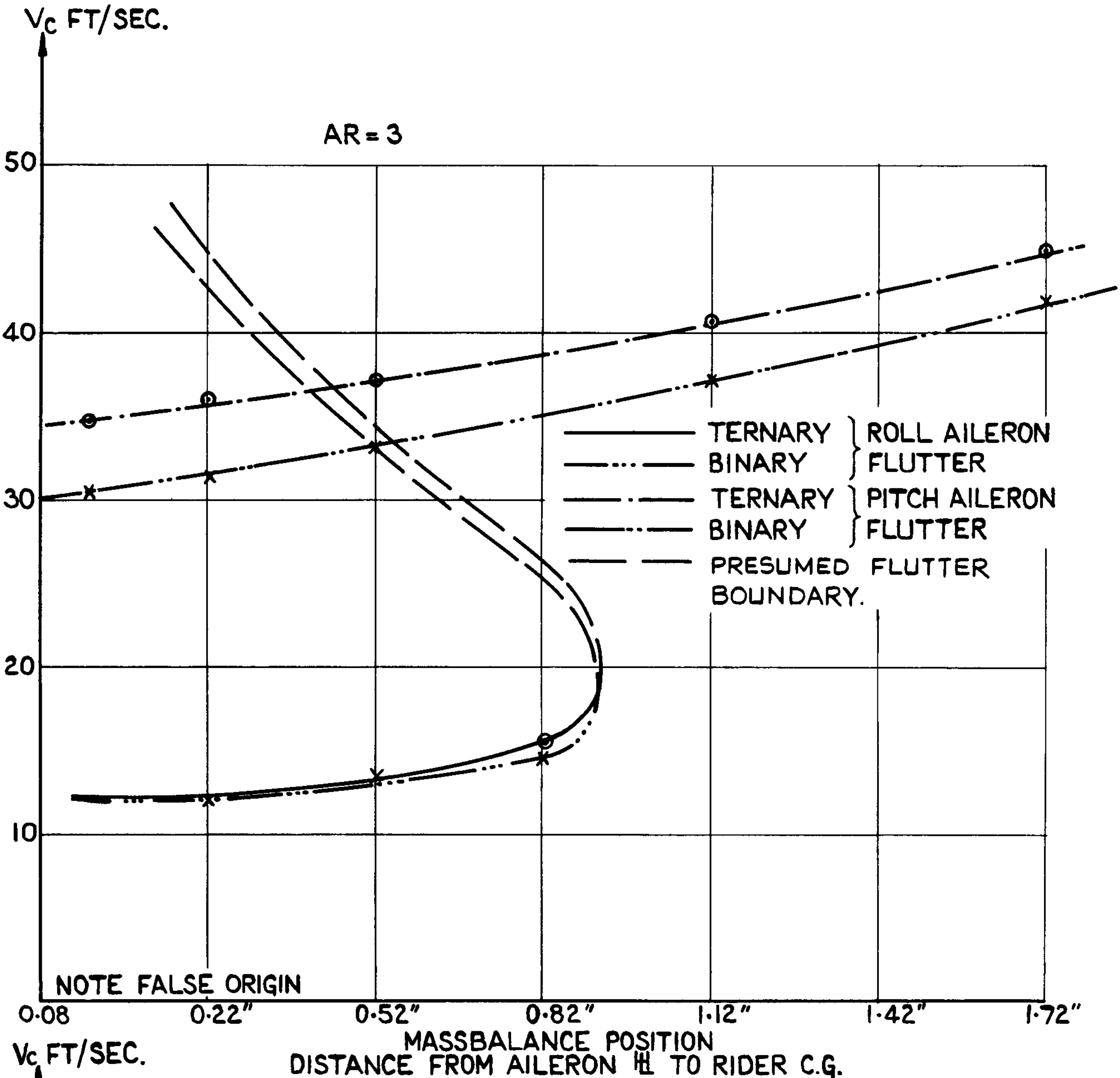


FIG.7. THE VARIATION OF FLUTTER SPEED WITH MASSBALANCE POSITION.



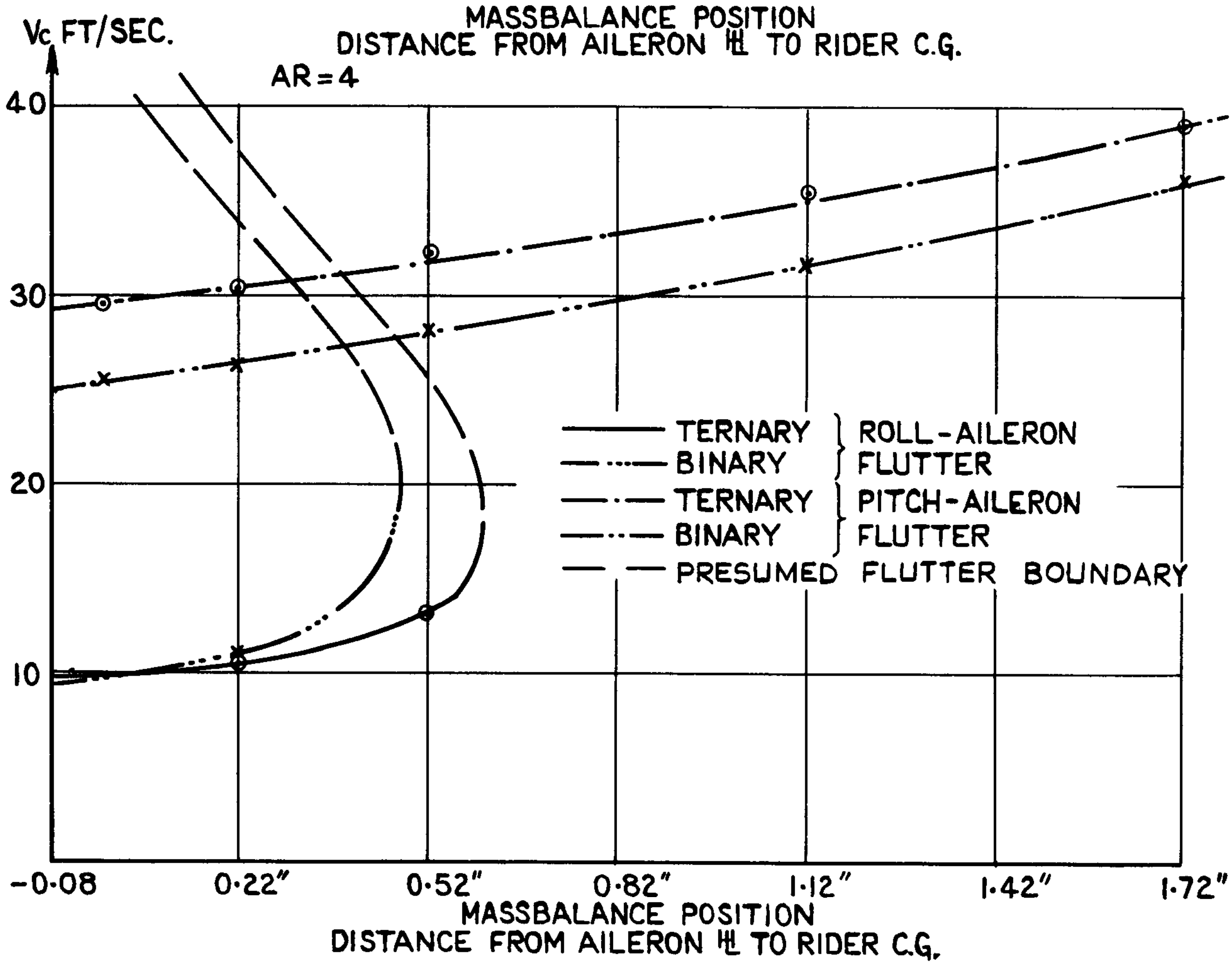


FIG.8. THE VARIATION OF FLUTTER SPEED WITH MASSBALANCE POSITION.

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