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A General Treatment of Static Longitudinal Stability with Propellers, with Application to Single-Engined Aircraft

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A General Treatment of Static Longitudinal Stability with Propellers, with Application to Single-Engined Aircraft

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Summary.—A general method of treatment of stick-fixed static longitudinal stability with propellers is given, distortion and compressibility effects being neglected.

Model full-throttle data on some single-engined fighters are analysed for the flaps-up condition to establish a basis of estimation of effect of propeller on stability for this type of design.

The general effect of propellers on manœuvre point, more particularly the effect on $H_m - K_n$, is considered in an appendix.

Conclusions

(1) The method given of stability analysis for single or multi-engined aeroplanes should prove simpler than earlier methods.

(2) From analysis of model tests on single-engined fighters tentative empirical factors have been obtained for estimating the full throttle stability in terms of that without propeller, for the flaps-up condition :—

(*a*) To estimate values of C_L (for aeroplane less tail) it seems sufficient merely to add the appropriate component of force on the propeller, calculated as if the propeller were acting alone, to the C_L without propeller.

(*b*) The model results indicate that, excluding the effect of thrust moment, the stability without tail is better at full throttle than for $T_c = 0$ over an incidence range including normal cruise and climb. This favourable effect of full throttle is attributed mainly to change of wing C_{m0} due to velocity increase in the slipstream and may be as much as $0 \cdot 04\bar{c}$ in neutral point position for climbing flight. It may be estimated very roughly by the method given in section 4.2, which expresses the effect as an equivalent change of thrust-line height : *viz.*

$$\text{Effective } z_p - \text{actual } z_p \simeq \frac{8}{\pi} \times \frac{S_s c_s}{2D^2 c} \times C_{m0s}$$

(*c*) For slope of tail-lift curve we suggest the multiplying factor $1 + 1 \cdot 5\, T_c$ as giving the effect of the slipstream

(*d*) For downwash derivative at the tail the data give

$$\frac{(1 - d\varepsilon/d\alpha)_{\text{Full throttle}}}{(1 - d\varepsilon/d\alpha)_{T_c = 0}} \simeq 1 - 6 \cdot 2\, T_c$$

Taken in conjunction with the result of Ref. 2 this gives

$$\frac{(1 - d\varepsilon/d\alpha)_{\text{Full throttle}}}{(1 - d\varepsilon/d\alpha)_{\text{No propeller}}} \simeq \left(1 - 1 \cdot 4\, \frac{dNc}{d\theta}\right)(1 - 6 \cdot 2\, T_c)$$

These formulae should not be used for values of T_c greater than about $0 \cdot 1$.

The effect on stability of downwash change due to the propeller is very much greater than the effect of variation of the velocity factor R from unity.

* R.A.E. Report Aero. 1944, received 16th August, 1944.

(3) The algebra of the Appendix shows that at high speed the difference of effects of propeller on manœuvre and stability margins should be small. At all speeds it will be algebraically greater for large than for small aircraft of the same geometry and the difference will increase with reduction of speed, at constant throttle.

When manœuvre point is required from model tests, these should be made at a number of values of T_c, the same values being taken at all incidences instead of using single T_c-values or non-overlapping T_c-ranges at the different incidences.

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1. *Introduction.*—Attempts have been made over a period of many years to establish general methods of estimation of the effect of propellers on longitudinal stability for multi-engined aircraft.

Bryant and McMillan (R. & M. 2310[1]) tackled the problem for the twin-engined aircraft, by carrying out a systematic programme of tests in the National Physical Laboratory Duplex Wind Tunnel on a model having the general proportions of the *Blenheim*. Among the quantities varied were tail span and height and propeller blade angle. Measurements of lift, drag and pitching moments were made over a range of values of wing incidence α and propeller thrust coefficients T_c for the model with and without tail, in the former case with the tail set at various angles. In addition, to give an idea of the physical nature of the slipstream a number of total-head surveys were made in the tailplane region for various combinations of α and T_c.

From these experimental results is devised a method of estimating effect of propellers on stability for twin-engined aircraft, the algebra of this method being applicable to aircraft other than twin-engined, though the numerical content is not. In spite of the generality of this algebra its complication renders it difficult of application, and this report presents a simpler treatment (section 3), which includes moreover the effect of the force on the propeller normal to its axis (the so-called propeller ' fin effect ').

At the same time it was felt that sufficient power-on model tests existed on single-engined aircraft to enable a tentative method of estimation of the effect of the propeller on stability to be established. This effect was investigated by the present author in Ref. 2 for single-engined aircraft with propeller at zero thrust, so that the numerical part of the present report (section 4) is an extension of Ref. 2 and utilises the same model tests.

2. *Notation.*

$\bar{c}$	length of wing mean chord = gross area ÷ span
C_m	pitching-moment coefficient about centre of gravity (h, k) of the aeroplane
C_{m0}, C_{mw}, C_{mw0}	pitching-moment coefficients of aeroplane less tail about the aerodynamic centre $(h_0, 0)$ without tail, the c.g. (h, k) and the point (h_0, k) respectively. We assume C_{m0} and h_0 are unaffected by the slipstream. (*See* however the discussion of section 4.2)
h_n, k	neutral point position, $H_n = h_n - h$ = c.g. margin
h_m, k	manœuvre point position, $H_m = h_m - h$ = manœuvre margin

$-dC_m/dC_L = K_n$ = static stability margin

The dimensionless co-ordinates h, h_0, h_n, h_m and k are referred to axes through the leading edge of the wing standard mean chord perpendicular to and along this chord and are ratios of actual lengths to the mean chord length c; k is positive if below the chord

C_L lift coefficient of aeroplane less tail assumed nearly equal to the lift coefficient of the aeroplane

$T_c, N_c = \dfrac{\text{propeller thrust, basic normal force}}{\rho V^2 D^2}$, the normal force being measured positive upwards

$\varkappa N_c$ value of normal force on propeller when effects of wing and body interferences are included

$z_p \bar{c}$ distance of the point (h_0, k) above the propeller thrust line in terms of $\bar{c}$

$x_p \bar{c}$ distance of the same point behind the propeller centre, measuring parallel to the thrust line

$$\gamma = \frac{2D^2}{S} \times z_p \text{ (thrust moment)} + \frac{8}{\pi} \times \frac{S_s c_s}{S\bar{c}} C_{m_0 s} \text{ (slipstream effect)}$$

C_{m0s} C_{m0} of part of wing in slipstream, when there is no slipstream

S_s gross wing area in slipstream

c_s mean chord of part of wing in slipstream

$$\delta = \frac{2D^2}{S} \times x_p \times \varkappa$$

a, a_1, a_2 lift coefficient derivatives of aeroplane less tail, tail elevators respectively—without propellers in all cases

$R_w a,\ R_T a_1,\ R_T a_2$ derivatives corresponding to a, a_1, a_2 but with propellers, for the steady flight condition in which T_c varies with α. We have assumed that the ratio of the lift derivatives of tail and elevators is the same with and without propellers

$R = R_T/R_w$

S gross wing area

ε mean downwash over the tailplane

$\bar{V}$ tail volume coefficient, using tail arm to (h_0, k)

θ propeller thrust-line incidence, in radians

Δ change due to installation of propellers

μ_1 relative density coefficient $\dfrac{w}{g\rho \times \text{tail arm}}$

Primes denote total differentiation with respect to C_L for any given flight condition: thus $\left(\frac{R_T}{C_L}\right)' = \frac{d}{dC_L}\left(\frac{R_T}{C_L}\right)$, and so on. The word 'trim' implies $C_m = 0$.

3. *Algebraic Treatment of Static Stability with Propellers*.—The notation above and the theory which follows consider specifically the case of the single-engined aeroplane, but the method is evidently also applicable to multi-engined types.

We write for the pitching-moment coefficient at zero elevator tab angle

$$C_m = C_{m0} + (h - h_0)C_L + k(C_{D0} - C_L^2/6) + \gamma T_c + \delta N_c$$
$$- R_T \bar{V}\{a_1(\alpha + \eta_T - \varepsilon) + a_2 \eta\} \quad \ldots \quad (1)$$

The third term is an approximation, the correct expression for elliptic lift distribution being $k\left\{C_{D0} - \frac{C_L^2}{a_0} + (C_L \times \text{no-lift angle of aeroplane less tail})\right\}$.

From the definition of C_{mw} and C_{mw0} we also have

$$C_{mw} = C_{m0} + (h - h_0)C_L + k(C_{D0} - C_L^2/6) + \gamma T_c + \delta N_c$$
$$C_{mw0} = C_{m0} + k(C_{D0} - C_L^2/6) + \gamma T_c + \delta N_c \, .$$

Differentiating (1) with respect to C_L at constant η and imposing the trim condition $C_m = 0$ after differentiation gives

$$\left(\frac{dC_m}{dC_L}\right)_{\text{trim}} = \left(\frac{d}{dC_L} - \frac{1}{R_T}\frac{dR_T}{dC_L}\right)\left\{C_{m0} + (h - h_0)C_L + k\left(C_{D0} - \frac{C_L^2}{6}\right)\right.$$
$$\left. + \gamma T_c + \delta N_c\right\} - \frac{\bar{V}a_1}{a}\frac{R_T}{R_w}\left(1 - \frac{d\varepsilon}{d\alpha}\right) \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (2)$$

So far we have followed the procedure of Bryant's R. & M. 2310[1]. If we now write $\left(\frac{d}{dC_L} - \frac{1}{R_T}\frac{dR_T}{dC_L}\right)f = R_T \frac{d}{dC_L}\left(\frac{f}{R_T}\right) = R_T\left(\frac{f}{R_T}\right)'$, where f is any function of C_L, equation (2) takes the form

$$\frac{1}{R_T}\left(\frac{dC_m}{dC_L}\right)_{\text{trim}} = (C_{m0} + k\,C_{D0})\left(\frac{1}{R_T}\right)' + \left\{(h - h_0) \times \frac{C_L}{R_T}\right\}' - \frac{k}{6}\left(\frac{C_L^2}{R_T}\right)'$$
$$+ \gamma\left(\frac{T_c}{R_T}\right)' + \delta\left(\frac{N_c}{R_T}\right)' - \frac{1}{R_w} \times \bar{V}\frac{a_1}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right) \, . \quad \ldots \quad \ldots \quad (3)$$

This result could have been obtained more simply by dividing equation (1) by R_T. Then differentiate and impose the trim condition $C_m = 0$. It should be clear that the differentations indicated by the dashes are complete, not partial. From a graphical standpoint if T_c/R_T, say, is plotted against C_L for a particular flight condition then $(T_c/R_T)'$ is the slope of the resulting curve.

Now the longitudinal c.g. position h is certainly independent of C_L or T_c: if C_{m0} and h_0 are also*, then the second term of (3) becomes $(h - h_0)(C_L/R_T)$.

Noting that by definition of h_n, $(dC_m/dC_L)_{\text{trim}} = 0$ when $h = h_n$, we obtain

$$(h_0 - h_n)\left(\frac{C_L}{R_T}\right)' = (C_{m0} + kC_{D0})\left(\frac{1}{R_T}\right)' - \frac{k}{6}\left(\frac{C_L^2}{R_T}\right)' + \gamma\left(\frac{T_c}{R_T}\right)'$$
$$+ \delta\left(\frac{N_c}{R_T}\right)' - \frac{1}{R_w}\bar{V}\frac{a_1}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right)$$
$$= \left(\frac{C_{mw0}}{R_T}\right)' - \frac{1}{R_w} \cdot \bar{V}\frac{a_1}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right) \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (4a)$$

and

$$\frac{1}{R_T}\left(\frac{dC_m}{dC_L}\right)_{\text{trim}} = (h - h_n)\left(\frac{C_L}{R_T}\right)' = \bar{V}a_2\frac{d\eta_{\text{trim}}}{dC_L}$$
$$= (C_{m0} + k\,C_{D0})\left(\frac{1}{R_T}\right)' + (h - h_0)\left(\frac{C_L}{R_T}\right)' - \frac{k}{6}\left(\frac{C_L^2}{R_T}\right)' + \text{etc.}$$
$$= \left(\frac{C_{mw}}{R_T}\right)' - \frac{1}{R_w}\bar{V}\frac{a_1}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right) \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (4b)$$

* The effect of slipstream on wing C_m is best included in the term γT_c (*see* section 4.2).

We may also show that

$$\left(\frac{dC_m}{dC_L}\right)_{\text{trim}} \text{due to tail} = -R_T'\frac{C_{mw}}{R_T} - R\bar{V}\frac{a_1}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right). \quad \ldots \quad \ldots \quad \ldots \quad (5)$$

Evidently the contribution $R_T\left(\frac{C_{mw}}{R_T}\right)'$ to $\left(\frac{dC_m}{dC_L}\right)_{\text{trim}}$ given by equation (4b) is made up of

(a) C_{mw}', the contribution of aeroplane less tail, which is quite independent of R_T, and

(b) $-R_T'\frac{C_{mw}}{R_T}$ due to the tail, arising from the variation of R_T with C_L.

In full-throttle flight the term (b) is small at high speed and may have either sign, but for cruise or climb C_{mw} is usually positive and R_T' of order 0·3, so that the term is stabilizing. Since the factor R by which the last term of equation (5) is multiplied also increases the stability,* it is evidently true to say that for cruising or climbing flight the total effect of the slipstream factors R_w and R_T is stabilizing. This is illustrated by the worked example of section 5.

We will now show how C_L, R_w, R_T, T_c, N_c and such derivatives as $\left(\frac{1}{R_T}\right)'$, $\left(\frac{C_L}{R_T}\right)'$, etc., are evaluated :—

(i) The first step is to find T_c as a function of C_L† for the power condition for which the stability is required and the appropriate aeroplane weight, etc. In doing this we may often use generalized propeller charts such as those of N.A.C.A. Reports 640[8], 658[9], but considerable errors may result if the propeller blade plan-form is unconventional, as for example in the case of blades obtained from conventional ones of larger diameter by cutting off the tips.

(ii) We then establish C_L as a function of α for this same power condition. It seems sufficient for this purpose simply to increase the lift coefficient without propellers by amount $\frac{2D^2}{S}(T_c \sin\theta + N_c \cos\theta) \doteqdot \frac{2D^2}{S}\theta_{\text{radn}}\left(T_c + \frac{dN_c}{d\theta}\right)$‡ for single-engined aeroplanes : even with two or more engines this approximation may be sufficient. This gives $R_w a$ and hence R_w; the matter is discussed in more detail in section 4.1 below.

(iii) R_T is in general a function of α, C_L, T_c, and geometrical parameters such as tail height, etc. This function has not yet been evaluated though Bryant (R. & M. 2310[1]), Falkner[3] and others have investigated the effects of several of the variables, more particularly for twin-engined types.

For a particular aeroplane with given engine boost and r.p.m., flying at a given height, R_T may be expressed, in theory at least, as a function of either α, C_L or T_c. Since the theoretical R_T vs. T_c relation for a tailplane completely immersed in a slipstream of uniform velocity is $R_T = 1 + \frac{8}{\pi}T_c$ it is evidently convenient to use T_c as our variable. We shall obtain later (section 4.3) an empirical relation between R_T and T_c for single-engined types.

(iv) $\varkappa N_c$ is calculated as $\varkappa\frac{dN_c}{d\theta} \times \theta$ where, it is suggested, $\frac{dN_c}{d\theta}$ should be given roughly its high-speed value (say for $J = 3{\cdot}0$, $T_c = 0$) as calculated by Rumph's method[4] and $\varkappa$ the value 1·3; the justification for all this is given in section 4.2 below. Since θ differs from α merely by a known constant we thus get $\varkappa N_c$ in terms of α.

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* Except at high speeds, $R_T > R_w$, *i.e.*, $R > 1$.

† For conventional designs, in the range from dive to climb, we may ignore the small difference between C_L (*i.e.*, lift coefficient for aeroplane less tail) and trimmed lift coefficient.

‡ *i.e.*, the lift coefficient corresponding to the direct forces on the propeller if acting alone.

As soon as corresponding values of α, C_L, T_c, N_c, R_T have been obtained for the given flight condition we may plot the terms $(C_{m0} + kC_{D0})(1/R_T)$, etc., against C_L and get the separate stability contribution of each term (except the last) on the right-hand side of equation (3) as the slope of the appropriate curve. The overall effect of these terms will best be got by plotting the sum C_{mw}/R_T against C_L. The evaluation of $1 - d\varepsilon/d\alpha$ in the last term is made by the method of section 4.4. below.

Values of $\left(\frac{dC_m}{dC_L}\right)_{\text{trim}}$, $h - h_n$, $\frac{d\eta_{\text{trim}}}{dC_L}$ may now be obtained by use of equations (4b).

4. *Analysis of Model Data on Single-engined Aeroplanes.*—The available data have been analysed to give the effect of propellers on

(a) slope of the lift curve for aeroplane less tail,

(b) pitching moment for aeroplane less tail,

(c) slope of tail lift-curve,

(d) downwash derivative at the tail.

Figs. 1 to 3 are small general arrangement drawings of the three fighter models only, and Fig. 4 shows the T_c vs. C_L relations used in reducing the model results to the flight so-called 'full-throttle' condition.

4.1. *Slope of Lift Curve for Aeroplane less Tail.*—Fig. 5 shows C_L against α from tests on five models, in each case

(a) including the contribution of the forces on the propeller,

(b) subtracting this contribution*, using the *basic* N_c,

(c) as measured without propeller.

For the three fighter designs it appears that there is little systematic difference of lift-curve slope between (b) and (c): this is not confirmed by the curves for the experimental types Supermarine S24/37 and Folland E28/40 where the slope for condition (b) is some 5 to 10 per cent greater than for (c).

Although the reason for this difference of results between the experimental and fighter designs is not clear we shall give more weight to the latter, as we are primarily concerned with fighters, and take it that the lift coefficient with propeller is to be got from that without propeller by adding $2D^2/S \times \theta \times (T_c +$ basic $dN_c/d\theta)$. This implies that the mutual interference on lift between propeller and wing plus body does not vary with incidence: *see* the comments at the foot of Fig. 1.

Note that Ref. 5 (Smelt and Davies) gives for the ΔC_L due to slipstream effect on the wing the expression

$$(\Delta C_L)\ \text{Slipstream} = \frac{\text{area of wing in slipstream}}{\text{total wing area}} \times s \times \{\lambda C_{L0} - 0{\cdot}6\, a_0 \psi\}\,. \quad \ldots \quad (6)$$

Where $1 + s$ is velocity factor at wing centre of pressure in slipstream

C_{L0} lift coefficient of part of wing in slipstream, when there is no slipstream

a_0 two-dimensional lift-curve slope

ψ angle of downwash of slipstream at the wing centre of pressure:

λ is a factor which is about unity for modern aeroplanes, whether single or multi-engined.

* No such subtraction was necessary for the S24/37 and E28/40 designs which were tested with propeller supported free of the model. So for these two models we have curves (b) and (c) only.

If we take the propeller to be 0·7 diameters ahead of the local wing centre of pressure we get $s \simeq 1{\cdot}8b$, where $(1 + 2b)^2 = 1 + \frac{8}{\pi} T_c$. For $T_c = 0{\cdot}1$ (a typical value for climb) this gives $s = 0{\cdot}10$. Taking also $\frac{dC_{L0}}{d\alpha} \simeq \frac{dC_L}{d\alpha}$ we get

$$\frac{d}{d\alpha}(\Delta C_L)_{\text{slipstream}} = \frac{\text{wing area in slipstream}}{\text{total wing area}} \times 0{\cdot}1 \left\{1 - 0{\cdot}6 \frac{a_0}{a} \frac{d\psi}{d\alpha}\right\} a \, .$$

The term in curly brackets is of order 0·8 and so the fractional increase in lift-curve slope due to slipstream is, very roughly,

$$0{\cdot}08 \times \frac{\text{wing area in slipstream}}{\text{total wing area}} \, .$$

We can now see that, for single-engined designs at least, where the fraction of wing area in the slipstream will be only of order 0·3, Ref. 5 would only predict some 2 to 3 per cent increase in lift-curve slope at the C_L corresponding to $T_c = 0{\cdot}1$.

Ignoring this increase therefore seems to be justified for single-engined aeroplanes.

The increase in lift-curve slope due to the direct forces on the propeller is of order 10 per cent for the three fighters of Fig. 5 and appears to be almost constant over a C_L-range including high speed and climb.

4.2. *Pitching Moment due to Propeller Normal Force plus Slipstream Effect on Aeroplane less Tail.*—These effects have been investigated for $T_c \simeq 0$[6, 7]. The main conclusions are:—

(a) The normal force on propeller alone is predictable with quite good accuracy by Rumph's method.[4]

(b) For the combination of propeller with aeroplane less tail on single-engined fighter designs the effective value of rate of change with incidence of propeller normal force (*i.e.*, that which would give the total observed change in dC_m/dC_L) is of order 30 per cent more than for the propeller alone ($\varkappa \simeq 1{\cdot}3$ for $T_c \simeq 0$). The conception of attributing all this change in dC_m/dC_L to the propeller (with a factor to allow for effect of the wing on the propeller normal force, commonly supposed to arise from the upwash at the propeller caused by the wing), is, of course, not strictly correct. The effect of the propeller slipstream on the wing plus body must also be considered, in general. However, the use of the factor $\varkappa$ may be justified as an empiricism, bearing in mind that it includes the slipstream effect.

We will now discuss the effects corresponding to (a) and (b) above but at full throttle instead of $T_c \simeq 0$.

(i) Normal forces have been measured at full throttle in a good many cases and generally speaking they are somewhat greater than at $T_c = 0$. Figs. 6a, 6b, 6c, illustrate this; all these curves have been drawn for blade angle of 50 deg at $0{\cdot}7R$.

(ii) Figs. 7a, 7b, 7c, show the increase in C_m due to propeller (excluding effect of thrust moment) against incidence for the aeroplane less tail, again for $T_c = 0$ and full throttle. The propellers of these models are those to which Fig. 6 applies.

The increase of stability for the full throttle condition over that for $T_c = 0$ is shown clearly in Fig. 7. The difference increases from roughly zero at $T_c = 0$ to quite considerable values in the climb region, though it may change sign quickly at still higher thrusts.

Estimates of the change in C_m of aeroplane less tail in passing from $T_c = 0$ to full throttle have been made on the basis of the formula $(\Delta C_m)_{\text{full throttle}} - (\Delta C_m)_{T_c = 0} \simeq + \frac{8}{\pi} \times \frac{S^s c_s}{S\bar{c}} \times C_{m0s} \times T_c$ (*see* section 2 for definitions of symbols S_s, c_s, C_{m0s}), values of C_{m0s} being calculated from thin aerofoil theory, and the effect of thrust moment being ignored for the moment.

Starting with C_m, α curves for $T_c = 0$ the corresponding curves were estimated for full throttle and are shown in Figs. 7. It will be seen that, up to climb T_c's at least, the estimated full throttle curves show some measure of agreement with experiment, and we suggest that the above method of allowing for stabilizing effect of the slipstream on the wing plus body be used until an improved one can be developed. Note that we have made no attempt to allow for any change in propeller normal force from $T_c = 0$ to full throttle; the method is in fact semi-empirical.

The allowance for slipstream effect is equivalent to changing the thrust line height from its true value of z_p (distance *below* (h_0, k)) to an effective value $z_p + \frac{8}{\pi} \cdot \frac{S_s c_s}{2D^2\bar{c}} \cdot C_{m0s}$ (*see* the expression for γ in section 2 above).

4.3. *Slope of Tail Lift-Curve.*—In order to reduce scatter due to experimental errors we have investigated the tail lift-curve slope relative to its value at $T_c = 0$, not relative to the value without propellers. A separate investigation has shown that a propeller running at $T_c = 0$ has no systematic effect on a_1.

Fig. 8 shows the results obtained for four fighter designs, R_T being determined as the ratio of $dC_m/d\eta$ for full throttle and $T_c = 0$ at the same wing incidence. The scatter when R_t is plotted against T_c is considerable, but some of it is probably due to errors in the model results. We should expect R_T to vary with the parameter propeller diameter $\div$ tail span: values of this ratio for the four models are :—

Typhoon	*Tempest II*	F1/43 (5-blader)	F1/43 (contra-propeller)
1·072	0·88	1·063	1·046

As a tentative value of R_T to be used for estimates we suggest $1 + 1{\cdot}5T_c$.

4.4. *Downwash Derivative at the Tail.*—No suitable complete model tests at full throttle have been made for more than one tail setting and so to get downwash we are forced to various subterfuges.

(a) By what we shall call the direct method we can find downwash values by assuming that the ratio $dC_m/d\eta \div dC_m/d\eta_T$, which can usually be determined without propellers, is the same with propellers.

(b) An indirect method whereby we can get $1 - d\varepsilon/d\alpha$ is suggested by equation (5) of section 3 above.

For the *Typhoon* and F1/43 (5-blader) both methods may be applied but for the *Tempest II* $dC_m/d\eta \div dC_m/d\eta_T$ cannot be found and so only the indirect method is available ; even then we only get $R\bar{V}\frac{a_1}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right)$, from which $1 - d\varepsilon/d\alpha$ cannot be found, a_1 being unknown. However we can find the ratio of $1 - d\varepsilon/d\alpha$ for the full throttle and $T_c = 0$ conditions, and this is plotted against T_c in Fig. 9 for all cases. For the *Typhoon* and F1/43 there are differences in the value obtained by the direct and indirect methods, due partly to the difficulty of correctly reading C_m vs. C_L and ε vs. α slopes when curvatures are large, as they often are at full throttle.

The turning-up of the curves at values of T_c of order 0·1 may be associated with the much earlier stall of the model propeller (blade angle 50 deg) than would occur in flight (constant r.p.m.) : we suggest the tentative relation

$$\frac{(1 - d\varepsilon/d\alpha)_{\text{full throttle}}}{(1 - d\varepsilon/d\alpha)_{T_c = 0}} = 1 - 6{\cdot}2T_c$$

This should hold up to values of T_c of at least $0 \cdot 1$ and may therefore be applied to the climb condition.

In Ref. 2 the approximate relation

$$\frac{(1 - d\varepsilon/d\alpha)_{Tc=0}}{(1 - d\varepsilon/d\alpha)_{\text{no propeller}}} = 1 - 1 \cdot 4 \frac{dN_c}{d\theta}$$

was given.

Hence

$$\frac{(1 - d\varepsilon/d\alpha)_{\text{full throttle}}}{(1 - d\varepsilon/d\alpha)_{\text{no propeller}}} \simeq \left(1 - 1 \cdot 4 \frac{dN_c}{d\theta}\right)(1 - 6 \cdot 2T_c),$$

where $dN_c/d\theta$ is the value for $T_c = 0$ and $J \simeq 3$.

It must be emphasised that this relation has been deduced from tests on only a few single-engined, rather similar, fighter designs and must therefore be used with caution. In particular it should not be applied with values of T_c much greater than $0 \cdot 1$.

The form of the relation is very convenient, for in conjunction with the formula $R_T \simeq 1 + 1 \cdot 5T_c$ (section 4.3) and the method given in section 4.1 for estimating R_w we can very quickly find

$$\left\{R \bar{V} \frac{a_1}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right)\right\}_{\text{full throttle}} \text{ from the value of } \left\{\bar{V} \frac{a_1}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right)\right\}_{\text{no propeller}}.$$

5. *Illustrative Example.*—Consider a hypothetical single-engined fighter with the following geometrical and aerodynamic characteristics:—

$$C_{m0} = -0 \cdot 02,\ h = 0 \cdot 25,\ h_0 = 0 \cdot 20,\ k = -0 \cdot 1,\ C_{D0} = 0 \cdot 015,\ \frac{2D^2}{S} = 1 \cdot 2,$$

$$\text{effective } z_p \left(i.e., \text{ actual } z_p + \frac{8}{\pi} \frac{S_s c_s}{2D^2 \bar{c}} \times C_{m0s}\right) = -0 \cdot 1,\ x_p = 1 \cdot 3,$$

$$dN_c/d\theta = 0 \cdot 2,\ \varkappa = 1 \cdot 3,\ a = 4 \cdot 0,\ a_1 = 3 \cdot 0,\ a_2 = 2 \cdot 0,\ \bar{V} = 0 \cdot 5,\ \theta = \alpha - 2 \text{ deg}.$$

No-lift angle of aeroplane less tail less propeller = 2 deg, $d\varepsilon/d\alpha = 0 \cdot 4$ and so tailplane contribution to $-dC_m/dC_L = 0 \cdot 225$, without propeller. Also, from the above values of z_p and x_p, $\gamma = -0 \cdot 12$, $\delta = +2 \cdot 03$.

Fig. 10a shows T_c vs α and C_L vs α curves, the latter with and without propeller: the estimated lift curve with propeller is nearly linear in the range under consideration ($\alpha = -2$ deg to $+8$ deg).

Fig. 10b gives plots against C_L of the terms $(C_{m0} + kC_{D0})(1/R_T)$, etc., and of their sum C_{mw}/R_T.

Fig. 10c shows R_T plotted against C_L.

Table 1 shows the paper-work necessary to find $-dC_m/dC_L$, $h - h_n$, $d\eta/dC_L$ and also $-(dC_m/dC_L)_{\text{tail}}$, $-(dC_m/dC_L)_{\text{due to propeller}}$: the latter has been split up into three parts :—

(a) Due to direct forces on the propeller: $-(\Delta C_{mw})'$

(b) Due to R not being unity: $(R - 1)\ \bar{V} \frac{a_1}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right)_{\text{full throttle}} + R_T' \cdot \frac{C_{mw}}{R_T}$

(c) Due to downwash change, if $R = 1$: $\bar{V} \frac{a_1}{a} \Delta\left(1 - \frac{d\varepsilon}{d\alpha}\right)$.

Table 1 gives a general idea of how the calculations may be made and of the amount of work involved: Table 2 summarises the more important (asterisked) columns of Table 1. Note that variation of R (from the value unity) produces stability changes which, although appreciable, are quite small compared with the effects of direct forces on and downwash due to the propeller.

6. *Conclusions.*—(i) The method given of stability analysis for single or multi-engined aeroplanes should prove simpler than earlier methods.

(ii) From analysis of model tests on single-engined fighters tentative empirical factors have been obtained for estimating the full throttle stability in terms of that without propeller:—

(a) To estimate values of C_L (for aeroplane less tail) it seems sufficient merely to add the appropriate components of direct propeller forces, calculated as if the propeller were acting alone, to the C_L without propeller.

(b) The model results indicate that without tail and excluding the effect of thrust moment the stability is better at full throttle than for $T_c = 0$ over an incidence range including cruise and climb. This favourable effect of full throttle is attributed mainly to change of wing C_{m0} due to velocity increase in the slipstream and may be as much as $0 \cdot 04\bar{c}$ in the neutral point position in the climb region. It may be estimated very roughly by the method given in section 4.2, which expresses the effect as an equivalent change of thrust-line height: *viz.*,

$$\text{Effective } z_p - \text{actual } z_p \simeq \frac{8}{\pi} \times \frac{S_s c_s}{2D^2 \bar{c}} \times C_{m0s} \, .$$

(c) For slope of tail lift curve we suggest the factor $1 + 1 \cdot 5T_c$ as giving the effect of the slipstream.

(d) For downwash derivative at the tail the data give

$$\frac{\left(1 - \frac{d\varepsilon}{d\alpha}\right)_{\text{full throttle}}}{\left(1 - \frac{d\varepsilon}{d\alpha}\right)_{T_c = 0}} \simeq 1 - 6 \cdot 2T_c \, .$$

Taken in conjunction with the result of Ref. 2 this gives

$$\frac{\left(1 - \frac{d\varepsilon}{d\alpha}\right)_{\text{full throttle}}}{\left(1 - \frac{d\varepsilon}{d\alpha}\right)_{\text{no propeller}}} \simeq \left(1 - 1 \cdot 4 \frac{dN_c}{d\theta}\right)(1 - 6 \cdot 2T_c) .$$

These formulae should not be used for values of T_c greater than about $0 \cdot 1$.

The effect on stability of downwash change due to the propeller is very much greater than the effect of variation of the velocity factor R from unity.

(iii) The algebra of the Appendix shows that at high speed the difference of effects of propeller on manœuvre and stability margins should be small. At all speeds it will be algebraically greater for large than for small aircraft of the same geometry and the difference will increase with reduction of speed, at constant throttle.

When manœuvre point is required from model tests, these should be made at a number of values of T_c, the same values being taken at all incidences instead of using single T_c-values or non-overlapping T_c-ranges at the different incidences.

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8	E. P. Hartman and D. Biermann ..	The Aerodynamic Characteristics of Full Scale Propellers having 2, 3 and 4 Blades of Clark Y and R.A.F. 6 Aerofoil sections. N.A.C.A. Rpt. 640. 1938.
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APPENDIX

Relative Effect of Propellers on the Manœuvre Margin H_m and on $K_n = -dC_m/dC_L$

We shall only consider this with $(\partial R_T/\partial\alpha)_{T_c \text{ const.}} = 0$.

Under these conditions the manoeuvre margin is defined by

$$\left(\frac{\partial C_m}{\partial C_L}\right)_{V \text{ const.}} - \frac{R_T \bar{V} a_1}{2\mu_1} = h - h_m \;(\text{or} - H_m)\,. \qquad \ldots \quad \ldots \quad \ldots \quad (8)$$

Since the speed V is constant, as well as the engine power, so is T_c and we may replace ' V const.' by ' T_c const.' in the above. Note that J and the propeller blade angle are fixed during the manœuvre; we have assumed that R_T is also fixed. The derivative $\partial C_m/\partial C_L$ is to be taken where $C_m = 0$ and the T_c will be that corresponding to straight flight at the speed in question.

Now from equation (1) of section 3 we get

$$\left(\frac{\partial C_m}{\partial C_L}\right)_{T_c \text{ const.}} = \frac{\partial}{\partial C_L}\left\{C_{m0} + (h - h_0)C_L + k\left(C_{D_0} - \frac{C_L^2}{6}\right) + \gamma T_c + \delta N_c\right\}_{T_c \text{ const.}}$$
$$- \frac{R_T}{R_{w1}}\bar{V}\frac{a_1}{a}\left(1 - \frac{\partial\varepsilon}{\partial\alpha}\right)_{T_c \text{ const.}} \qquad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (9)$$

where R_{w1} is the value of R_w taken at constant T_c.

Before taking partial derivatives C_m, C_{m0}, N_c are to be expressed as functions of C_L and T_c, ε as a function of α and T_c. This being understood, we shall from now on omit the suffix ' T_c const.'

Equation (9) now gives

$$\frac{\partial C_m}{\partial C_L} = h - h_0 - \frac{k}{3} C_L + \delta \frac{\partial N_c}{\partial C_L} - \frac{R_T \bar{V}}{R_{w1}} \frac{a_1}{a} \left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right).$$

The manoeuvre margin H_m is given by

$$H_m = -\frac{\partial C_m}{\partial C_L} + \frac{R_T \bar{V} a_1}{2\mu_1} \text{ (\textit{see} equation (8))}$$

i.e.,

$$H_m + h = h_0 + \frac{k}{3} C_L - \delta \frac{\partial N_c}{\partial C_L} + \frac{R_T \bar{V}}{R_{w1}} \frac{a_1}{a} \left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right) + \frac{R_T \bar{V} a_1}{2\mu_1}.$$

So, using now suffix ' 0 ' to denote absence of propellers,

$$\Delta H_m = -\delta \frac{\partial N_c}{\partial C_L} + \bar{V} \frac{a_1}{a} \left[\frac{R_T}{R_{w1}}\left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right) - \left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right)_0\right] + \frac{\bar{V} a_1}{2\mu_1} (R_T - 1) \ldots \quad (10a)$$

It is more profitable to compare ΔH_m with $\Delta(dC_m/dC_L) = \Delta K_n$ than with ΔH_n:—

$$\Delta K_n = -\gamma \frac{dT_c}{dC_L} - \delta \frac{dN_c}{dC_L} + \frac{\bar{V} a_1}{a} \left[\frac{R_T}{R_w}\left(1 - \frac{d\varepsilon}{d\alpha}\right) - \left(1 - \frac{d\varepsilon}{d\alpha}\right)_0\right] + \frac{R_T'}{R_T} C_{mw} \quad .. \quad (10b)$$

This follows from equation (4b), remembering that $R_T(C_{mw}/R_T)' = C_{mw}' - R_T'/R_T \times C_{mw}$. Of course $(\partial\varepsilon/\partial\alpha)_0 = (d\varepsilon/d\alpha)_0$, both derivatives applying to the ' no propeller ' case. If further we take $\partial N_c/\partial C_L \simeq dN_c/dC_L$, $R_{w1} \simeq R_w$ (there is some justification for these approximations) we get

$$\Delta H_m - \Delta K_n \simeq \gamma \frac{dT_c}{dC_L} - \Delta \frac{a_1}{a} R \left(\frac{d\varepsilon}{d\alpha} - \frac{\partial \varepsilon}{\partial \alpha}\right) + \frac{\bar{V} a_1}{2\mu_1} (R_T - 1) - \frac{R_T'}{R_T} C_{mw}. \quad .. \quad (10c)$$

Of the terms on the right-hand side the first is small at high speed but may be considerable in the climb region and of either sign, according to the sign of γ. The second term is positive and will increase with reduction of speed; unfortunately the single-engined model tests analysed in this report were made with T_c varying with α according to a fixed-throttle steady-flight condition and so give no data on $\partial\varepsilon/\partial\alpha$. This emphasises that to find manoeuvre point from model tests these tests should be made at a number of fixed values of T_c, the same value of T_c being taken at all incidences instead of using single T_c-values or non-overlapping T_c-ranges at the different incidences.

μ_1 may vary from values less than ten for very large aircraft flying at low height to more than a hundred for small fighters at high altitude; in the first case the third term of equation (10c) is of order $0 \cdot 1(R_T - 1)$, in the second case $0 \cdot 01(R_T - 1)$; again the term is very small at high speed, where $R_T \simeq 1$, and increases as the speed falls.

The last term —$(R_T'/R_T)\, C_{mw}$ is usually very small and positive at high speed, negative for climb; for the latter condition it is fairly sensitive to c.g. position, via the term $(h - h_0)C_L$ of C_{mw}. Column (S) of Table 1 gives values of $R_T'/R_T \times C_{mw}$ for the hypothetical aeroplane of section 5, which has $h - h_0 = 0 \cdot 05$.

To sum up, we can say that $\Delta H_m - \Delta K_n$ should be small but not necessarily negligible at high speed. At all speeds it will be algebraically greater for large than for similar small aircraft and the difference will increase with reduction of speed.

TABLE 1

Calculation of Full-Throttle Stability for a Hypothetical Single-engined Fighter

Hence $R_w = 4{\cdot}30 \div 4{\cdot}0 = 1{\cdot}075$ (*see* Fig. 9)

α deg	θ deg	T_c	$T_c + dN_c/d\theta$	TC_L	C_L (No prop.)	Full Throttle C_L	$R_T = 1+1{\cdot}5T_c$	$R = \frac{R_T}{R_w}$	N_c	(A) $\frac{\gamma T_c}{R_T}$	(B) $\frac{\delta N_c}{R_T}$	(C) $\frac{C_{m0}+kC_{D0}}{R_T}$	(D) $\frac{(h-h_0)C_L}{R_T}$	(E) $-\frac{k}{6}\frac{C_L^2}{R_T}$	$1-6{\cdot}2T_c$	(F) $R\,\bar{V}\,\frac{a_1}{a}(1-d\varepsilon/d\alpha)$ Full throttle	(G) (A)+(B)+(C)+(D)+(E)	(H) $\frac{d(G)}{dC_L}$
−2	−4	0	0·200	−0·017	0	−0·017	1·00	0·930	−0·014	0	−0·0283	−0·0215	−0·0008	0	1·00	0·1508	−0·0506	0·141$_5$
0	−2	0·011	0·211	−0·009	0·140	0·131	1·016	0·945	−0·007	−0·0012	−0·0140	−0·0212	0·0064	0·0003	0·932	0·1424	−0·0297	0·138
2	0	0·034	0·234	0	0·279	0·279	1·051	0·978	0	−0·0039	0	−0·0205	0·0132	0·0012	0·790	0·1251	−0·0100	0·130
4	2	0·062	0·262	0·011	0·419	0·430	1·093	1·018	0·007	−0·0068	0·0130	−0·0196	0·0196	0·0028	0·616	0·1018	+0·0090	0·124
6	4	0·093	0·293	0·024	0·558	0·582	1·139	1·059	0·014	−0·0098	0·0250	−0·0189	0·0256	0·0049	0·424	0·0729	0·0268	0·113
8	6	0·125	0·325	0·041	0·698	0·739	1·187	1·103	0·021	−0·0126	0·0359	−0·0181	0·0312	0·0076	0·225	0·0403	0·0440	0·107

α deg	(I) $R_T\times$(H)	(J) (F) − (I)	(K) $\frac{-(J)}{R_T\bar{V}a_2}$	(L) $\left(\frac{C_L}{R_T}\right)$	(M) $R_T\times$(L) $=\phi$	$\frac{(J)}{(M)}$	(N) $-\frac{1}{3}k\times C_L$, No prop.	(O) $(C_m)'$ No tail, No prop.	(P) 0·225 − (O)	(Q) (J) − (P)	(R) R_T'	(S) (R) × (G)	(F) + (S)	(T) $\frac{(1-d\varepsilon/d\alpha)_{f.th}}{(1-d\varepsilon/d\alpha)_{\text{no prop.}}}$	(U) −0·225 × [1 − (T)]	(V) 0·225 × (T)	(W) (R − 1) × (V)	(X) (W) + (S)	(Q) − (U) − (X)
−2	0·1413	0·009	−0·009	1·00	1·00	−0·009	0	0·050	0·175	−0·166	0	0	0·151	0·720	−0·063	0·162	−0·011	−0·011	−0·092
0	0·140	0·002	−0·002	0·93	0·945	−0·002	0·005	0·055	0·170	−0·168	0·178	−0·0053	0·137	0·671	−0·074	0·151	−0·008	−0·013	−0·081
2	0·137	−0·012	+0·011	0·88	0·93	+0·013	0·009	0·059	0·166	−0·178	0·278	−0·0028	0·122	0·569	−0·097	0·128	−0·003	−0·006	−0·075
4	0·1357	−0·034	0·031	0·82	0·90	0·038	0·014	0·064	0·161	−0·195	0·292	+0·0026	0·104$_5$	0·443$_5$	−0·125	0·100	+0·002	+0·005	−0·075
6	0·1291	−0·056	0·049	0·76	0·87	0·064	0·019	0·069	0·156	−0·212	0·300	0·0080	0·081	0·305	−0·156	0·069	0·004	0·012	−0·068
8	0·127	−0·087	0·073	0·72	0·85$_5$	0·102	0·025	0·075	0·150	−0·237	0·316	0·0139	0·054	0·162	−0·189	0·036	0·004	0·018	−0·066
		*	*			*				*			*		*			*	*

TABLE 2

Stability Data Abstracted from Table 1

α deg	T_c	Full throttle C_L	$\left(-\frac{dC_m}{dC_L}\right)$ trim	$\frac{d\eta}{dC_L}$	$h-h_n$	$\left(-\frac{dC_m}{dC_L}\right)$ Trim. Due to tail. With Prop	$\left(-\frac{dC_m}{dC_L}\right)_{trim}$ Due to the Propeller: Direct force on prop.*	Variation of R	Down-wash change	Total
−2	0	−0·017	0·009	−0·009	−0·009	0·151	−0·092	−0·011	−0·063	−0·166
0	0·011	0·131	0·002	−0·002	−0·002	0·137	−0·081	−0·013	−0·074	−0·168
2	0·034	0·279	−0·012	0·011	0·013	0·122	−0·075	−0·006	−0·097	−0·178
4	0·062	0·430	−0·034	0·031	0·038	0·1045	−0·075	+0·005	−0·125	−0·195
6	0·093	0·582	−0·056	0·049	0·064	0·081	−0·068	0·012	−0·156	−0·212
8	0·125	0·739	−0·087	0·073	0·102	0·054	−0·066	0·018	−0·189	−0·237

* The favourable slipstream effect on wing + body is included with the thrust moment effect, which along with the effect of normal force on the propeller, makes up the values in this column. The value of $\frac{dC_m}{dC_L}$ or $h - h_n$ without propeller is 0·175.

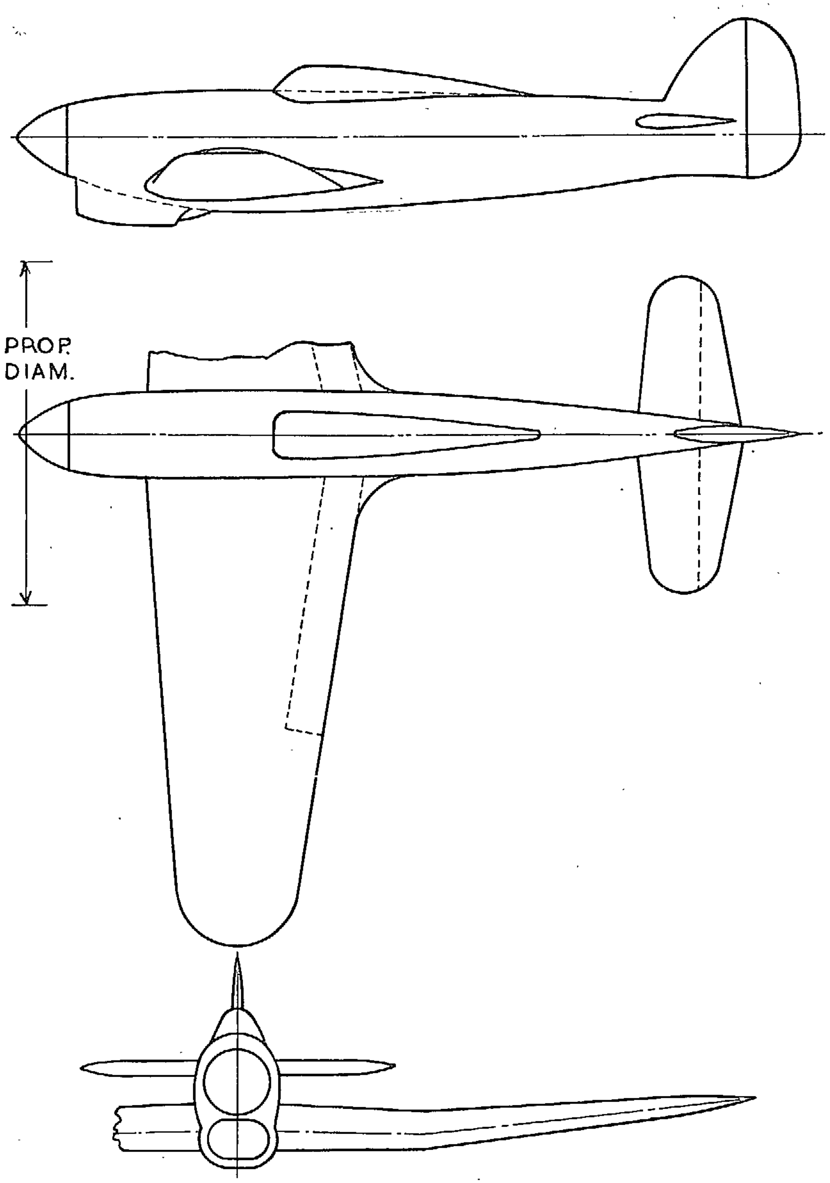


FIG. 1. *Typhoon.*

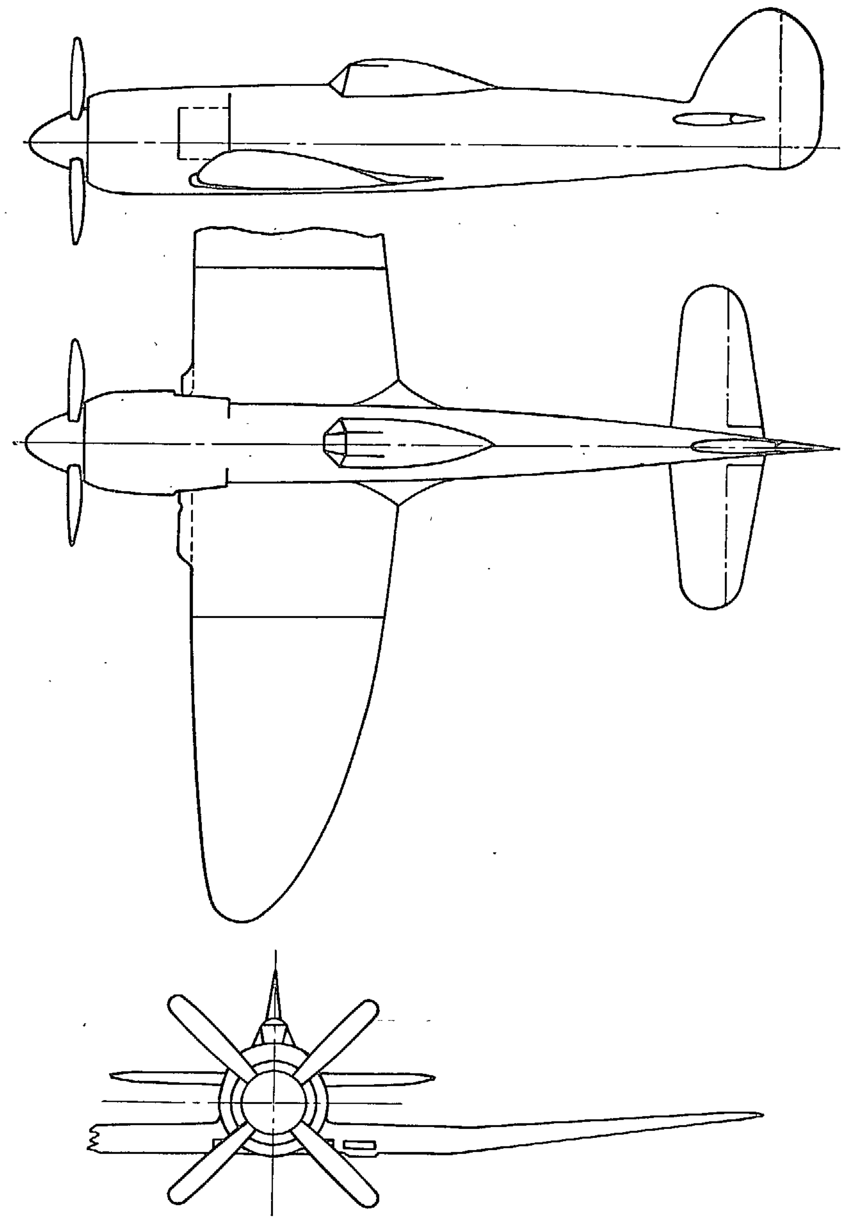
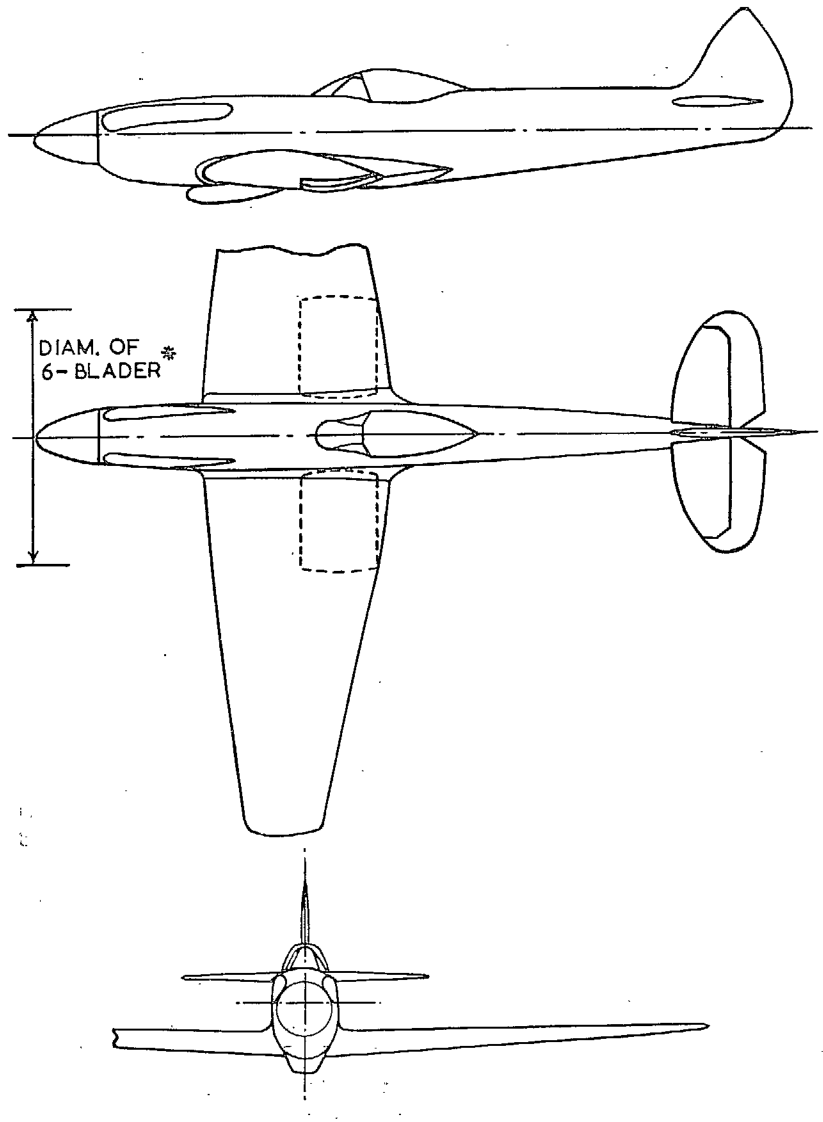


FIG. 2. General arrangement of *Tempest II* model.



* THE 5-BLADED PROPELLER IS OF SLIGHTLY (ABOUT 1·7%) LARGER DIAMETER

FIG. 3. Supermarine F1/43 (*Spiteful*).

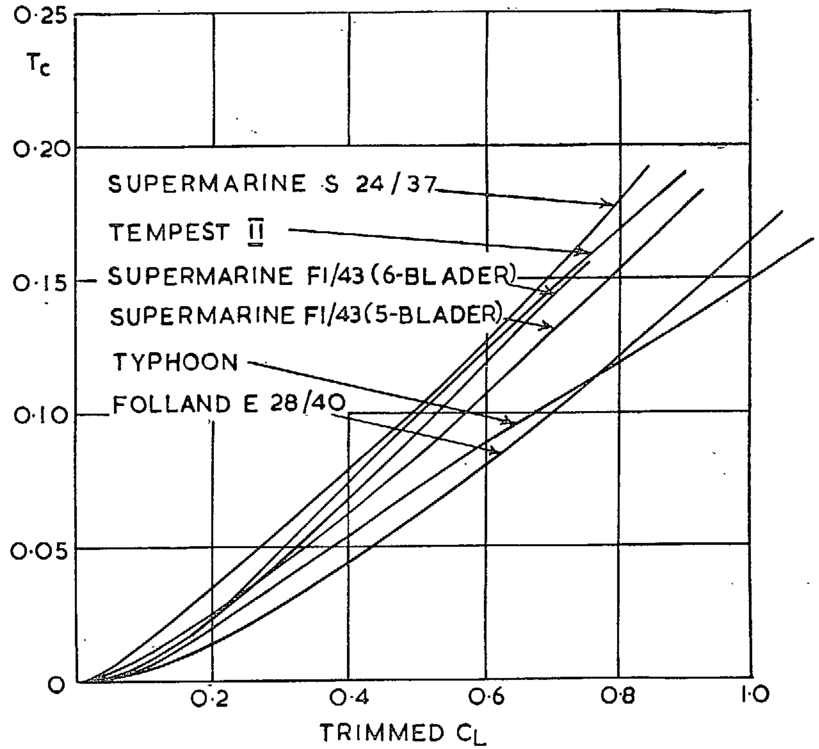


FIG. 4. T_c vs. C_L relations used to apply model results to condition of steady flight at 'full throttle' (*see* Figs. 5 to 8).

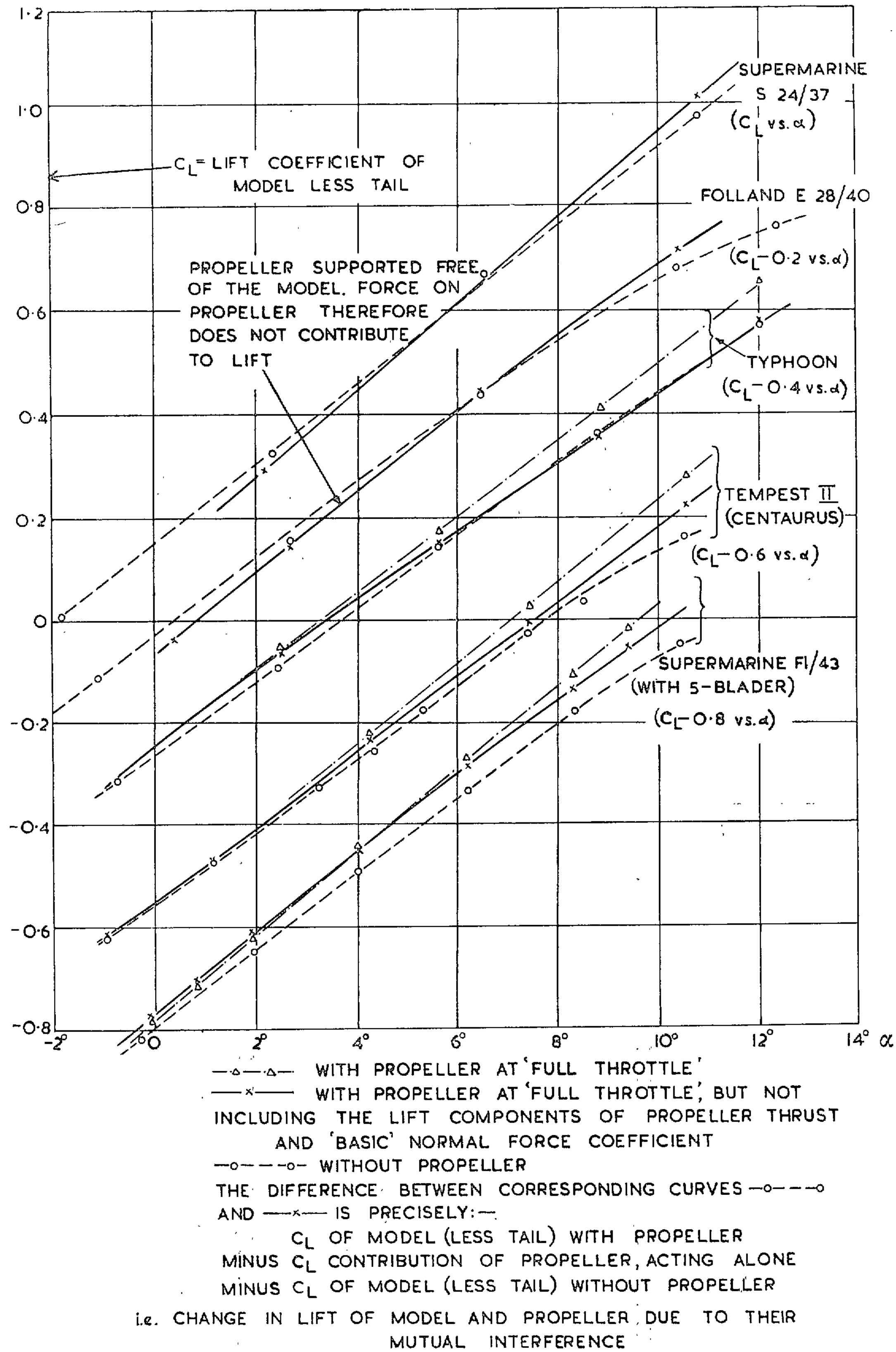


FIG. 5. The mutual lift interference between propeller and aeroplane less tail for five single-engined aircraft at full throttle.

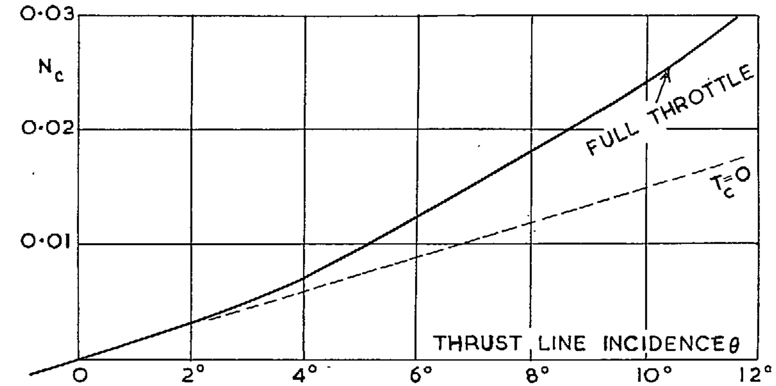


FIG. 6a. N_c against θ for the *Typhoon* propeller.

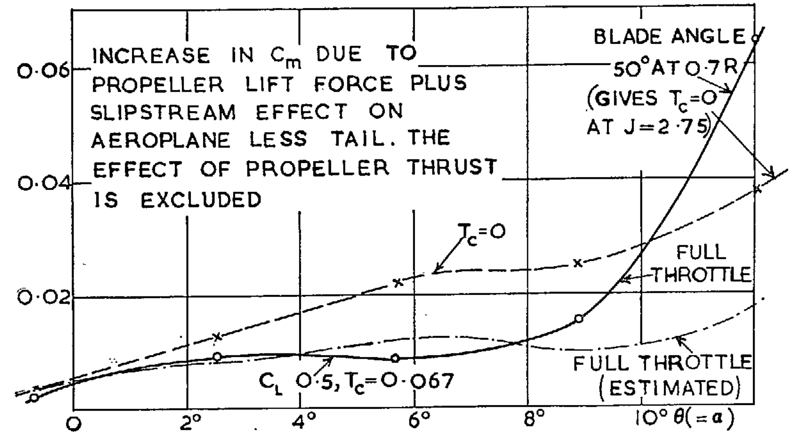


FIG. 7a. *Typhoon* with de Havilland 3-blader.

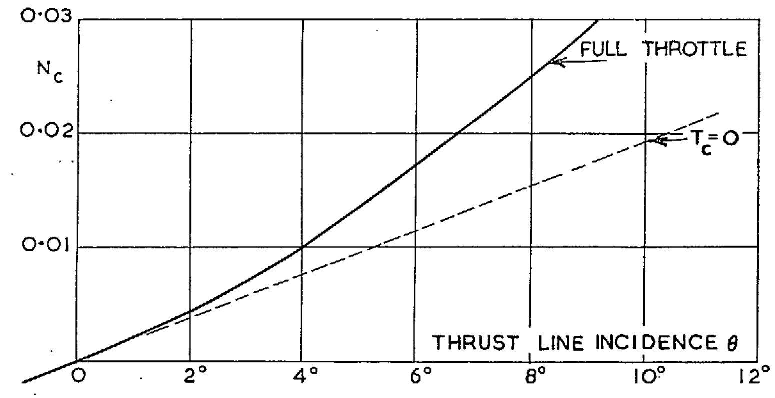


FIG. 6b. N_c against θ for the *Tempest II* propeller.

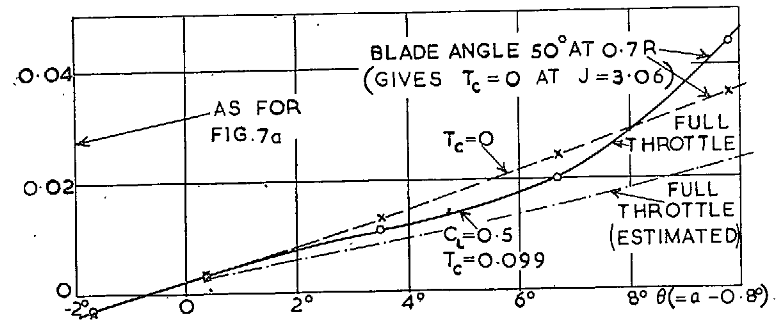


FIG. 7b. *Tempest II* (Centaurus) with 4-blader.

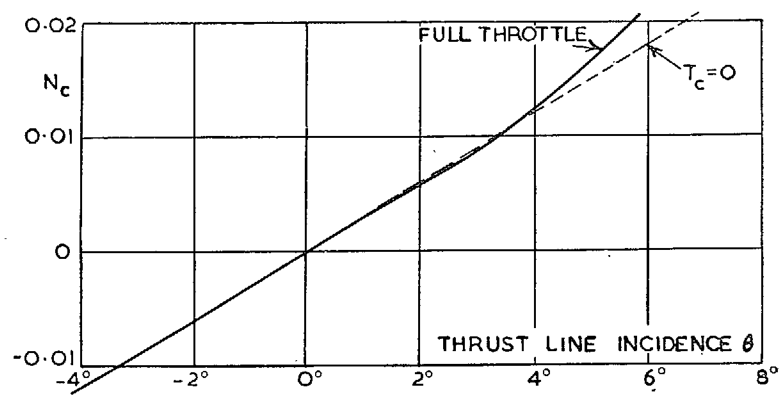


FIG. 6c. N_c against θ for the F1/43 5-bladed propeller.

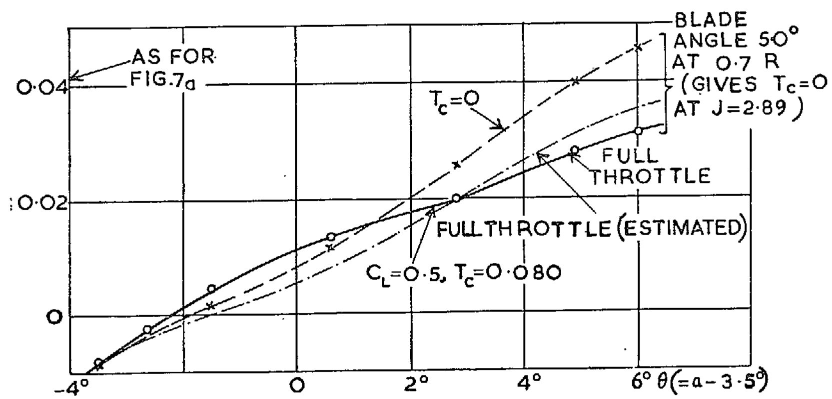


FIG. 7c. Supermarine F1/43 with 5-blader.

Pitching moment due to propeller 'lift' force plus slipstream effect on aeroplane less tail, for three single-engined models.

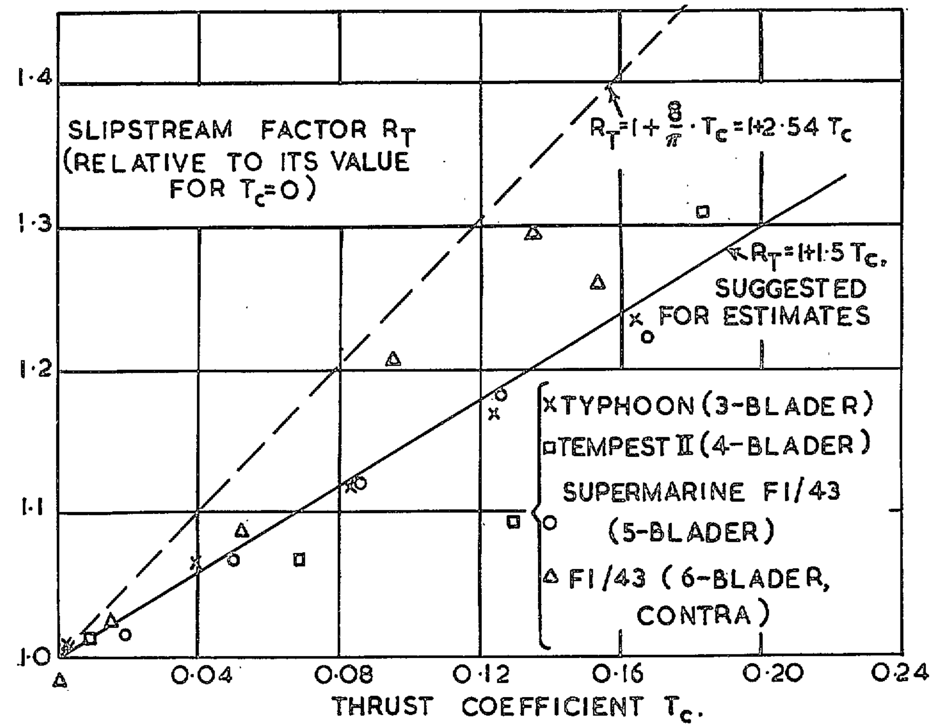


FIG. 8. The slipstream factor R_T for some single-engined fighters. (α varying with T_c according to full-throttle condition.)

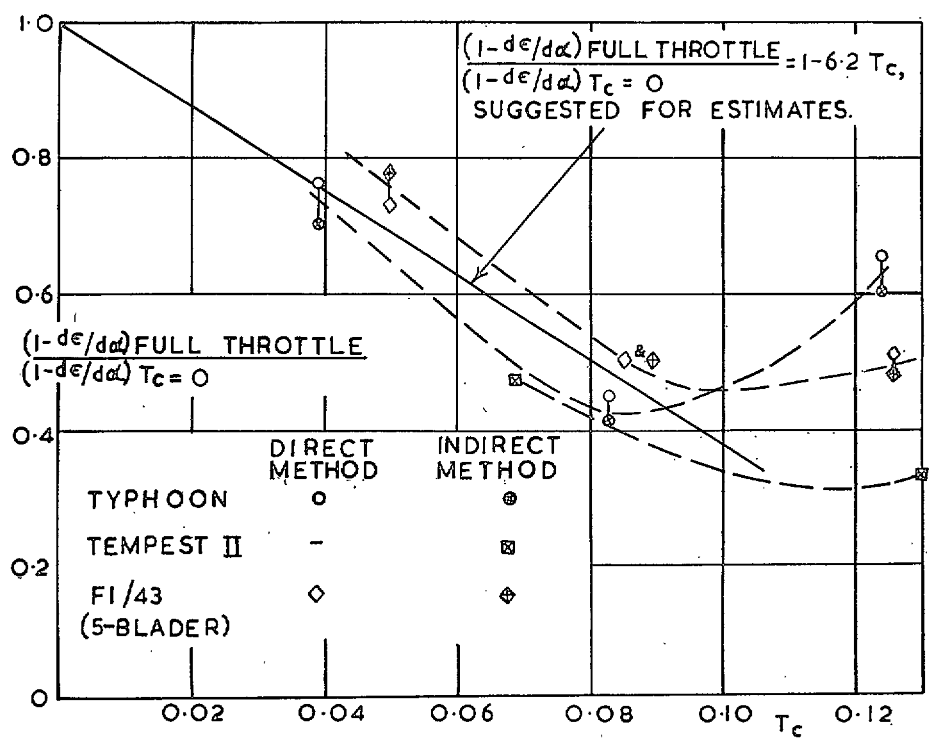
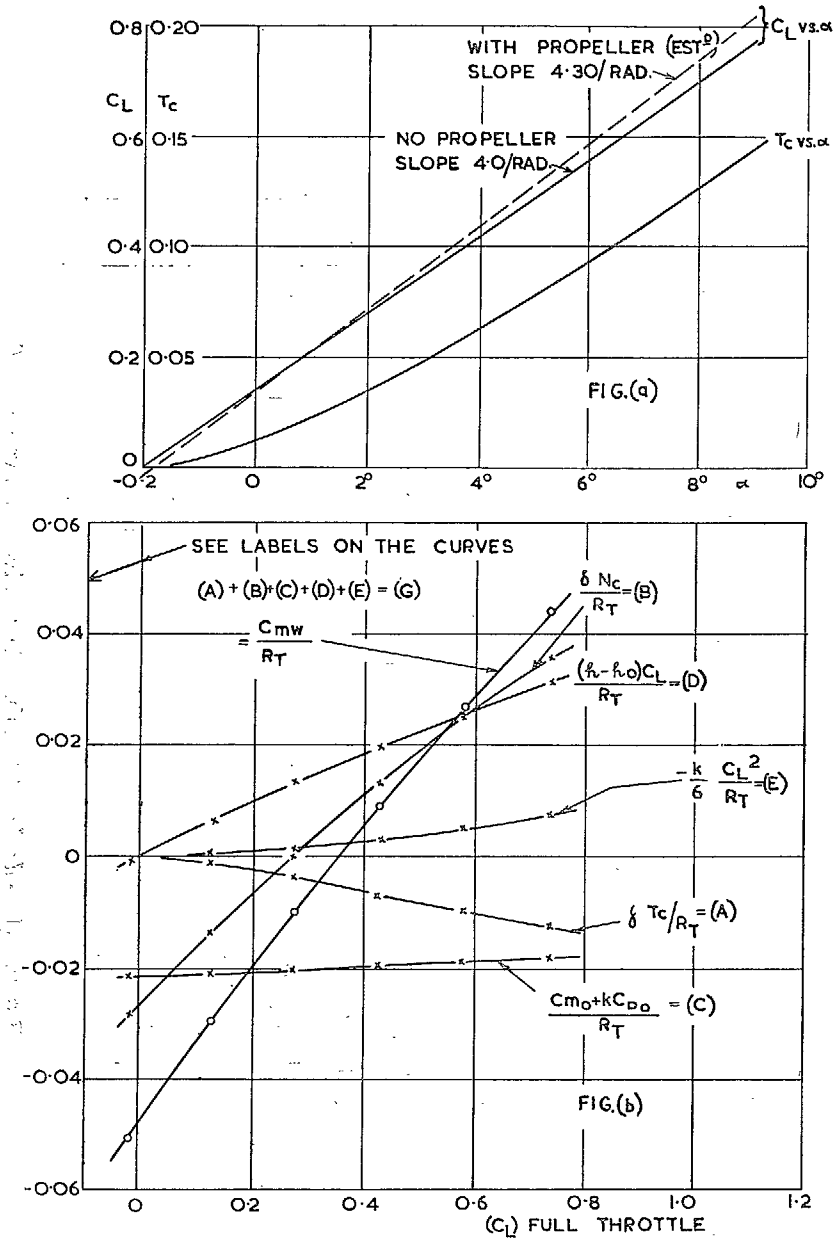


FIG. 9. $(1 - d\varepsilon/d\alpha)$ at full throttle relative to its value for $T_c = 0$.

(99332) Wt. 15/680 K.5 10/53 Hw.



FIGS. 10a and 10b. Hypothetical single-engined fighter. Full-throttle stability without tail.

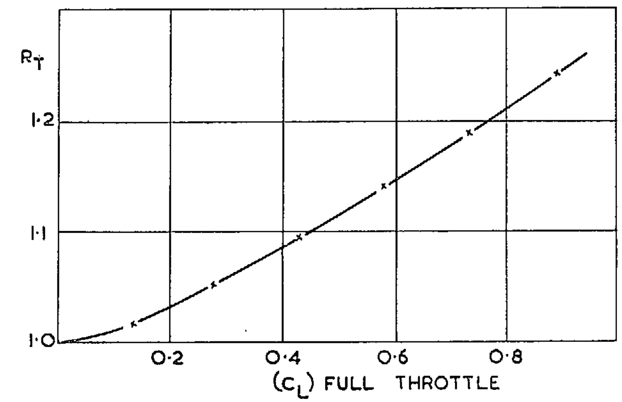


FIG. 10c. R_T against C_L for the hypothetical fighter design.

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