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SOME THEORETICAL DEVELOPMENTS IN AIRPLANE STATIC LONGITUDINAL STABILITY AND CONTROL

ARMY AIR FORCES
AIR MATERIEL COMMAND
Wright Field Dayton, Ohio

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ARMY AIR FORCES TECHNICAL REPORT

No. 5167

SOME THEORETICAL DEVELOPMENTS

IN AIRPLANE STATIC LONGITUDINAL STABILITY AND CONTROL

Title

BY

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SOME THEORETICAL DEVELOPMENTS IN AIRPLANE STATIC LONGITUDINAL STABILITY AND CONTROL

SUMMARY

In this report, theoretical methods are developed for predicting some static longitudinal stability and control characteristics of airplanes from their basic geometry. The concepts of the stick-fixed, stick-free, and maneuvering neutral points are established and methods given for their evaluation. Methods for finding these neutral points from flight tests are discussed briefly, and the results of some calculated propeller-off neutral points compared against wind-tunnel tests. Finally, an example is worked out in detail to demonstrate the methods developed herein.

DATE AND PLACE OF INVESTIGATION

The investigation was conducted at Wright Field, Dayton, Ohio, from 1 May 1944 to 1 June 1944.

OBJECT

To present methods for evaluating an airplane's static longitudinal stability characteristics from its basic geometry.

INTRODUCTION

The static longitudinal stability and control characteristics of any airplane are made known to the pilot through the variation of stick position and stick force with airspeed in unaccelerated flight and in the variation of stick position and stick force per unit normal acceleration in accelerated flight.

It is the duty of the airplane designer to insure that his airplane will have stability and control characteristics such that the variations given above will be satisfactory to the pilot and that sufficient control is furnished for achieving the design aerodynamic limits in the most adverse configurations. The designer of military aircraft is guided in this by the basic requirements set forth in Army Air Forces Specification C-1815, "Stability and Control Requirements for Airplanes," which was written after considerable study correlating pilots' opinions of the airplane's "feel" and control needs with the quanitative requirements as laid down.

The design of the airplane for static longitudinal stability and control is related very closely to its anticipated center of gravity travel and its position relative to the wing mean aerodynamic chord. In order to position the airplane's c.g. range properly, it is imperative that the airplane designer have some knowledge of the center of gravity locations for which the airplane will be satisfactory to the pilot. The most aft c.g. allowable is the one at which the airplane's stability vanishes, and it is with this limit that this paper will deal.

The center of gravity positions at which the stability vanishes is termed the "neutral point," and for every airplane there are four major neutral points. As it is of considerable interest to the designer to be able to predict these for a new design, methods are developed herein for their evaluation from the basic airplane geometry.

An analytical treatment of the effects of the propeller and power are not attempted as they are beyond the scope of this paper. A statistically developed correction factor ΔP is given for the effects of propeller and power for first estimate purposes only. These factors are given with some reservations, as considerable error can be involved in their use; but if they are considered qualitatively they can be very useful.

RESULTS AND DISCUSSION

1. A study of static longitudinal stability and control is a study of the moments about the airplane's lateral or Y axis and their variation with change in airplane lift coefficient. The forces and moments in the airplane's plane of symmetry are depicted in sketch 1.

Taking moments about the center of gravity, the following equation is obtained after neglecting the moment about the horizontal tail's aerodynamic center and the chordwise force component of the horizontal tail.

$$M_{c.g.} = N \cdot x_a + C \cdot z_a + M_0 + M_{fus} + M_{NAC} - N_t \cdot \ell_t \qquad (1)$$

Dividing by qsc, we transfer the above equation to coefficient form

$$C_{m_{c.g.}} = C_N \frac{x_a}{c} + C_X \frac{z_a}{c} + C_{m_0} + C_{m_{fus}} + C_{m_{NAC}} - C_{N_t} \frac{S_t}{S_w} \frac{\ell_t}{c} \eta_t \qquad (2)$$

Differentiating this equation with respect to C_L yields the stability equation (3)

$$\frac{dC_m}{dC_L}_{c.g.} = \frac{dC_N}{dC_L} \frac{x_a}{c} + \frac{dC_X}{dC_L} \frac{z_a}{c} + \frac{dC_m}{dC_L}_{fus} + \frac{dC_m}{dC_L}_{nac} - \frac{dC_{N_t}}{dC_L} \frac{S_t}{S_w} \frac{\ell_t}{c} \eta_t$$

which on simplification and making use of the assumption that $C_N = C_L$ and $C_{N_t} = C_{L_t}$

$$\frac{dC_m}{dC_L}_{c.g.} = \frac{x_a}{c} + C_L \left(\frac{2}{\pi e A} - \frac{1}{a_w}\right) \frac{z_a}{c} + \frac{dC_m}{dC_L}_{fus} + \frac{dC_m}{dC_L}_{nac} - \frac{a_t}{a_w} \frac{S_t}{S_w} \frac{\ell_t}{c} \eta_t \left(1 - \frac{d\epsilon}{d\alpha}\right) \qquad (4)$$

The first term of equation (4) is the major wing contribution to the airplane's static longitudinal stability. It is the only term that is affected by a longitudinal shift of the center of gravity and is consequently of extreme importance. It is equal to the difference in percent m.a.c. between the wing aerodynamic center and the airplane's center of gravity.

$$\text{i.e.}\quad \frac{x_a}{c} = c.g. - a.c.$$

The theoretical calculation of this contribution is as accurate as is the theory on prediction of the aerodynamic center location on any given wing. Unfortunately present knowledge on this point is not very accurate, and the error involved in picking an aerodynamic center location is usually found to be the largest error in these calculations.

The theoretical location of the aerodynamic center from wing theory is at the quarter chord position. However, experience has shown that the aerodynamic center may be actually as far forward as 20 percent and as far aft as 28 percent. There have been several methods developed for prediction of the aerodynamic center location, but none have been found any more accurate than the simple formula given below, which is used in this report.

$$a.c._{WING} = \frac{(a.c.)_{o_{ROOT}} \times \text{Taper Ratio} + (a.c.)_{o_{TIP}}}{\text{Taper Ratio} + 1}$$

The second term of equation (4) is the contribution of the wing chordwise force component to the airplane's static longitudinal stability and is usually small compared to the first term. It is a function of C_L and at the higher lift coefficients tends to stabilize a high-wing airplane and to destabilize a low-wing airplane. Its effect on present-day airplanes is usually very small and is neglected in this analysis.

The third and fourth terms of equation (4) are the stability contributions of the fuselage and nacelles. These contributions are almost invariably unstable and may be computed from the formulae given below.

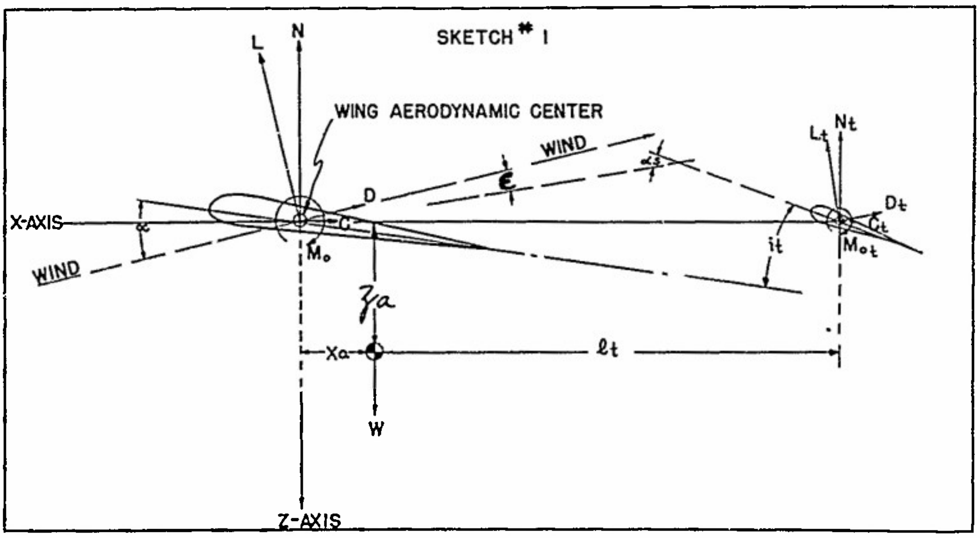
$$\left(\frac{dC_m}{dC_L}\right)_{fus} = \frac{K_f w_f^2 L_f}{a_w S_w c}$$

$$\left(\frac{dC_m}{dC_L}\right)_{NAC} = \frac{K_n w_n^2 L_n N}{a_w S_w c}$$

The coefficients K_f and K_n are functions of the airplane's c.g. position with respect to the fuselage or nacelles and are given in figure 7 as functions of the position of the wing root ¼-chord in percent body length. These constants were determined in systematic tests run by the NACA and compiled in this form in reference 1.

The slope of the lift curve of the wing (a_w) is obtained by reference to figure 1. In this figure the lift curve slope is plotted versus aspect ratio for various values of the section slope of the lift curve.

The fifth term of equation (4) is the tail's contribution to the stability. To evaluate this term, it is necessary



to know the slope of lift curve of the horizontal tail. This slope may be obtained by reference to figure 2. It is also necessary to determine the rate of change of downwash with wing angle of attack $\left(\frac{d\epsilon}{d\alpha}\right)$. This characteristic may be evaluated from figures 3, 4, 5, and 6, obtained directly from reference 1. The tail efficiency, η_t, varies somewhat, but on the average may be taken as .9. The geometric term $\left(\frac{S_t l_t}{S_w c}\right)$ is called the tail volume coefficient and is given the symbol $\bar{V}$. The tail contribution to stability is then a function of this coefficient, the tail efficiency, the rate of change of downwash and the ratio of the lift curve slope of the horizontal tail to that of the wing, which in itself is a function of the surface aspect ratios.

Equation (4) rearranged in the light of the above discussion becomes

$$\frac{dC_m}{dC_L}\text{c.g.} = \text{c.g.} - \text{a.c.} + \frac{K_f w_f^2 L_f}{a_w S_w c} + \frac{K_n w_n^2 L_n N}{a_w S_w c} - \frac{a_t}{a_w}\bar{V}\eta_t\left(1 - \frac{d\epsilon}{d\alpha}\right) \quad (5)$$

With equation (5), the immediate question arises as to what is the center of gravity position that corresponds to neutral stability or the center of gravity position that makes $\left(\frac{dC_m}{dC_L}\right)_{c.g.}$ equal to zero.

This c.g. position is readily obtainable from equation (5) by equating $\left(\frac{dC_m}{dC_L}\right)_{c.g.}$ to zero and solving for the c.g. position. (6)

$$N_o = (\text{c.g.})_{\frac{dC_m}{dC_L}=0} = \text{a.c.} - \frac{K_f w_f^2 L_f}{a_w S_w c} - \frac{K_n w_n^2 L_n N}{a_w S_w c} + \frac{a_t}{a_w}\bar{V}\eta_t\left(1 - \frac{d\epsilon}{d\alpha}\right)$$

This center of gravity position in percent m.a.c., giving $\left(\frac{dC_m}{dC_L}\right)_{c.g.} = 0$, is called the stick fixed neutral point and is referred to as N_o in the rest of this paper.

Once the neutral point of any airplane is obtained, the slope of the pitching moment curve may be determined at once from the very simple relationship

$$\frac{dC_m}{dC_L} = \text{c.g.} - N_o \quad (7)$$

The slope of the pitching moment curve versus lift coefficient is numerically equal to the difference between the c.g. in question and the neutral point, where both are expressed in percent m.a.c.

The effect of the propellers on the static longitudinal stability can be approximated roughly from the following table which was developed statistically from flight tests of numerous types of airplanes:

TABLE I

Tractor Airplanes Airplane Type	Δ P Forward Shift of Neutral Point (% m.a.c.)
Single-engine fighters and attack airplanes	6%
Twin-engine fighters and attack airplanes	10%
Multi-engined bombers	15%
Pusher type airplanes	No shift

2. The determination of the stick-fixed neutral point, N_o, is of extreme importance in the analysis of any airplane design, as many other factors depend on this one point. The movement of the stick to produce changes in speed is definitely a function of the distance of c.g. from the neutral point. The latter is usually called the static margin stick-fixed and will be referred to throughout this paper as H_o. The static margin then is also equal numerically to the slope of the pitching moment curve $\left(\frac{dC_m}{dC_L}\right)_{c.g.}$ through equation (7).

The variation of elevator angle required to change the trim C_L can be developed quite rapidly from the slope of the pitching moment curve. The change in trim is accomplished by an adjustment of equation (2) through a variation in the tail term.

$$(C_{m_{c.g.}})_t = -C_{L_t}\bar{V}\eta_t \quad (8)$$

The variation of this term comes as the result of the ability of the elevator to rotate the tail zero lift line, thereby causing changes in the tail lift coefficient.

The variation of tail angle of zero lift with elevator deflection is a function of the area ratio between the elevator and horizontal tail. The ability of an elevator to shift the zero lift line is called the "elevator effectiveness" and is referred to by the Greek letter τ. The variation of τ with the area ratio $\left(\frac{S_e}{S_t}\right)$ is given in figure 8.

The change in pitching moment per degree change in elevator angle can be written as follows:

$$\Delta C_{m_{c.g.tail}} = -\Delta C_{L_{t_{\delta_e}}}\bar{V}\eta_t \quad (9)$$

$$\text{or } \Delta C_{m_{c.g.tail}} = -a_t\tau\bar{V}\eta_t\Delta\delta_e \quad (10)$$

$$\text{or } \frac{dC_m}{d\delta_e} = -a_t\tau\bar{V}\eta_t \quad (11)$$

It can be seen from equation (10) that the change in pitching moment with change in elevator angle is independent of lift coefficient. Therefore, the slope of C_m vs. C_L will not be changed by elevator deflection.

The variation of elevator angle required with change in airplane lift coefficient can be obtained from the combination of the two derivatives $\left(\frac{dC_m}{dC_L}\right)_{c.g.}$ and $\left(\frac{dC_m}{d\delta_e}\right)$ as follows:

$$\frac{d\delta_e}{dC_L} = -\frac{dC_m/dC_L}{dC_m/d\delta_e} \tag{12}$$

$$\text{or } \frac{d\delta_e}{dC_L} = -\frac{H_o}{a_t V \eta_t} \tag{13}$$

This derivative, being a direct function of the static margin H_o, vanishes when $H_o = 0$, or in other words, when the c.g. is located on the airplane's stick-fixed neutral point. This relationship is used to determine the stick-fixed neutral point by flight test. Measurements of elevator angle versus C_L are obtained for three different c.g. positions as widely separated as possible and the slopes obtained plotted versus c.g. position and extrapolated to zero. The c.g. thus obtained is the stick-fixed neutral point, N_o.

3. The second way that the airplane's stability is felt by the pilot is through the variation of stick force with change in C_L from a trimmed speed. In the case of stick-fixed stability, the pilot feels the airplane to be neutrally stable when $d\delta_e/dC_L = 0$, and this was shown to occur when (dC_m/dC_L) stick-fixed $= 0$. This second stability, felt by the pilot is dF_s/dC_L, and it is also considered to be neutral when this derivative is zero. This condition does not depend on the static margin, H_o, as $d\delta_e/dC_L$ does, but is closely associated with the variation of dC_m/dC_L with elevators free to float with the wind. This derivative is termed the static longitudinal stability, stick-free, and must be analyzed separately.

If left free, the elevator will float with or against the wind, depending on its aerodynamic balance, and thereby cause changes in the rate of the lift contribution of the tail, changing its stability contribution.

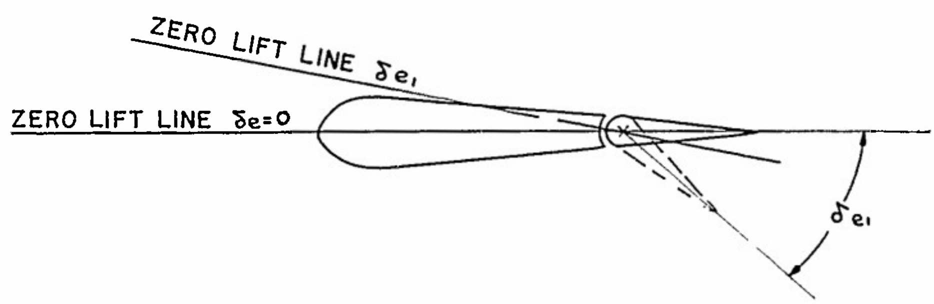
The floating characteristics of the elevator can be determined from the elevator hinge moment coefficient, which is usually expressed as

$$C_{h_e} = C_{h_o} + C_{h_\alpha} \cdot \alpha_s + C_{h_\delta} \cdot \delta_e \tag{14}$$

The floating angle for the case where $C_{h_e} = 0$ can be obtained by equating equation (14) to zero and solving for the elevator angle

$$\delta_e = -\frac{C_{h_\alpha}}{C_{h_\delta}} \alpha_s \tag{15}$$

SKETCH # 2



No attempt is made in this paper to present variations of the hinge moment parameters with various types and amounts of aerodynamic balance. A typical variation of the parameters C_{h_α} and C_{h_δ} with percent overhang balance is given in figure 9.

To evaluate the slope of the pitching moment curve with elevator free, power off, it is necessary to determine the influence on stability of the free elevators.

$$\left(\frac{dC_m}{dC_L}\right)_{\text{STICK FREE}} = \left(\frac{dC_m}{dC_L}\right)_{\text{STICK FIXED}} + \Delta\left(\frac{dC_m}{dC_L}\right)_{\substack{\text{DUE TO FREEING}\\ \text{ELEVATOR}}} \tag{16}$$

The stability contribution of the free elevator can be developed as follows:

$$\Delta\left(\frac{dC_m}{dC_L}\right)_{\substack{\text{DUE TO FREEING}\\ \text{ELEVATOR}}} = \left(\frac{d\delta_e}{dC_L}\right)_{C_{h_e}=0} \times \frac{dC_m}{d\delta_e} \tag{17}$$

$$\left(\frac{d\delta_e}{dC_L}\right)_{C_{h_e}=0} = -\frac{C_{h_\alpha}}{C_{h_\delta}}\left(1-\frac{d\epsilon}{d\alpha}\right)\frac{1}{a_w} \tag{18}$$

$$\text{so } \Delta\left(\frac{dC_m}{dC_L}\right)_{\substack{\text{DUE TO FREEING}\\ \text{ELEVATOR}}} = \frac{C_{h_\alpha}}{C_{h_\delta}}\left(1-\frac{d\epsilon}{d\alpha}\right)\frac{a_t}{a_w}\,\tau\,\bar{V}\eta_t \tag{19}$$

$$\textit{finally} \left(\frac{dC_m}{dC_L}\right)_{\text{STICK FREE}} = \left(\frac{dC_m}{dC_L}\right)_{\text{STICK FIXED}} + \frac{C_{h_\alpha}}{C_{h_\delta}}\left(1-\frac{d\epsilon}{d\alpha}\right)\frac{a_t}{a_w}\,\tau\,\bar{V}\eta_t \tag{20}$$

Substitution of equation (5) for $\left(\frac{dC_m}{dC_L}\right)$ stick fixed and rearranging gives

$$\left(\frac{dC_m}{dC_L}\right)_{\text{STICK FREE}} = c.g. - a.c. + \frac{K_f w_f^2 L_f}{a_w S_w c} + \frac{K_n w_n^2 L_n N}{a_w S_w c} - \frac{a_t}{a_w}\bar{V}\eta_t\left(1-\frac{d\epsilon}{d\alpha}\right)\left(1-\frac{C_{h_\alpha}}{C_{h_\delta}}\tau\right) \tag{21}$$

This is the equation for static longitudinal stability stick-free, power off. The effect of freeing the controls enters the tail stability term as the multiplying factor $\left(1-\frac{C_{h_\alpha}}{C_{h_\delta}}\tau\right)$. For an airplane whose horizontal tail has no change in hinge moment with angle of attack ($C_{h_\alpha}=0$), this term is zero, and there is no effect of freeing the elevator on the static longitudinal stability. As the value of τ is usually near .5, a ratio of $\frac{C_{h_\alpha}}{C_{h_\delta}}=2$ can obviate the whole tail contribution to stability on freeing the elevators. For this reason, the importance of careful design of the elevator aerodynamic balance to insure adequate stick-free stability can be readily appreciated.

The determination of the stick-free neutral point, referred to throughout this report as N_o' can be obtained from equation (21) by equating it to zero and solving for the c.g.

$$N_o' = c.g._{\left(\frac{dC_m}{dC_L}\right)_{\substack{\text{STICK}\\ \text{FREE}}}=0} = a.c. - \frac{C_{m_0}}{C_L} - \frac{K_f w_f^2 L_f}{a_w S_w c_w} - \frac{K_n w_n^2 L_n N}{a_w S_w c_w} + \frac{a_t}{a_w}\bar{V}\eta_t\left(1-\frac{d\epsilon}{d\alpha}\right)\left(1-\frac{C_{h_\alpha}}{C_{h_\delta}}\tau\right) \tag{22}$$

The difference between the stick-fixed and stick-free neutral points may be obtained by subtracting equation (5) from equation (22)

$$N_o - N_o' = \frac{a_t\bar{V}\tau\eta_t\left(1-\frac{d\epsilon}{d\alpha}\right)C_{h_\alpha}}{a_w C_{h_\delta}} \tag{23}$$

In the discussion of stick-fixed stability, it was shown that a very close correlation exists between the slope of the pitching moment curve $\left(\frac{dC_m}{dC_L}\right)$ stick fixed and the variation of elevator angle with lift coefficient (equation 13). The second stability characteristic felt by the pilot is the variation of stick force as the speed is changed away from some trim point. Unfortunately the correlation between $\left(\frac{dC_m}{dC_L}\right)_{\text{stick free}}$ and the variation $\left(\frac{dF_s}{dC_L}\right)$ is not as simple as it was in the stick-fixed case, since $\left(\frac{dF_s}{dC_L}\right)$ depends very closely on the trim tab setting and the trim speed.

The pilot's force at the top of the stick can be given as the following function of the elevator hinge moment

$$F_s = G \cdot HM_e \tag{24}$$

$$\text{or } F_s = G b_e c_e^2 q \eta_t C_{h_e} \tag{25}$$

The hinge moment coefficient may be expressed as

$$C_{h_e} = C_{h_0} + C_{h_\alpha}\cdot\alpha_s + C_{h_{\delta_e}}\cdot\delta_e + C_{h_{\delta_t}}\cdot\delta_t \tag{26}$$

$$\alpha_s = \alpha_w - \epsilon + i_t \tag{27}$$

$$\textit{and } \delta_e = \delta_{e_0} - \left(\frac{dC_m}{dC_L}\right)_{\text{STICK FIXED}}\cdot\frac{C_L}{C_{m_\delta}} \tag{28}$$

$$\textit{where } C_{m_\delta} = dC_m/d\delta_e$$

Substitution of (26), (27), and (28) into (25) gives

$$F_s = G\eta_t q b_e c_e^2 \left\{ C_{h_0} + C_{h_\alpha}\left[\alpha_0 + \frac{d\alpha}{dC_L} C_L \left(1 - \frac{d\epsilon}{d\alpha}\right) + i_t\right] + C_{h_\delta}\left[\delta_{e_0} - \left(\frac{dC_m}{dC_L}\right)_{\text{STICK FIXED}} \frac{C_L}{C_{m_\delta}}\right] + C_{h_{\delta t}} \delta_t \right\} \quad (29)$$

Substituting $q = \frac{1}{C_L}\frac{W}{S_w}$ and differentiating with respect to C_L yields the desired slope

$$\frac{dF_s}{dC_L} = -\frac{G\eta_t \frac{W}{S_w} b_e c_e^2}{C_L^2}\left(C_{h_0} + C_{h_\alpha}\cdot\alpha_0 + C_{h_\alpha}\cdot i_t + C_{h_{\delta_e}}\cdot\delta_{e_0} + C_{h_{\delta_t}}\cdot\delta_t\right) \quad (30)$$

This brings out the fact that the rate of change of stick force with lift coefficient is independent of c.g. position, or the amount of static longitudinal stability when the trim tab is held fixed. It also shows that the slope of $\left(\frac{dF_s}{dC_L}\right)$ is a direct function of the trim tab setting. A nose-up tab will give a more stable slope of stick-force versus C_L than will a nose-down tab.

From the above it is obvious that the variation of stick force versus lift coefficient with fixed trim is independent of the stick free stability. In order to show the correlation between the stick-free pitching moment versus lift coefficient curve and the stick-force versus lift coefficient curve, it is necessary to do one of three things.

The first is to find the variation of $\left(\frac{F_s}{q}\right)$ with C_L. This can be obtained from equation (29) by dividing out q and differentiating with respect to C_L, giving

$$\frac{dF_s/q}{dC_L} = -\frac{G\eta_t b_e c_e^2 C_{h_\delta}}{C_{m_\delta}}\left(\frac{dC_m}{dC_L}\right)_{\text{STICK FREE}} \quad (31)$$

The variation of $\left(\frac{F_s}{q}\right)$ with C_L is a direct function of the stick-free stability and is independent of trim tab setting.

The second method is to specify trim $F_s = 0$ at a particular $C_L = C_{L_{trim}}$. If $C_{L_{trim}}$ is substituted in equation (29) and $(C_{h_{\delta t}}\cdot\delta_t)$ is solved for $F_s = 0$ at $C_{L_{trim}}$ and also substituted in equation (29), the equation for stick force becomes

$$F_s = -\frac{G\eta_t \frac{W}{S_w} b_e c_e^2 C_{h_\delta}}{C_{m_\delta}}\left(1 - \frac{C_{L_{TRIM}}}{C_L}\right)\left(\frac{dC_m}{dC_L}\right)_{\text{STICK FREE}} \quad (32)$$

which upon differentiation with respect to C_L gives

$$\frac{dF_s}{dC_L} = -\frac{G\eta_t \frac{W}{S_w} b_e c_e^2 C_{h_\delta} C_{L_{TRIM}}}{C_{m_\delta} C_L^2}\left[\frac{dC_m}{dC_L}\right]_{\text{STICK FREE}} \quad (33)$$

This indicates that if the airplane is trimmed by means of the trim tab to a given trim C_L, then the variation of $\left(\frac{dF_s}{dC_L}\right)$ at any particular C_L is a direct function of the stick-free stability.

The third method is to investigate the tab angle required to vary the trim C_L at $F_s = O$. Solving equation (29) for this condition and differentiating

$$\frac{d\delta_t}{dC_L} = \frac{C_{h_\delta}}{C_{m_\delta} C_{h_{\delta_t}}}\left[\frac{dC_m}{dC_L}\right]_{\text{STICK FREE}} \quad (34)$$

The variation of tab angle with trim lift coefficient then is a direct function of the stick-free longitudinal stability, and therefore vanishes at the stick-free neutral point.

From the development just given it can be seen that there is a very close relationship between the slope $\left(\frac{dC_m}{dC_L}\right)_{\text{stick free}}$ and the variation of stick force with speed away from the trim speed. The relationship is not as clean cut as it is between $\left(\frac{dC_m}{dC_L}\right)_{\text{stick fixed}}$ and the variation of elevator angle with speed, but it is none the less important, and the stick-free neutral point must be determined or evaluated in any preliminary design.

4. The third way in which longitudinal stability of the airplane is felt by the pilot is in the stick forces required to produce normal acceleration on the airplane in maneuvering flight. Certain airplanes are required to be highly maneuverable with their primary function requiring nearly continuous accelerated flight. It is obvious, therefore, that for these airplanes the maneuvering characteristics may be all important and must be carefully considered in the design.

The acceleration of the airplane, due to the unbalanced lift forces perpendicular to the flight path, manifests itself as a curvature of this flight path and a rotation of the airplane about its Y axis. This angular velocity about the Y axis produces damping moments due to the horizontal tail and fuselage tending to stop the rotation. These damping moments must be overcome by the application of more up elevator and generally more stick force than required to change the trim lift coefficient in unaccelerated flight. This phenomenon gives rise to a sort of pseudo-stability or apparent stability, for when the airplane is balanced at either the stick-fixed or stick-free neutral points, it will be necessary to deflect the elevator and increase the stick force respectively in order to increase the lift coefficient in accelerated flight.

As both the elevator angle and stick force variations in accelerated flight are directly affected by the airplane's stability and therefore the c.g. position, it becomes obvious that there is some c.g. behind the stick-fixed and stick-free neutral points respectively for which the elevator deflections and stick forces in accelerated flight vanish. These are called the maneuvering neutral points and are termed N_m and N_m' respectively. The following development will indicate the calculations required to determine these neutral points.

The angular velocity about the airplane's Y axis (Q) is a function of the load factor, the airplane's speed, and the type of accelerated flight. If the airplane is pulled up in a vertical plane, the angular velocity about the Y axis can be expressed as follows:

$$Q_{PULL\text{-}UP} = \frac{g}{V}(n-1) \tag{35}$$

In turning flight, the angular velocity of the airplane becomes

$$Q_{TURN} = \frac{g}{V}\left(n - \frac{1}{n}\right) \tag{36}$$

The expression for change in elevator angle to change the trim from one C_L to a higher C_L at constant speed and therefore at a particular normal acceleration is given by the following expression

$$\Delta\delta_e = -Q\frac{dC_m}{dQ}\frac{d\delta_e}{dC_m} - \left(\frac{dC_m}{dC_L}\right)_{STICK\ FIXED}\frac{d\delta_e}{dC_m}\Delta C_L \tag{37}$$

which upon substitution of the pull-up value for Q and letting $\mu = \frac{m}{\rho S_w l_t}$ becomes:

$$\left(\frac{\Delta\delta_e}{n-1}\right)_{PULL\text{-}UPS} = \frac{C_L}{2\mu}\left[\frac{a_t\bar{V} - 2\mu\left(\frac{dC_m}{dC_L}\right)_{STICK\ FIXED}}{C_{m\delta}}\right] \tag{38}$$

or when the turning Q is substituted

$$\left(\frac{\Delta\delta_e}{n-1}\right)_{TURNS} = \frac{C_L}{2\mu}\left[\frac{a_t\bar{V}\left(\frac{n+1}{n}\right) - 2\mu\left(\frac{dC_m}{dC_L}\right)_{STICK\ FIXED}}{C_{m\delta}}\right] \tag{39}$$

It can be seen from these equations that the elevator angle per "g" variation is a direct function of the speed. The faster the speed of the airplane, the lower the gradient of elevator angle required per "g."

The stick-fixed maneuvering neutral point, N_m, can be obtained by equating equations (38) and (39) to zero and substituting equation (7):

$$N_m = N_0 + \frac{a_t\bar{V}}{2\mu}\quad PULL\text{-}UPS \tag{40}$$

$$N_m = N_0 + \frac{a_t\bar{V}}{2\mu}\left(1 + \frac{1}{n}\right)\quad TURNS \tag{41}$$

The expressions for stick force required to produce a one "g" change in normal acceleration is developed as follows:

$$F_s = \Delta C_{h_e} S_e c_e \frac{g}{l_t} G + KG(n-1) \tag{42}$$

where K equals the elevator weight moment and $\Delta C_{h_e} = C_{h_\alpha}\cdot\Delta\alpha_s + C_{h_\delta}\cdot\Delta\delta_e$

The change of angles of attack at the stabilizer can be written as follows:

$$\Delta\alpha_s = \frac{\Delta C_L}{a_w}\left(1 - \frac{d\epsilon}{d\alpha}\right) + \frac{Ql_t}{V} \tag{43}$$

Also from (38)

$$\Delta\delta_e = (n-1)\frac{C_L}{2\mu}\left[\frac{a_t\bar{V} - 2\mu\left(dC_m/dC_L\right)_{STICK\ FIXED}}{C_{m\delta}}\right] \tag{44}$$

Substitution into (42) and rearranging gives:

$$\left(\frac{F_s}{n-1}\right)_{PULL\text{-}UPS} = \frac{WGS_e c_e C_{h\delta} c}{a_t l_t S_t T}\left[\frac{dC_m}{dC_L}\right]_{STICK\ FREE} + 57.3\,GS_e c_e \frac{\rho}{2} g l_t\left[C_{h_\alpha} - \frac{C_{h\delta}}{T}\right] + KG \tag{45}$$

In turns a similar development gives

$$\left(\frac{F_s}{n-1}\right)_{TURNS} = \frac{WGS_e c_e C_{h\delta} c}{a_t l_t S_t T}\left[\frac{dC_m}{dC_L}\right]_{STICK\ FREE} + 57.3\,GS_e c_e \frac{\rho}{2} g l_t\left(\frac{n+1}{n}\right)\left[C_{h_\alpha} - \frac{C_{h\delta}}{T}\right] + KG \tag{46}$$

These equations give the stick force per "g" variations for an airplane in pull-ups or turns. The first term is the contribution due to stability, the second term is the contribution of the airplane's damping and the third term is the contribution of any weight moment in the elevator system.

As the first term is the only term which is affected by the airplane's stability, it is the only term affected by a shift of the airplane's center of gravity. It is convenient and useful to determine how the stick force per "g" variation changes with shift in c.g. This can be obtained very readily from equation (45).

$$\frac{d(F_s/n-1)}{d(\%\,c.g.\ shift)} = \frac{W/S\, G\, C_{h\delta} S_e c_e}{100\, C_{m\delta}} \tag{47}$$

At the stick-free neutral point, there is no stability contribution and all the stick-force per "g" is contributed by the damping term and the elevator weight moment.

$$\left(F_s/g\right)_{DAMPING\ (PULL\text{-}UP)} = 57.3\,G\,S_e c_e \frac{\rho}{2} g l_t\left[C_{h_\alpha} - \frac{C_{h\delta}}{T}\right] \tag{48}$$

$$\left(F_s/g\right)_{WEIGHT\ MOMENT} = KG \tag{49}$$

The stick force per "g" at any c.g. may be obtained rather conveniently from an equation combining (47), (48), and (49).

$$\left(\frac{F_s}{g}\right)_{PULL\text{-}UP} = 57.3\, G S_e c_e \frac{W}{S} \rho\, l_t \left[C_{h_\alpha} - \frac{C_{h_\delta}}{\tau}\right] + KG - \frac{W}{S}\frac{G S_e c_e}{C_{m_\delta}} C_{h_\delta} (c.g. - N_0') \tag{50}$$

and the maneuvering neutral point in pull-ups is obtained by equating (50) to zero and solving for the c.g.

$$N_m' = N_0' + \frac{57.3\, C_{m_\delta}}{\frac{W}{S} C_{h_\delta}} \left(\frac{\rho}{2} g\, l_t \left[C_{h_\alpha} - \frac{C_{h_\delta}}{\tau}\right] + \frac{K}{G S_e c_e\, 57.3} \right) \tag{51}$$

5. A convenient chart may be drawn which shows the whole longitudinal stability picture. This chart is drawn with F_s/g as the ordinate and c.g. position as abscissa. The stick-fixed neutral point, N_o, is determined from equation (6) which is rewritten here, plus the effects of power determined from table I

$$N_0 = a.c. - \frac{K_f w_f^2 L_f}{a_w S_w c} - \frac{K_n w_n^2 L_n N}{a_w S_w c} + \frac{a_t}{a_w} \bar{V} \eta_t \left(1 - \frac{d\epsilon}{d\alpha}\right) - \Delta P \tag{52}$$

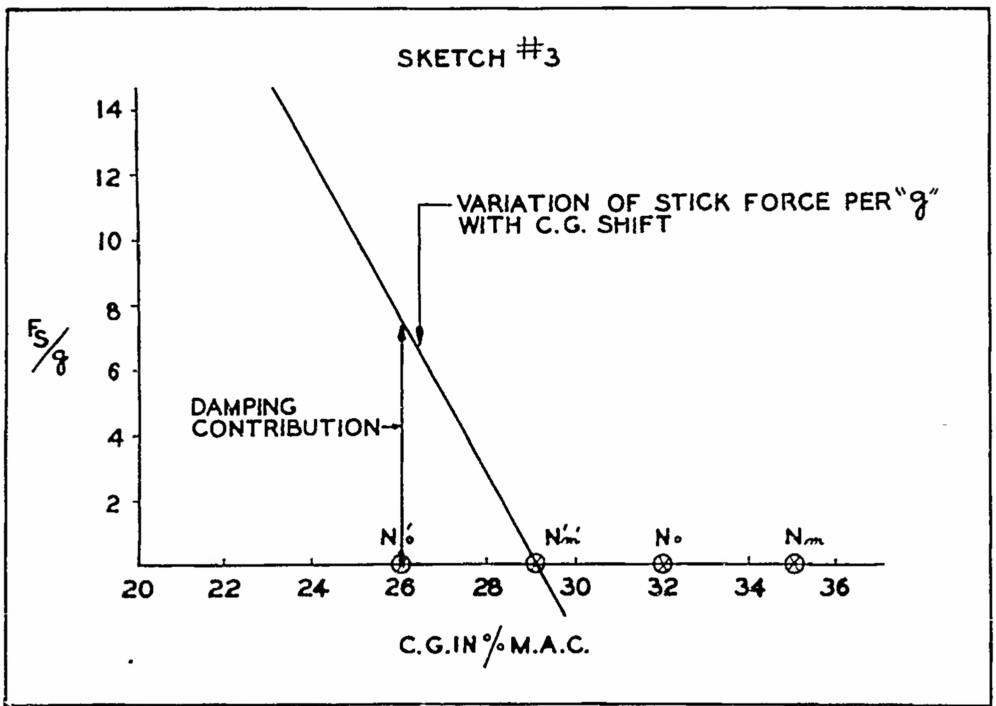
This c.g. is plotted on the curve on the zero F_s/g axis.

The stick-free neutral point N_o' is next determined from equation (23) using N_o as determined above.

$$N_0 - N_0' = \frac{a_t \bar{V} \tau \eta_t \left(1 - \frac{d\epsilon}{d\alpha}\right)}{a_w C_{h_\delta}} C_{h_\alpha} \tag{53}$$

This point is also plotted on the zero F_s/g axis. Next, the F_s/g contribution due to damping and weight moment, if any, are computed from equations (48 and 49), and these are added at the stick-free neutral point. The change of F_s/g per percent c.g. shift is determined from equation (47) and a straight line of this slope drawn through the F_s/g point determined from the damping and weight moment contributions. Where this line intersects the zero F_s/g axis is the maneuvering neutral point stick-free N_m'. As a check, this point can be obtained by using equation (51) to compute the difference between the stick-free neutral point N_o' and the stick-free maneuvering neutral point N_m'. Finally, the maneuvering stick-fixed neutral point is determined from equation (40). A typical plot for an arbitrary airplane is shown below.

The stick-fixed and stick-free neutral points (N_o and N_o') as developed are independent of altitude and lift



coefficient. The stick-fixed and stick-free maneuvering neutral points (N_m and N_m') are independent of speed, but vary with altitude, the effect being that increased altitude moves these neutral points forward, or in other words for any given c.g. an increase in altitude reduces both the gradients of F_s/g and δ_e/g seriously and is the main reason for pilots' complaints of loss of longitudinal stability with altitude.

6. The results of a comparison between calculated and wind-tunnel stick-fixed, propeller-off neutral points has been made and the results shown in figures 10 and 11. These comparisons indicate that the neutral point stick-fixed, propeller-off, can be predicted with an error of usually not more than 2 percent m.a.c. The biggest sources of error are in the determination of the wing aerodynamic center, and of the slope of the lift curve of the tail. This last is affected by the low Reynolds number of the model under test and the inaccuracy of the resulting extrapolation to the proper Reynolds number.

ILLUSTRATIVE EXAMPLE

A typical example is presented using the methods developed herein. The basic geometry of the airplane, a single-engine low-wing fighter, is as follows:

Wing Area	213 sq. ft.
Weight	7847 lb.
Wing span	34 ft.
Airfoil section, root	NACA 0015
Airfoil section, tip	NACA 23009
Aspect ratio	5.42
m.a.c.	6.72 ft.
Taper ratio	1.97
Wing incidence to thrust line	2°
Horizontal tail area	40.0 sq. ft..
Horizontal tail span	13 ft.
Tail incidence from thrust line	2¼° up
Elevator area aft hinge line	12.50 sq. ft.
Elevator balance area	3.65 sq. ft.
Tail length	16.2 ft.
Horizontal distance wing root chord trailing edge to elevator a.c.	10.0 ft.
Vertical distance wing root chord trailing edge to elevator hinge line	3.4 ft.
Horizontal distance wing root ¼ chord to elevator a.c.	16. 2 ft.
Ratio tail span to wing span	.383
Overall length of fuselage	30.1 ft.
Maximum width of fuselage	2.9 ft.
Mean chord of elevator aft of hinge line	1.13 ft.
Stick, elevator gearing	.607 rad/ft.
Elevator effectiveness from figure 8	$\tau = .51$

Hinge moment parameters (from wind-tunnel test) $C_{h_\alpha} = -.002$, $C_{h_\delta} = -.0041$

Elevator weight moment $KG = O$

1. Determination of stick-fixed neutral point (N_0)
Use equation (6)

$$N_0 = a.c. - \frac{K_f w_f^2 L_f}{a_w S_w c} + \frac{a_t}{a_w}\bar{V}\eta_t\left(1-\frac{d\epsilon}{d\alpha}\right) - \Delta P$$

a. To determine a. c.
Airfoil section data
Root NACA 0015 $(a.c.)_0 = .238$
Tip NACA 23009 $(a.c.)_0 = .241$

$$a.c._{wing} = \frac{(a.c.)_{0_{ROOT}} \times T.R. + (a.c.)_{0_{TIP}}}{T.R. + 1}$$

$$a.c._{wing} = \frac{.238 \times 1.97 + .241}{1.97 + 1} = .239$$

b. To determine fuselage term
Location root ¼ chord point in per cent body length $= .36$
From figure 7, $K_f = .82$
From figure 1, $a_w = .072$

$$\therefore \text{FUSELAGE TERM} = \frac{K_f w_f^2 L_f}{a_w \times S_w \times c \times 57.3} = \frac{.82 \times 2.9^2 \times 30.1}{.072 \times 57.3 \times 213.0 \times 6.72} = .036$$

c. To determine tail term
From figure 2, $a_t = .058$

$$\bar{V} = \frac{S_t l_t}{S_w c} = .452$$

$$m_0 = \frac{\text{Vert. dist. root chord trailing edge to horiz. tail a.c.}}{\text{semi-span}} = .2$$

$$x_t = \frac{\text{Horiz. dist. root chord trailing edge to horiz. tail a.c.}}{\text{semi-span}} = .588$$

$$x = \frac{\text{Horiz. dist. root ¼ chord point horiz. tail a.c.}}{\text{semi-span}} = .953$$

From figure 4 $\frac{d\epsilon}{d\alpha} = .47$ aspect ratio 6 T.R. 2:1

From figure 5 $\frac{d\epsilon}{d\alpha} = .465$ correct for taper ratio

From figure 5 $\frac{d\epsilon}{d\alpha} - .495$ correct for aspect ratio

From figure 6 $\frac{d\epsilon}{d\alpha} = .45$ correct for average downwash

d. To determine ΔP
From Table I $\qquad \Delta P = .06$

$$N_o = .239 - .035 + .180 - .06 = .324$$

2. Determination of stick-free neutral point (N_o')

$$N_o' = N_o - \frac{a_t \bar{V} T \eta_t (1 - d\epsilon/d\alpha) C_{h\alpha}}{a_w C_{h\delta}}$$

$$N_o' = N_o - \frac{.058 \times .452 \times .51 \times .9 \times .55 \times -.002}{.072 \times -.0041}$$

$$N_o' = .324 - .045 = .278$$

3. Determination of stick-fixed maneuvering neutral point—(pull-ups) (N_m)

$$N_m = N_o + \frac{a_t \bar{V}}{2\mu} \qquad \mu = \frac{m}{\rho S l_t} = 29.7 \text{ at see level}$$

$$N_m = .324 + \frac{.058 \times .452 \times 57.3}{2 \times 29.7}$$

$$N_m = .324 + .025 = .349$$

4. Determination of stick-free maneuvering neutral point—pull-ups (N_m')

$$C_{m\delta} = -a_t T \bar{V} \eta_t = -.058 \times .51 \times .452 \times .9 = -.0127$$

$$N_m' = N_o' + \frac{57.3 C_{m\delta}}{W/S C_{h\delta}} \left(\frac{\rho}{2} g l_t \left[C_{h\alpha} - \frac{C_{h\delta}}{T} \right] \right)$$

$$N_m' = .278 + \frac{57.3 \times -.0127}{36.8 \times .0041} \left(\frac{.002378}{2} \times 32.2 \times 16.2 \left[-.002 + \frac{.0041}{.51} \right] \right)$$

$$N_m' = .27 + .018 = .296$$

5. Development of stability diagram.

$$\left(F_S/g\right)_{DAMPING\,(PULL\text{-}UP)} = G S_e c_e \frac{\rho}{2} g l_t \left[C_{h\alpha} - \frac{C_{h\delta}}{T} \right] 57.3$$

$$= \frac{.607 \times 12.5 \times 1.13 \times \rho \times 16.2 \times 32.2 \times .006 \times 57.3}{2}$$

$$= 765 \rho = 1.820 \#/g \text{ at see level}$$

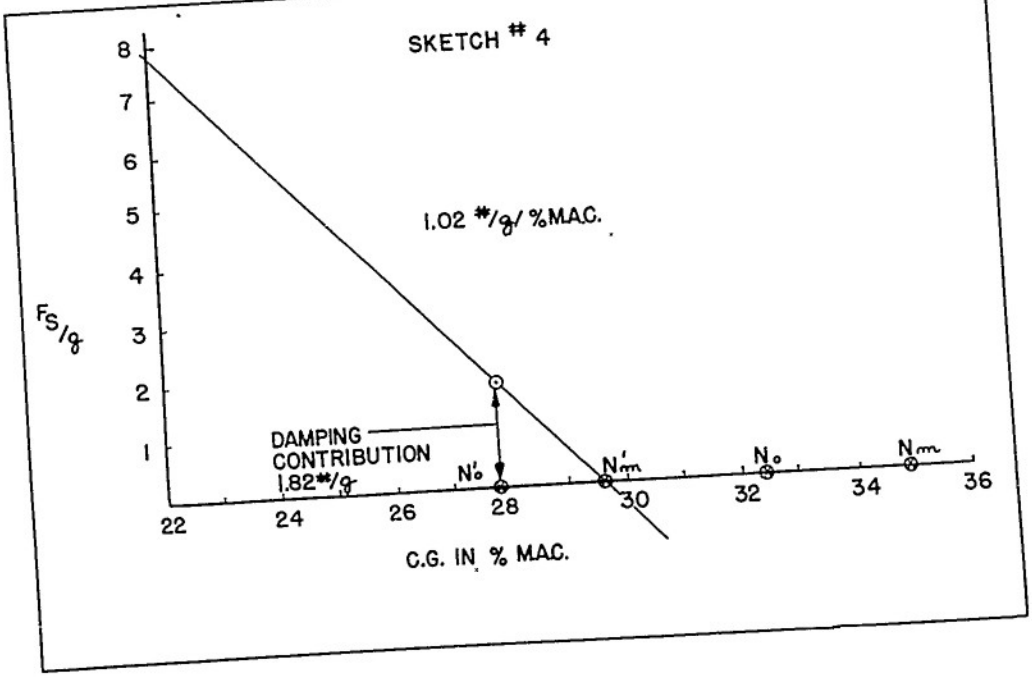
$$\frac{d(F_S/g)}{d(\%\text{ c.g. shift})} = \frac{W/S\, G C_{h\delta} S_e c_e}{100 C_{m\delta}}$$

$$= \frac{36.8 \times .607 \times -.0041 \times 1.13 \times 12.50}{100 \times -.0127}$$

$$= 1.02 \#/g/\%\text{ c.g. SHIFT}$$

NOMENCLATURE

M — Moment, lb. ft.
C — Chordwise force, lb.
N — Normal force, lb.
L — Lift force, lb.



S_w — Wing area, sq. ft.
c — Mean aerodynamic chord, ft.
a.c. — Wing aerodynamic center
$(a.c.)_o$ — Airfoil section aerodynamic center
c.g. — Airplane center of gravity
x_a — Horizontal distance a.c. to c.g.
z_a — Vertical distance a.c. to c.g.
A — Aspect ratio
T.R. — Taper ratio
l_t — Distance c.g. to a.c. of horizontal tail
S_t — Area horizontal tail, sq. ft.
L_f — Fuselage length, ft.
L_n — Nacelle length, ft.
N — Number of nacelles
w_f — Maximum width fuselage, ft.
w_n — Maximum width nacelle, ft.
K_f or K_n — Fuselage, nacelle factor
b_e — Elevator span, ft.
S_e — Elevator area aft of hinge line, sq. ft.
c_e — Mean chord of elevator aft of hinge line, ft.
ρ — Air density, slugs/cu. ft.
q — Dynamic pressure, lb/sq. ft.
V — Free stream velocity, ft/sec.
m — Mass of airplane, slugs
g — Acceleration due to gravity, ft/sec.^{2}
n — Airplane load factor
Q — Airplane angular velocity about Y axis, radians/sec.
μ — Airplane density factor
K — Elevator weight moment, lb. ft.
$\bar{V}$ — Tail volume coefficient
ΔP — Neutral point shift due to propeller and power
F_s — Force applied at top of stick, lb.
G — Stick, elevator gearing, radians per ft.
a_w — Slope of lift curve of wing, per degree
a_t — Slope of lift curve of horizontal tail, per degree
α_w — Geometric wing angle of attack, degrees
α_t — Tail angle of attack from zero lift, degrees
α_o — Wing angle of attack for zero lift, degrees
i_t — Incidence of horizontal tail from reference line, degrees
α_s — Angle of attack of stabilizer, degrees
ϵ — Downwash angle, degrees
δ_e — Elevator deflection, degrees
δ_{eo} — Elevator angle for airplane trim at zero lift
δ_t — Elevator tab angle, degrees
η_t — Tail efficiency
C_m — Moment coefficient
C_n — Normal force coefficient
C_L — Lift coefficient
C_{Lt} — Tail lift coefficient
HM_e — Elevator hinge moment, lb. ft
C_{h_e} — Elevator hinge moment coefficient
$C_{h_{e_o}}$ — Elevator hinge moment coefficient. $\delta_e = 0, \alpha_s = 0$
K — Elevator weight moment, lb. ft.
τ — Elevator effectiveness
C_{h_α} — $\left(\frac{\partial C_{h_e}}{\partial \alpha_s}\right)_{\delta_e, \delta_t}$ per degree
C_{h_δ} — $\left(\frac{\partial C_{h_e}}{\partial \delta_e}\right)_{\alpha_s, \delta_t}$ per degree
$C_{h_{\delta t}}$ — $\left(\frac{\partial C_{h_e}}{\partial \delta_t}\right)_{\delta_e, \alpha_s}$ per degree
C_{m_δ} — $\left(\frac{dC_m}{d\delta_e}\right)$ per degree
N_o — Stick-fixed neutral point
N_o' — Stick-free neutral point
N_m — Maneuvering stick-fixed neutral point
N_m' — Maneuvering stick-free neutral point
m_o — $\frac{\text{Vertical distance root chord trailing edge to horiz. tail a.c.}}{\text{semi-span}}$
X_t — $\frac{\text{Horizontal distance root chord trailing edge to horiz. tail a.c.}}{\text{semi-span}}$
X — $\frac{\text{Horizontal distance root 1/4 chord to horizontal tail a.c.}}{\text{semi-span}}$

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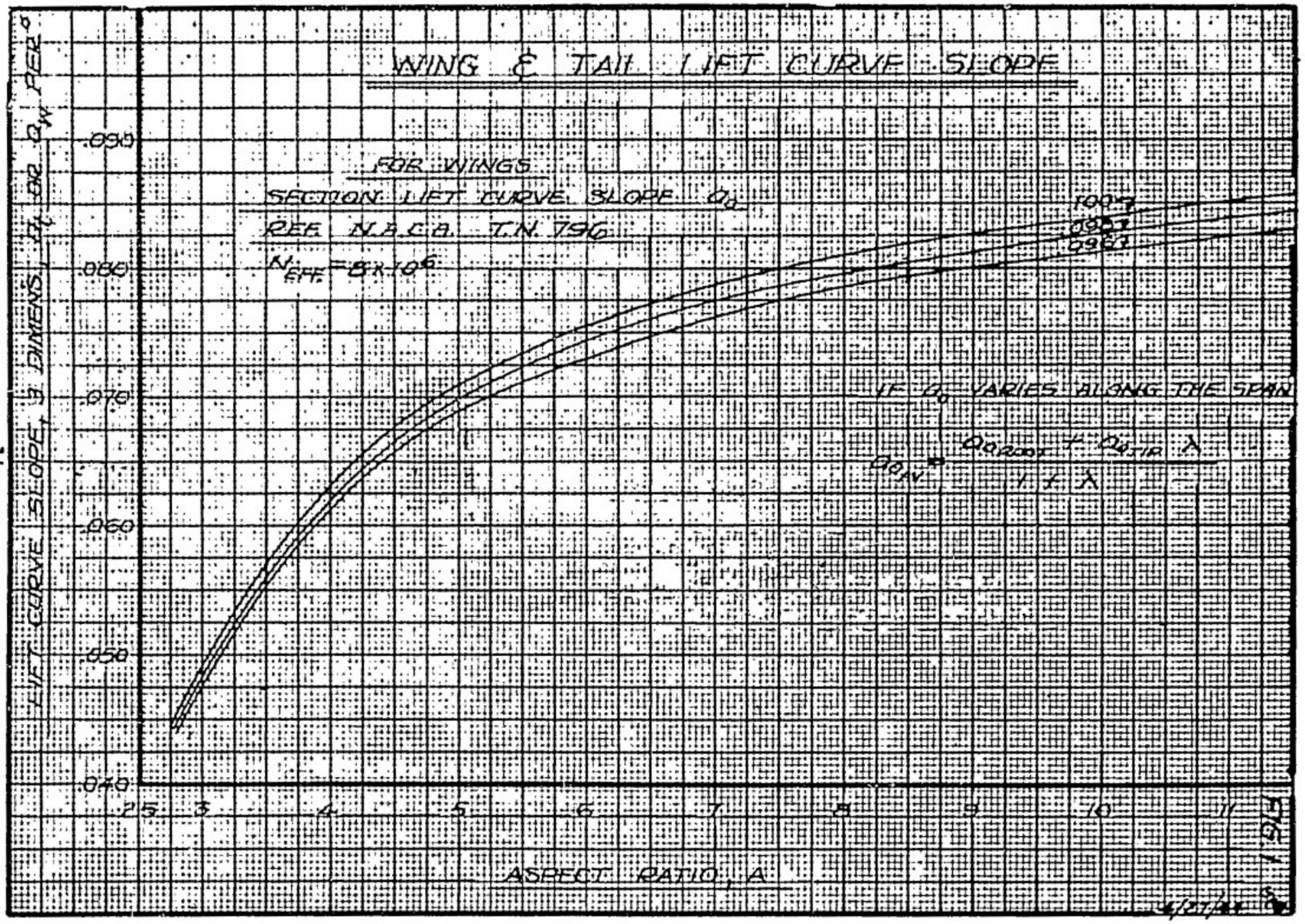
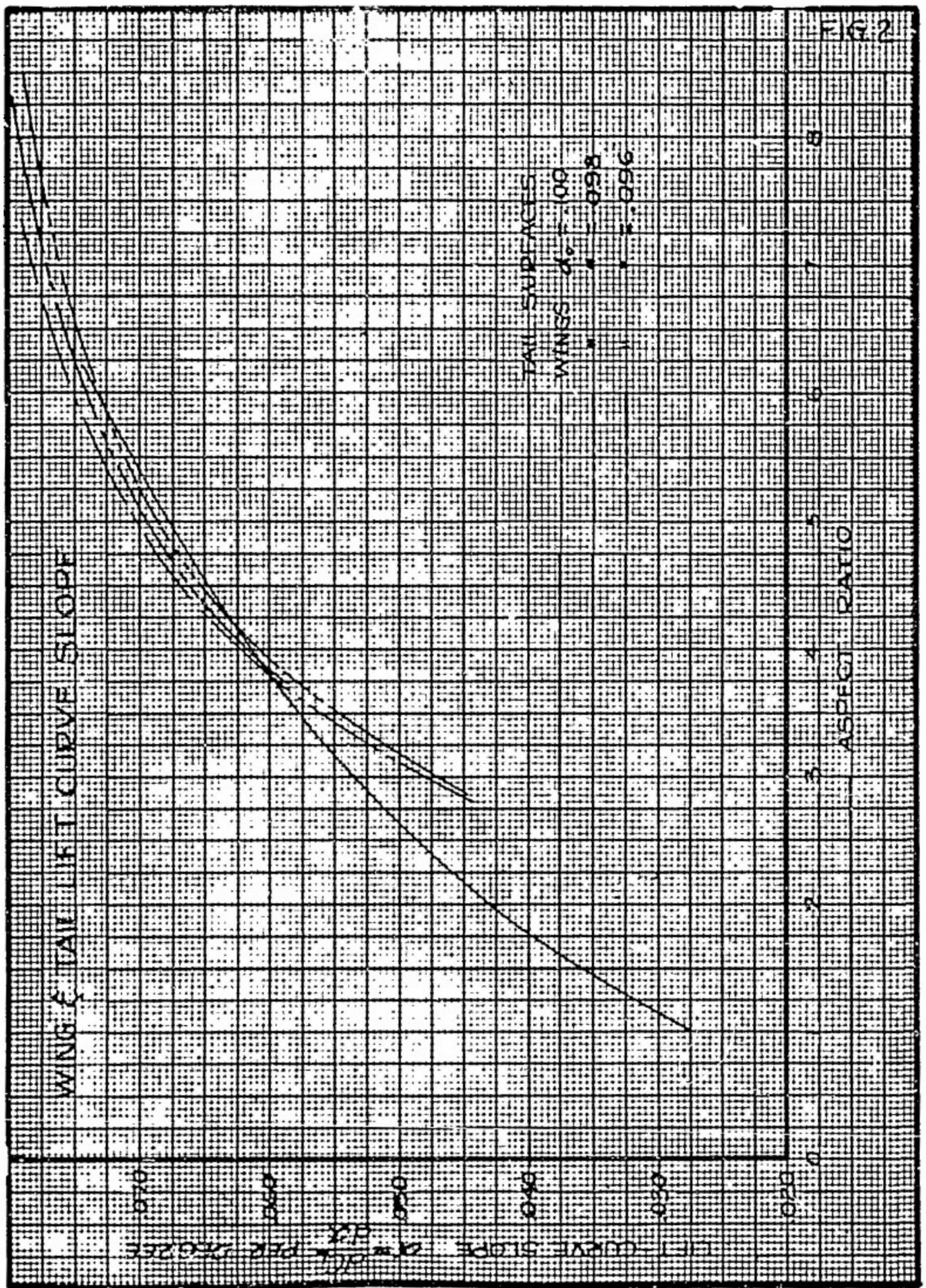
3. Sears, R. I., Wind-Tunnel Data on the Aerodynamic Characteristics of Airplane Control Surfaces. NACA A. C. R. 3108.



WING & TAIL LIFT CURVE SLOPE
FOR WINGS
SECTION LIFT CURVE SLOPE a_0
REF. N.A.C.A. T.N. 796
$N_{EFF.} = 8 \times 10^6$
IF a_0 VARIES ALONG THE SPAN
$a_{0_{AV}} = \frac{a_{0_{ROOT}} + a_{0_{TIP}} \lambda}{1 + \lambda}$
.090
.080
.070
.060
.050
.040
2.5
3
4
5
6
7
8
9
10
11
LIFT CURVE SLOPE, 3 DIMENS. a_t OR a_W PER°
ASPECT RATIO, A
FIG. 1

5167


WING & TAIL LIFT CURVE SLOPE
FIG. 2
TAIL SURFACES
WINGS a0 = .100
" = .098
" = .096
ASPECT RATIO
0
2
3
4
5
6
7
8
.070
.060
.050
.040
.030
.020
LIFT CURVE SLOPE a = dCL/dα PER DEGREE

5167

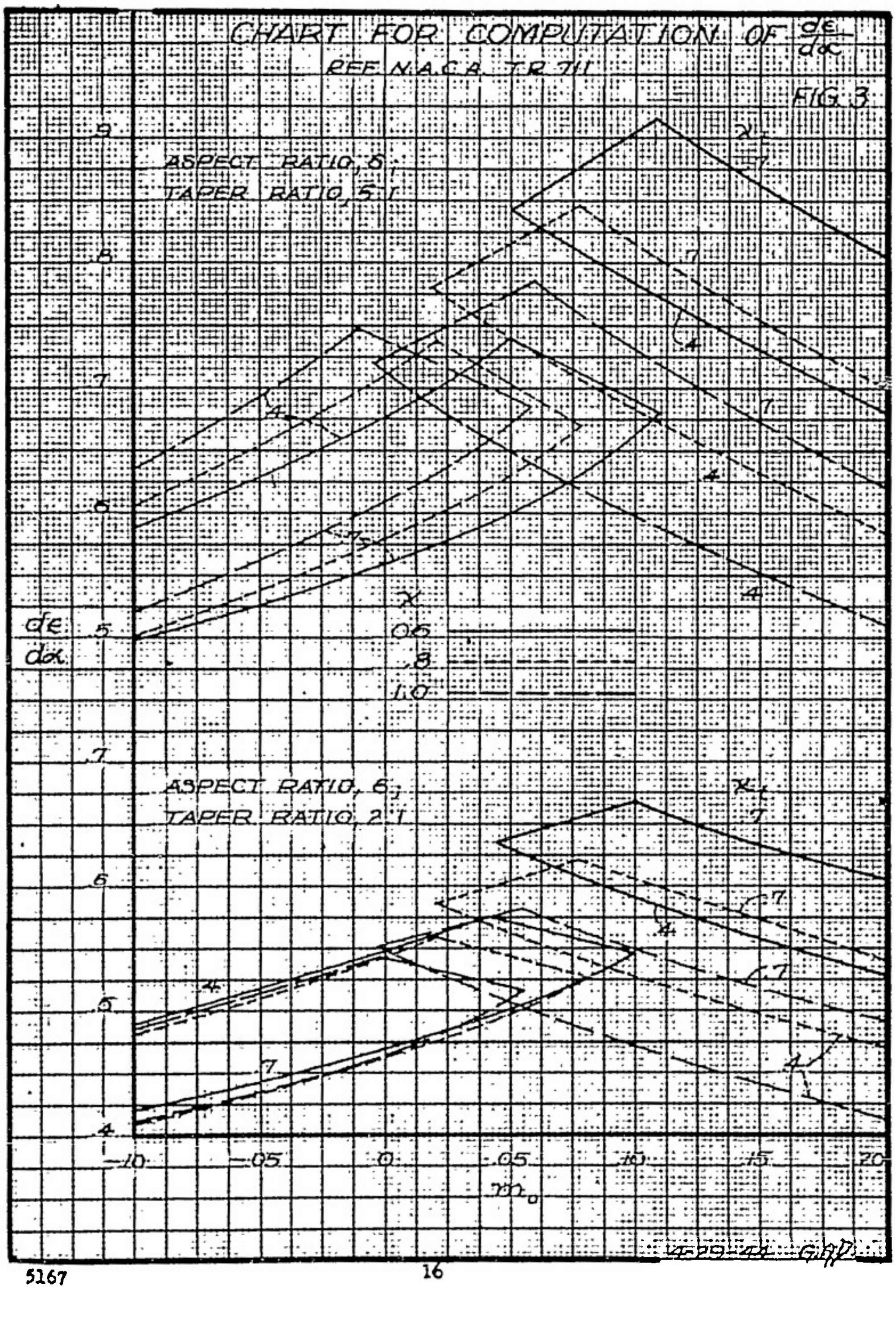


CHART FOR COMPUTATION OF $\frac{d\epsilon}{d\alpha}$
REF. N.A.C.A. T.R. 711
FIG. 3
ASPECT RATIO, 6:1
TAPER RATIO, 5:1
ASPECT RATIO, 6:1
TAPER RATIO, 2:1
$\frac{d\epsilon}{d\alpha}$
λ
0.6
.8
1.0
m_0
4-29-44 GAD

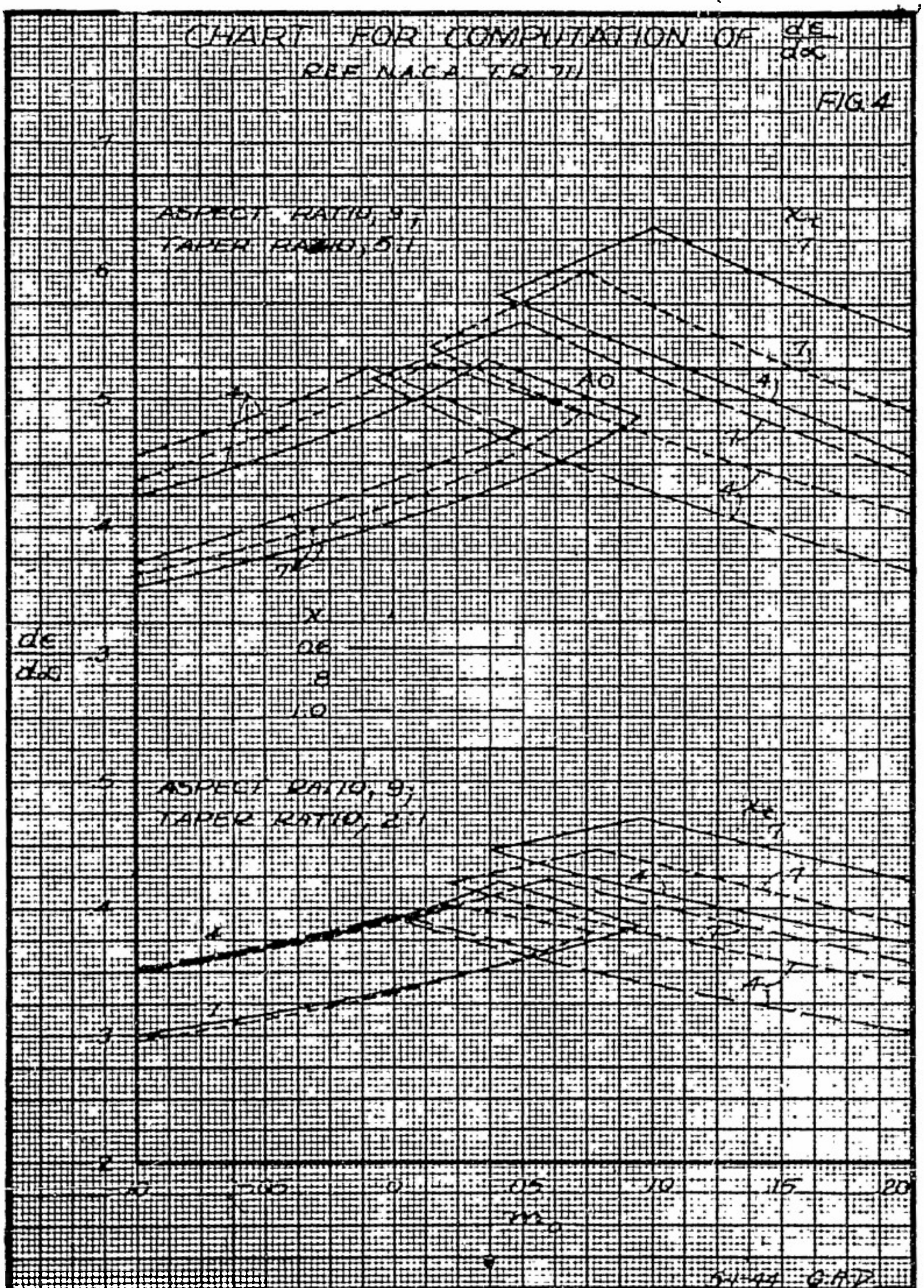


CHART FOR COMPUTATION OF dε/dα
REF. N.A.C.A. T.R. 711
FIG. 4
ASPECT RATIO, 9; TAPER RATIO, 5:1
ASPECT RATIO, 9; TAPER RATIO, 2:1
dε/dα
m₀
6-1-44 G.H.D.

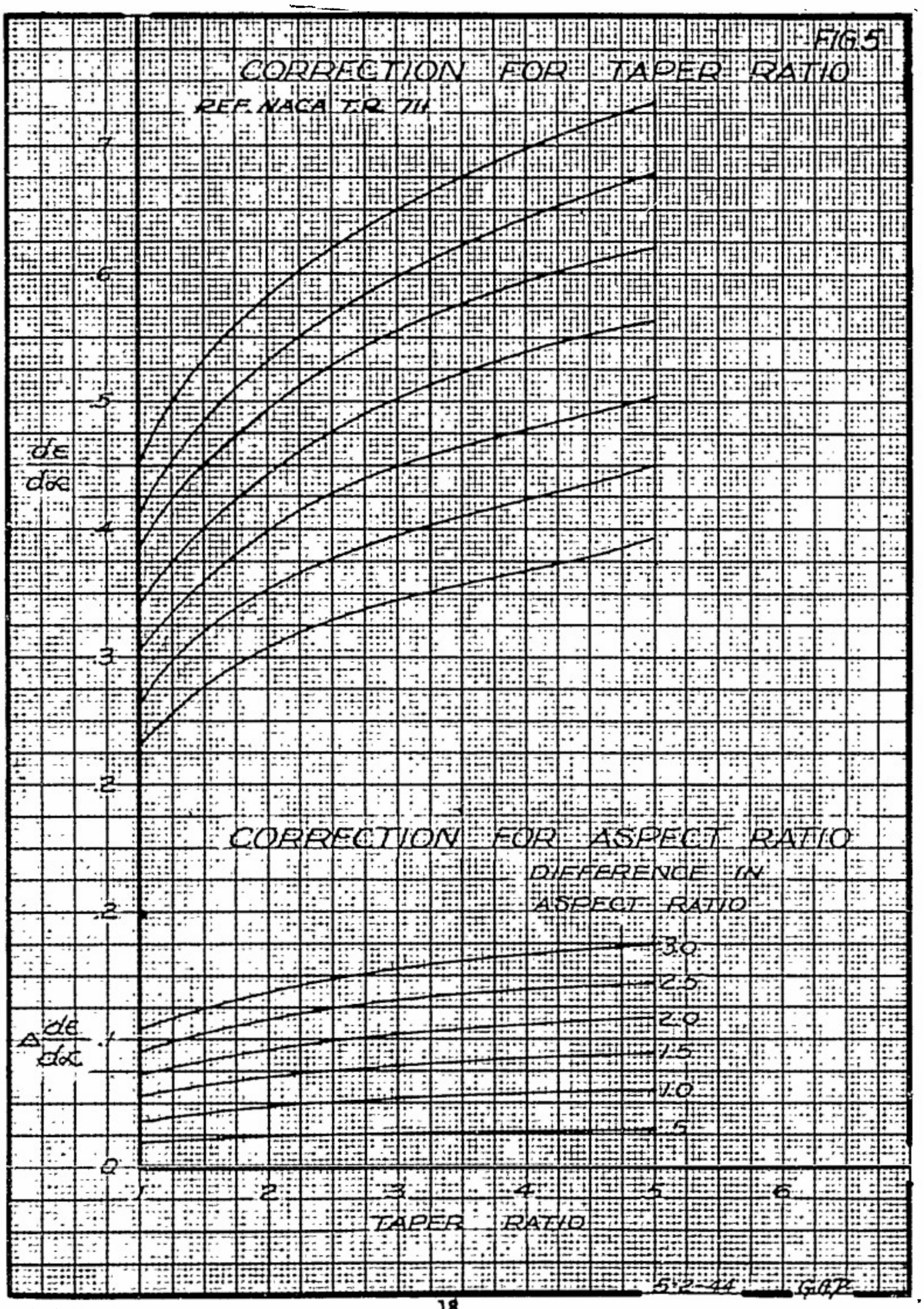
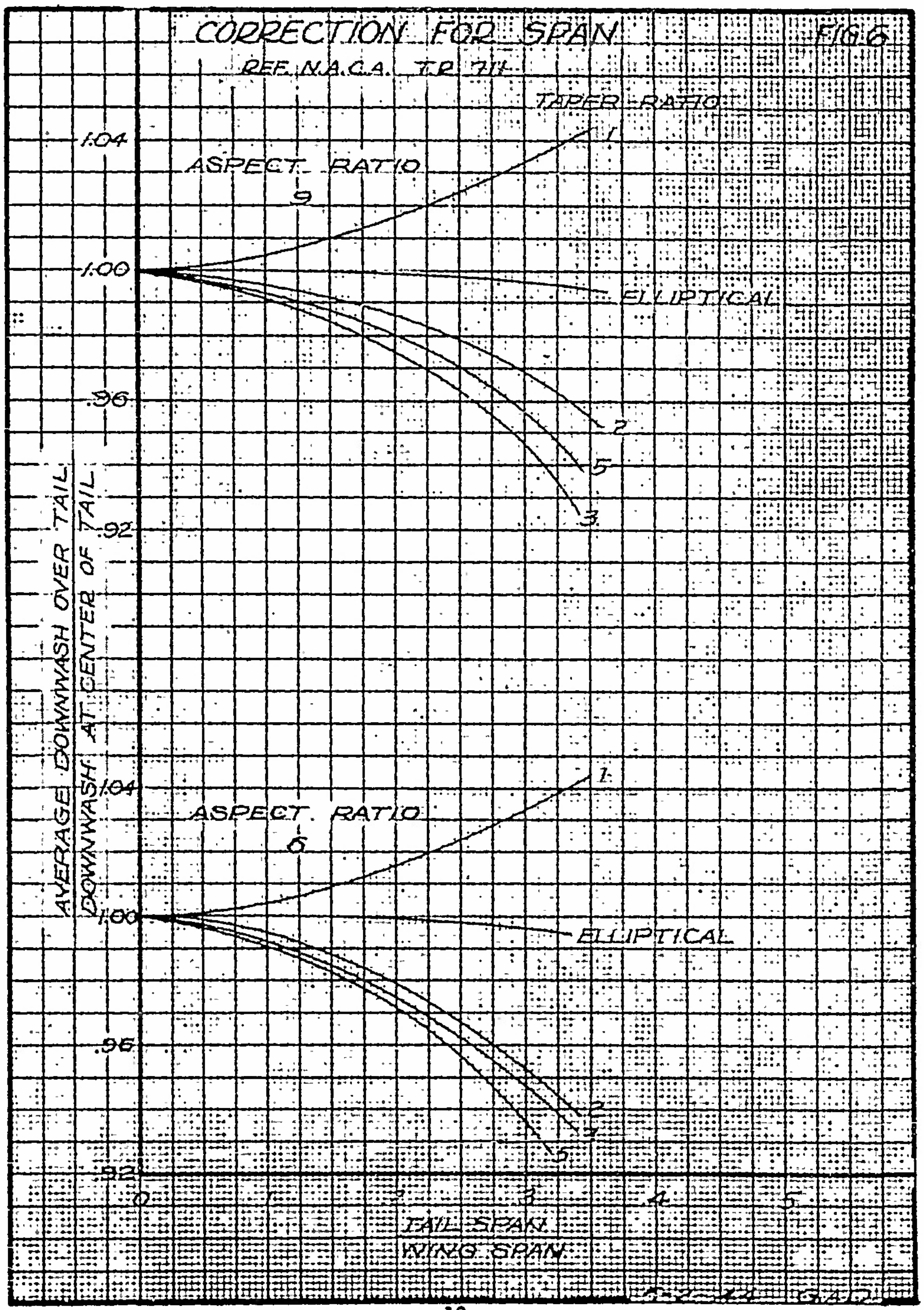
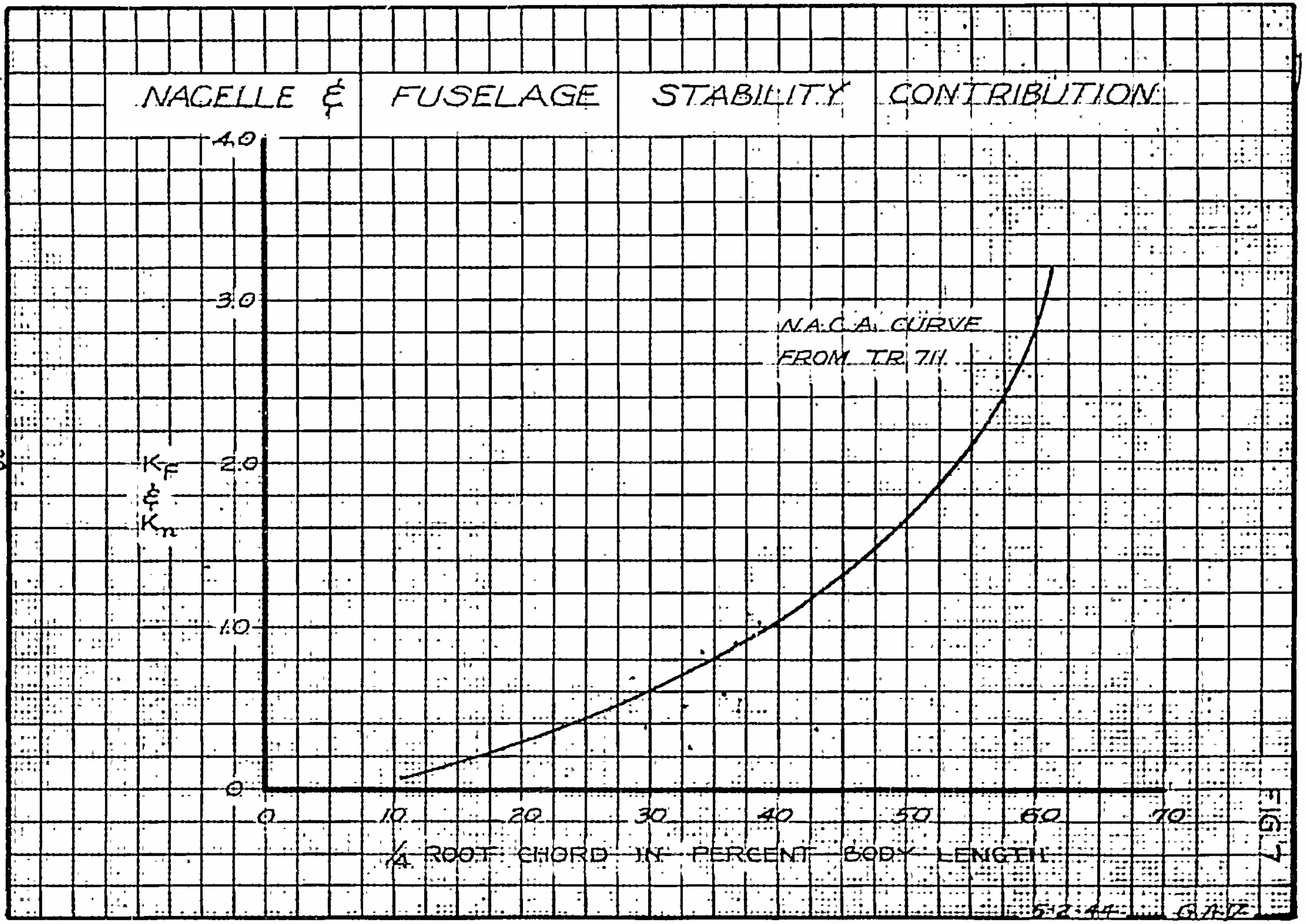


FIG 5
CORRECTION FOR TAPER RATIO
REF. NACA T.R. 711
CORRECTION FOR ASPECT RATIO
DIFFERENCE IN ASPECT RATIO
3.0
2.5
2.0
1.5
1.0
.5
TAPER RATIO
5-2-44
G.A.P.

5167


CORRECTION FOR SPAN
FIG 6
REF. N.A.C.A. T.R. 711
TAPER RATIO
ASPECT RATIO
9
ASPECT RATIO
6
ELLIPTICAL
1.04
1.00
.96
.92
0
1
2
3
4
5
AVERAGE DOWNWASH OVER TAIL
DOWNWASH AT CENTER OF TAIL
TAIL SPAN
WING SPAN


NACELLE & FUSELAGE STABILITY CONTRIBUTION
N.A.C.A. CURVE
FROM T.R. 711
K_F & K_n
4.0
3.0
2.0
1.0
0
0
10
20
30
40
50
60
70
1/4 ROOT CHORD IN PERCENT BODY LENGTH
FIG. 7
5-2-44 G.H.Z.

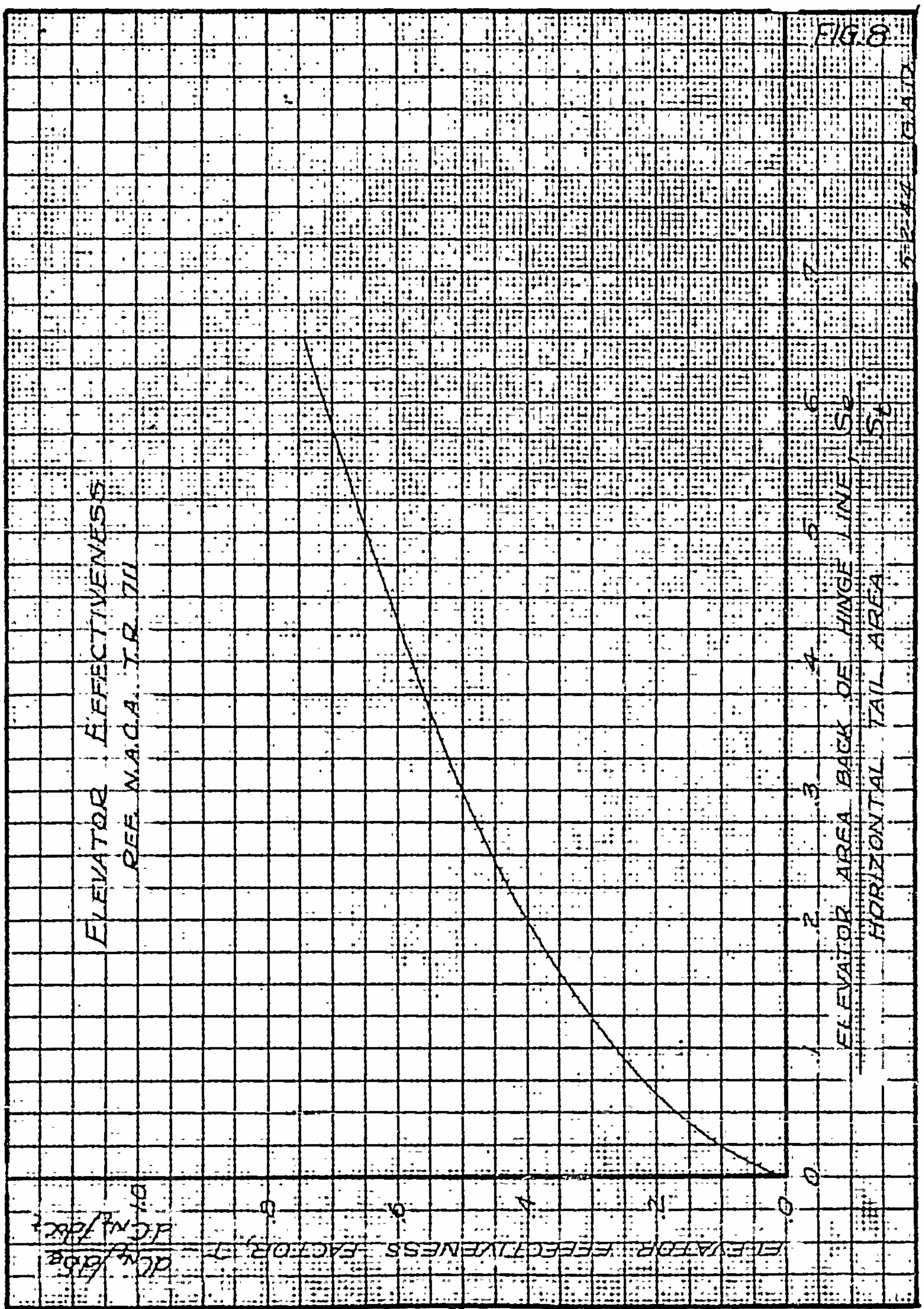


FIG. 8
ELEVATOR EFFECTIVENESS
REF. N.A.C.A. T.R. 711
ELEVATOR AREA BACK OF HINGE LINE / Se
HORIZONTAL TAIL AREA / St
0
.1
.2
.3
.4
.5
.6
.7
ELEVATOR EFFECTIVENESS FACTOR, τ
0
.2
.4
.6
.8
1.0

5757

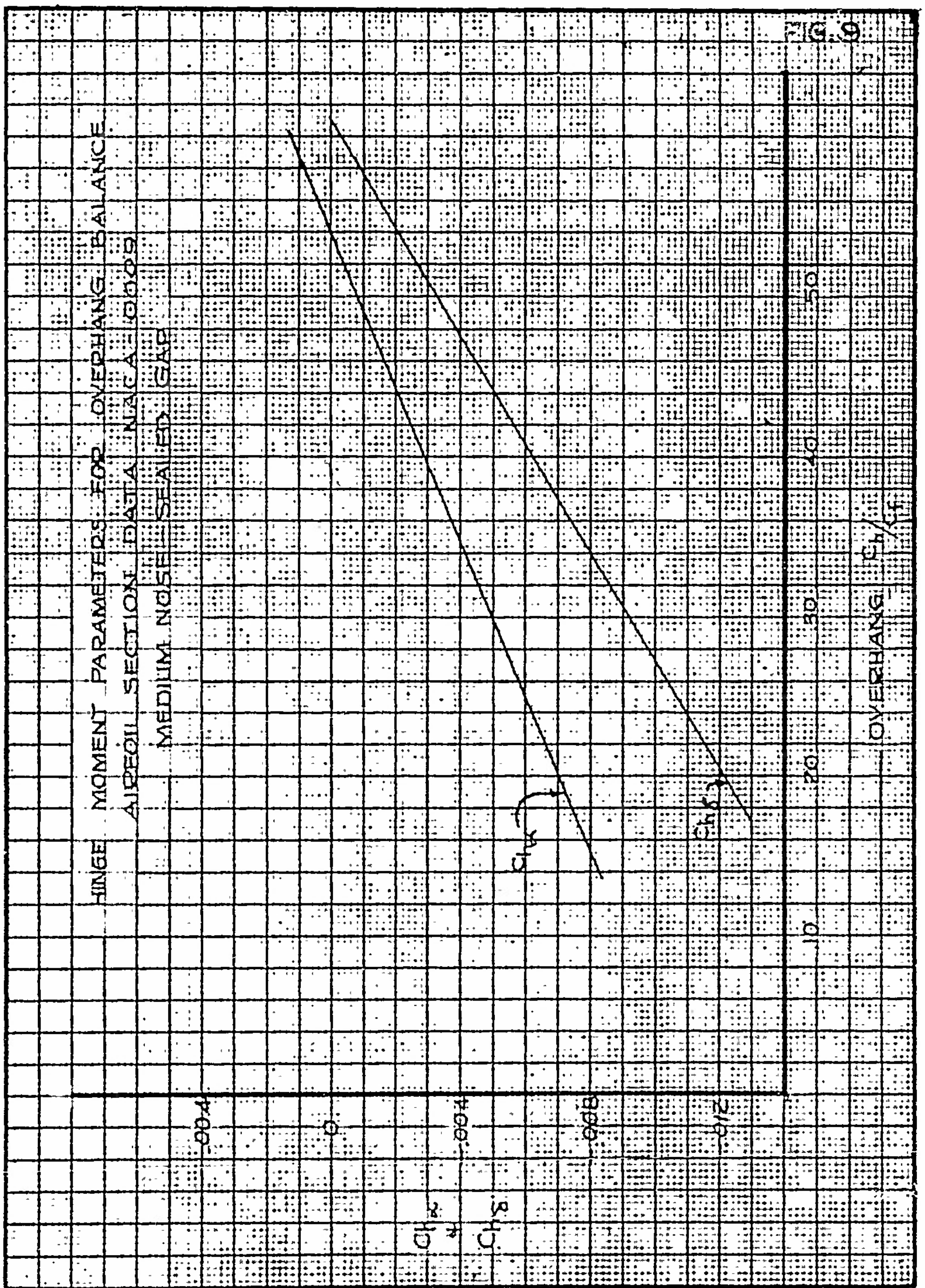
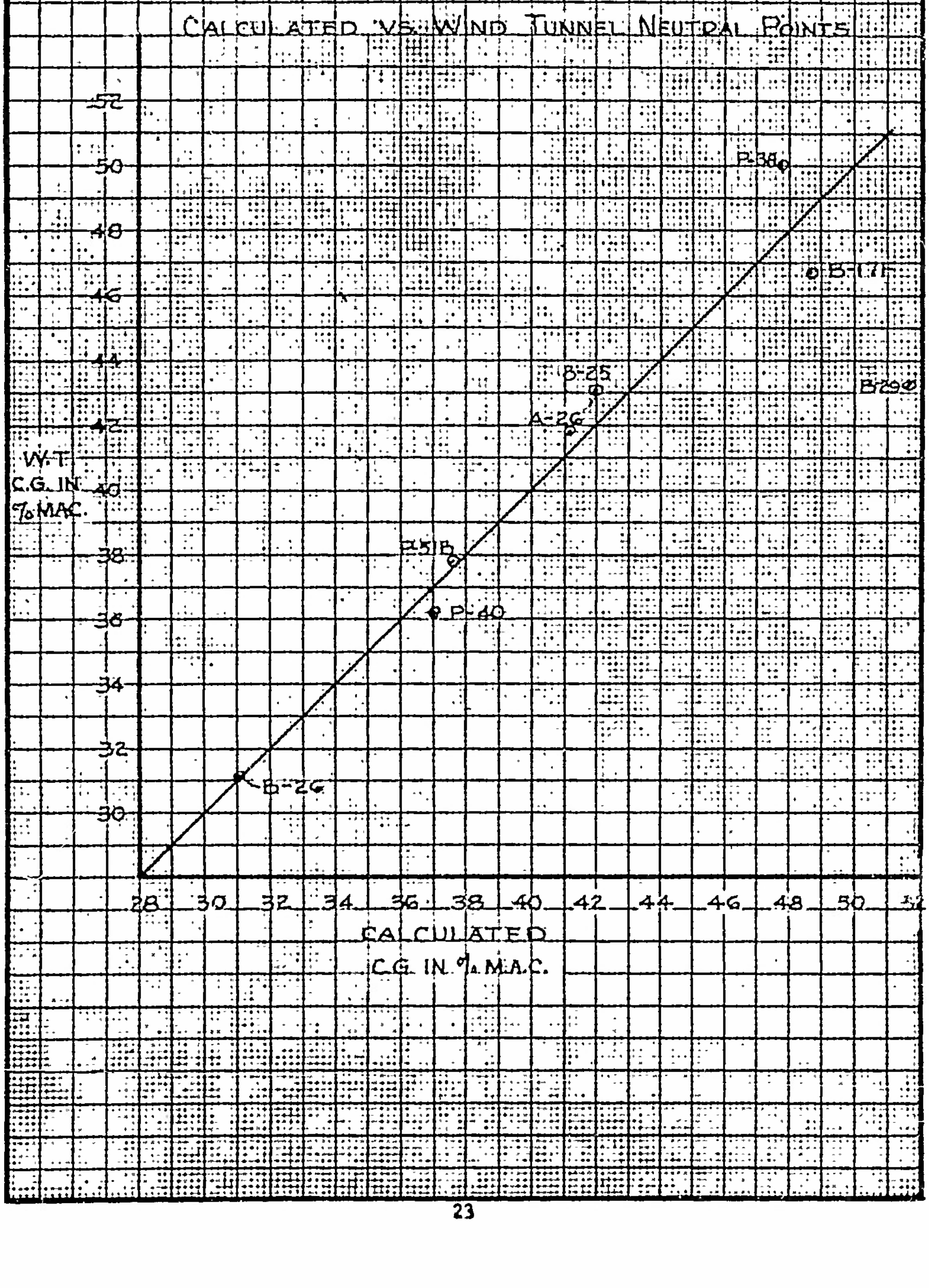
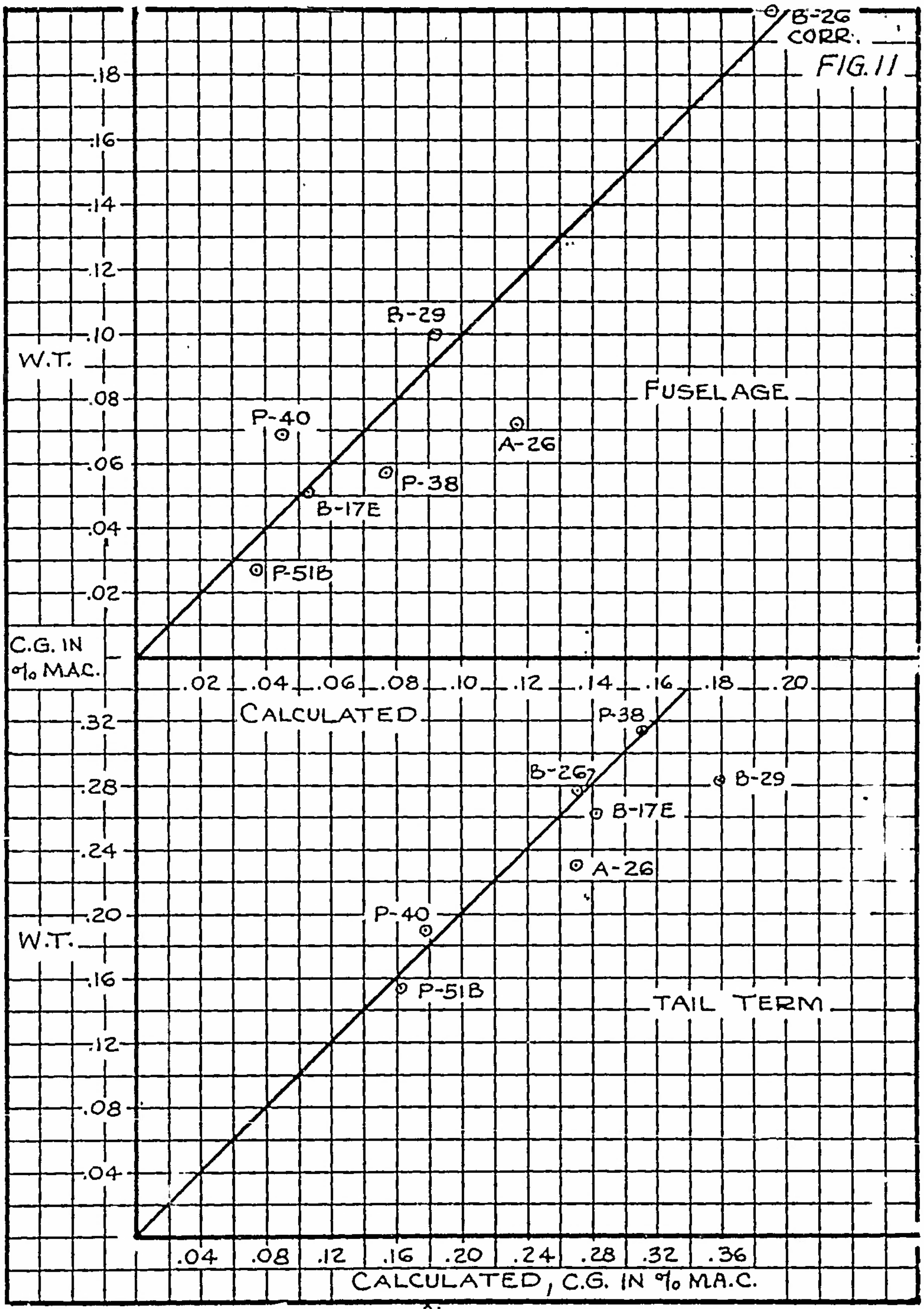


FIG. 9
HINGE MOMENT PARAMETERS FOR OVERHANG BALANCE
AIRFOIL SECTION DATA NACA 0009
MEDIUM NOSE SEALED GAP
OVERHANG C_b/C_f
10
20
30
40
50
-.004
0
.004
.008
.012
C_{h_α}
C_{h_δ}
C_{h_α} & C_{h_δ}

FIG. 10


CALCULATED VS. WIND TUNNEL NEUTRAL POINTS
W.T. C.G. IN % MAC.
52
50
48
46
44
42
40
38
36
34
32
30
28
30
32
34
36
38
40
42
44
46
48
50
52
CALCULATED C.G. IN % M.A.C.
P-38
B-17F
B-25
B-29
A-26
P-51B
P-40
B-26

167


B-26 CORR.
FIG. 11
.18
.16
.14
.12
.10
W.T.
.08
.06
.04
.02
C.G. IN % MAC.
B-29
P-40
A-26
P-38
B-17E
P-51B
FUSELAGE
.02 .04 .06 .08 .10 .12 .14 .16 .18 .20
CALCULATED
.32
.28
.24
.20
W.T.
.16
.12
.08
.04
P-38
B-26
B-29
B-17E
A-26
P-40
P-51B
TAIL TERM
.04 .08 .12 .16 .20 .24 .28 .32 .36
CALCULATED, C.G. IN % M.A.C.

5167

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