AERODYNAMIC CENTRE OF WING-FUSELAGE COMBINATIONS

1. NOTATION AND UNITS

		SI	*British*
A	aspect ratio of equivalent wing planform, b^2/S		
a	lift-curve slope of equivalent wing planform	radian^{-1}	radian^{-1}
b	span of wing planform	m	ft
$\bar{c}$	standard (or geometric) mean chord of equivalent wing planform	m	ft
$\bar{\bar{c}}$	aerodynamic mean chord of equivalent wing planform	m	ft
c_f	root chord of true wing planform at side of projected fuselage planform (see Sketch 3.1)	m	ft
c_r	root chord of equivalent wing planform	m	ft
c_{ref}	general reference chord for stability calculations	m	ft
c_t	tip chord of equivalent wing planform	m	ft
c_0	centre-line chord of equivalent wing planform	m	ft
d	width of fuselage at leading edge of root chord of equivalent wing planform	m	ft
F	function used in estimating effect of fuselage on aerodynamic centre position (Equation (2.2))		
G	function used in estimating effect of fuselage on aerodynamic centre position (Equation (2.2))		
h	height of fuselage at leading edge of root chord of equivalent wing planform	m	ft
K_1, K_2	functions used in estimating effect of fuselage on aerodynamic centre position (Equation (2.2))		
l	overall fuselage length	m	ft
M	free-stream Mach number		
m	fuselage length forward of leading edge of root chord of equivalent wing planform (see Sketch 3.1)	m	ft
N	number of cranks in leading edge of true semi-wing planform		

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n	fuselage length aft of trailing edge of root chord of equivalent wing planform (see Sketch 3.1)	m	ft
S	area of equivalent wing planform	m^2	ft^2
S_e	area of planform of true wing outside projected fuselage planform	m^2	ft^2
s	semi-span of wing planform, $b/2$	m	ft
$s_{l,0}$	spanwise distance to side of fuselage planform at point where leading edge of root chord of true wing planform meets side of projected fuselage planform (see Sketch 3.1)	m	ft
$s_{l,\,i}$	spanwise distance to crank i in leading edge of planform of true wing for $i = 1, 2, \ldots, N$ (see Sketch 3.1)	m	ft
$\bar{x}$	chordwise location of aerodynamic centre of equivalent wing planform, measured positive aft from leading edge of aerodynamic mean chord of equivalent wing planform	m	ft
$\bar{\bar{x}}$	chordwise location of leading edge of aerodynamic mean chord of equivalent wing planform, measured positive aft from apex of equivalent wing planform	m	ft
x_f	chordwise location of leading edge of c_f measured positive aft from fuselage nose (see Sketch 3.1)	m	ft
x_h	chordwise location of aerodynamic centre of wing-fuselage combination, measured positive aft from leading edge of aerodynamic mean chord of equivalent wing planform	m	ft
Δx_h	forward shift in chordwise location of aerodynamic centre due to presence of fuselage, $\bar{x} - x_h$	m	ft
$x_{h\ ref}$	chordwise location of aerodynamic centre of wing-fuselage combination, measured positive aft from zero-datum axis (see Sketch 4.1)	m	ft
x_{nose}	chordwise location of fuselage nose, measured positive aft from zero-datum axis (see Sketch 4.1)	m	ft
x_{ref}	chordwise location of general stability calculation reference point, measured positive aft from zero-datum axis (see Sketch 4.1)	m	ft
$\Delta x_{l,\,i}$	local increase at notch in chordwise distance of wing leading-edge aft of fuselage nose	m	ft
β	compressibility factor, $(1 - M^2)^{½}$		
Λ_0	leading-edge sweepback angle of equivalent wing planform	degree	degree
$\Lambda_{½}$	half-chord sweepback angle of equivalent wing planform	degree	degree

$\Lambda_{l,\,i}$	leading-edge sweepback angle of true wing planform between spanwise positions $s_{l,\,i-1}$ and $s_{l,\,i}$ for $i = 1, 2, \ldots, N$	degree	degree
$\Lambda_{l,\,N+1}$	leading-edge sweepback angle of true wing planform between spanwise positions $s_{l,\,N}$ and s	degree	degree
λ	taper ratio of equivalent wing planform, c_t/c_0		

Note: Amendment B of this Item in 1996 involved a reassessment of the rate at which the function F, see Section 2, varied with increase of forebody length. The rate apparent in the data of Derivation 2 was found to be too high in the light of additional data acquired since issue in 1976, Derivations 13 and 14, and from a re-analysis of the data in Derivations 5 to 12 made independently of the data of Derivation 2. The overall form of F in Figure 1 is similar to that of the original Figure but the variation with m/c_r is slower above $m/c_r = 1.5$.

2. OUTLINE OF METHOD

The chordwise position of the wing-fuselage aerodynamic centre, x_h, is calculated by finding the chordwise location of the aerodynamic centre, $\bar{x}$, of a straight-tapered wing planform equivalent to the true wing planform, and then incorporating a correction term Δx_h to allow for the effect of the fuselage. The equivalent wing planform is defined in Section 3. Measured positive aft from the leading edge of the aerodynamic mean chord of the equivalent wing planform, and expressed as a fraction of that chord, the position of the aerodynamic centre is given by the equation

$$\frac{x_h}{\bar{\bar{c}}} = \frac{\bar{x}}{\bar{\bar{c}}} - \frac{\Delta x_h}{\bar{\bar{c}}} \ . \tag{2.1}$$

The value of $\bar{x} / \bar{\bar{c}}$ is calculated from Item No. 70011, and the correction term $\Delta x_h / \bar{\bar{c}}$ is given by the equation

$$\frac{\Delta x_h}{\bar{\bar{c}}} = \frac{c_r d^2 F G}{\bar{\bar{c}} \ a \ S}\left[1 + 0.15\left(\frac{h}{d} - 1\right)\right] - (K_1 + \lambda K_2) \ , \tag{2.2}$$

where

F is a function of m/c_r and n/c_r and is plotted in carpet form in Figure 1, see note on page 3,

G is a function of $\beta d/c_r$ and is plotted in Figure 2,

K_1 is a function of d/b, $A \tan \Lambda_{1/2}$ and λ and is plotted in Figure 3 in carpets of $A \tan \Lambda_{1/2}$ and λ for $d/b = 0.08$, 0.12 and 0.16 (see Note below)

and

K_2 is a function of βA and $A \tan \bar{\bar{c}}$ and is plotted in carpet form in Figure 4.

(Note: to determine K_1 at other values of d/b, the values of K_1 at the required $A \tan \Lambda_{1/2}$ and λ should be obtained from each of the carpets, and plotted against d/b. The value of K_1 at the required value of d/b is then read from this cross-plot.)

Equation (2.2) was developed from Derivations 1 to 4. Derivation 2 gives a method, based on a series of wind-tunnel tests, for predicting the effect of adding a fuselage to high aspect ratio , unswept wings. This provided the first term in Equation (2.2) which allows for the influence of fuselage length , width and width to height ratio. The function F allows for the effect of varying the fuselage forebody and afterbody lengths; the function G allows for the effect of varying the fuselage width. Derivation 1 suggests that the effect of wing sweepback on aerodynamic centre position can be treated by allowing for the change in spanwise load-grading over the gross wing caused by adding a fuselage. This approach has been adopted here and Derivation 4, which used the method of Derivation 3 to generate systematic theoretical data for the movement in aerodynamic centre position caused by this change in loading, was employed to establish the second term in Equation (2.2) by empirical means. The function K_1 represents the primary effect of wing sweepback and was constructed for $\beta A = 10$; the term λK_2 represents a correction to K_1 to allow for variations in βA. The geometric parameters needed to evaluate F, G, K_1 and K_2 are defined in Section 3.

Section 4 explains how the aerodynamic centre can be referred to reference positions and chords other than those assumed in Equation (2.1).

3. EQUIVALENT WING PLATFORM*

In calculating x_h for the wing-fuselage combination it is necessary to estimate the aerodynamic centre and lift-curve slope of the gross wing. This was done using Item No. 70011 which presents charts for determining these parameters in terms of aspect ratio, half-chord sweep angle and taper ratio, but only for straight-tapered wing planforms. An equivalent straight-tapered gross wing planform is therefore used to represent the true wing, see Sketch 3.1. When there are no cranks in the true wing planform the equivalent planform results simply from extrapolating to the centre-line the leading and trailing edges of the true wing planform. When there are cranks in the true wing a more complicated construction is needed and the equivalent wing planform is established by defining its tip and root chords in relation to the configuration planform as described here.

The geometric parameters needed to define and position the equivalent wing planform are calculated from the true wing-fuselage planform geometry. Both the fuselage and the wing are projected on to the same horizontal plane so that the vertical position of the wing on the fuselage is immaterial. In this way the intersections of the leading and trailing edges of the wing planform with the sides of the projected fuselage planform define a root chord for the true wing in terms of its length, c_f, and the position of its leading edge from the fuselage nose, x_f, see Sketch 3.1. Usually, the part of the projected fuselage intersected by the true wing planform has parallel sides and c_f is well defined by this procedure. In situations where this part of the fuselage is slightly curved, c_f should be taken as the chordwise distance from the point where the wing leading-edge intersects the fuselage planform to the wing trailing-edge, extended into the fuselage if necessary. This chord then defines a parallel-sided fuselage planform over the wing which is used in the construction of the equivalent gross wing planform.

If the true wing planform has a streamwise tip chord, this is taken , fixed relative to the fuselage planform, as the tip chord, c_t, of the equivalent wing planform. If the true wing has a non-streamwise tip chord or a small curved tip then the linear leading and trailing edges just inboard of the tip, at $0.9s$ say, are extrapolated outboard and their streamwise separation at maximum span is taken as c_t. The span of the equivalent wing planform is thus the span of the true wing planform.

The root chord of the equivalent wing planform is defined on the side of the fuselage planform. Its length, c_r, is determined such that the exposed planforms of the equivalent and true wings (*i.e.* the planforms outside the fuselage planform) have equal areas. Thus

$$c_r = \frac{S_e}{s - s_{l,0}} - c_t \qquad (3.1)$$

where S_e is the area of the exposed panels of the true wing planform. Note that if the true wing does not have a streamwise tip chord, S_e is calculated to the constructed tip chord, c_t.

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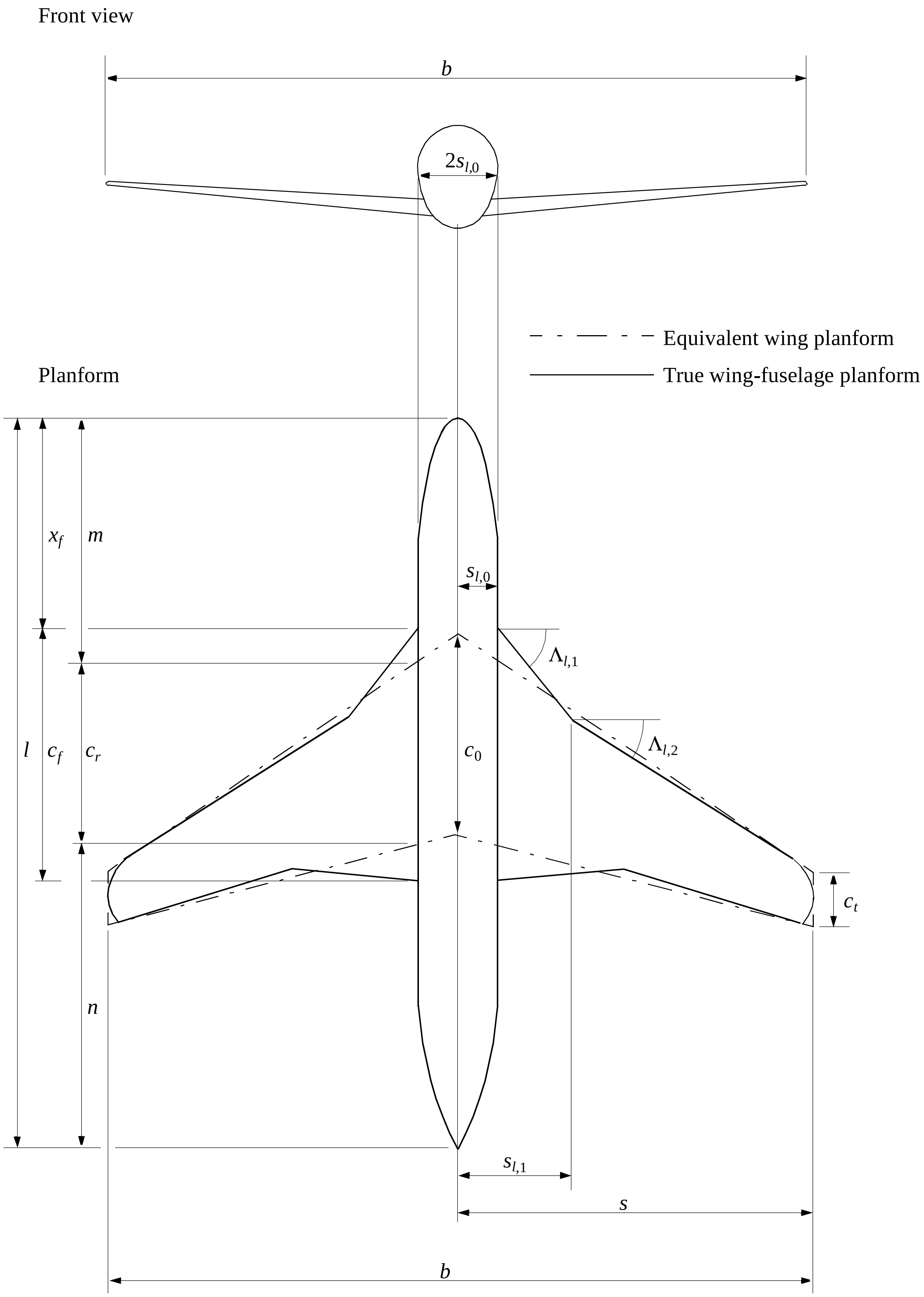
* Since the equivalent wing planform is a frequently used artifice a more detailed treatment is given in Addendum A to Item No. 76003.



Front view
b
2$s_{l,0}$
Equivalent wing planform
True wing-fuselage planform
Planform
x_f
m
$s_{l,0}$
$\Lambda_{l,1}$
$\Lambda_{l,2}$
l
c_f
c_r
c_0
c_t
n
$s_{l,1}$
s
b

Sketch 3.1

If there are N cranks in the leading edge of the exposed wing planform, the distance of the leading edge of c_r from the fuselage nose, m, is given by the equation

$$m = x_f + \sum_{i=1}^{N} (\tan\Lambda_{l,i} - \tan\Lambda_{l,i+1}) \frac{(s_{l,i} - s_{l,0})(s - s_{l,i})}{(s - s_{l,0})} . \tag{3.2}$$

This equation results from constructing the straight leading-edge of the equivalent wing so that the area enclosed by the leading edge of the true wing exposed planform and the straight line joining the leading edges of c_f and c_t is equal to the triangular area enclosed by the straight lines joining the leading edges of c_r, c_f and c_t. If there is a notch in the leading edge of the true wing, so that $\Lambda_{l,i} = \pm 90°$, a geometric construction on this principle may be used to determine m, instead of Equation (3.2).*

If there are no cranks in the leading edge of the true wing planform,

$$m = x_f . \tag{3.3}$$

Having defined c_r and c_t, the equivalent gross wing planform is obtained by extrapolating to the fuselage centre-line the lines joining their leading and trailing edges, see Sketch 3.1. With the equivalent wing planform established, the remaining parameters needed in the calculation of x_h can be determined and they are presented briefly in the following equations.

If there are N cranks in the leading edge of the true wing planform,

$$\tan\Lambda_{1/2} = \sum_{i=1}^{N} (\tan\Lambda_{l,i} - \tan\Lambda_{l,i+1}) \left[\frac{s_{l,i} - s_{l,0}}{s - s_{l,0}}\right]^2 + \tan\Lambda_{l,N+1} + \frac{c_t - c_r}{2(s - s_{l,0})} . \tag{3.4}$$

If there are notches in the leading edge of the true wing a geometric construction using the known positions of c_r and c_t may be used to determine $\tan\Lambda_{1/2}$, instead of Equation (3.4)†.

If there are no cranks,

$$\tan\Lambda_{1/2} = \tan\Lambda_{l,1} + \frac{c_t - c_r}{2(s - s_{l,0})} , \tag{3.5}$$

where $\Lambda_{l,1}$ is here the leading-edge sweep of the true wing planform.

$$n = l - m - c_r . \tag{3.6}$$

$$c_0 = \frac{s\,c_r - s_{l,0}\,c_t}{s - s_{l,0}} . \tag{3.7}$$

$$\lambda = \frac{c_t}{c_0} . \tag{3.8}$$

* In Equation (3.2) this would amount to setting $\tan\Lambda_{l,i} = 0$ and adding to the summation a term $\Delta x_{l,i}(s + s_{l,0} - 2s_{l,i})/(s - s_{l,0})$, where $\Delta x_{l,i}$ is the local increase in the chordwise distance of the wing leading-edge aft of the fuselage nose.

† In Equation (3.4) this would amount to setting $\tan\Lambda_{l,i} = 0$ and adding to the summation a term $2\Delta x_{l,i}(s_{l,i} - s_{l,0})/(s - s_{l,0})^2$.

$$\bar{c} = c_0 \frac{(1+\lambda)}{2} \tag{3.9}$$

$$\bar{\bar{c}} = 2c_0 \frac{(1+\lambda+\lambda^2)}{3(1+\lambda)} \tag{3.10}$$

$$S = b\bar{c} \tag{3.11}$$

$$A = \frac{b^2}{S} \tag{3.12}$$

$$\tan\Lambda_0 = \tan\Lambda_{½} + \frac{2(1-\lambda)}{A(1+\lambda)} \tag{3.13}$$

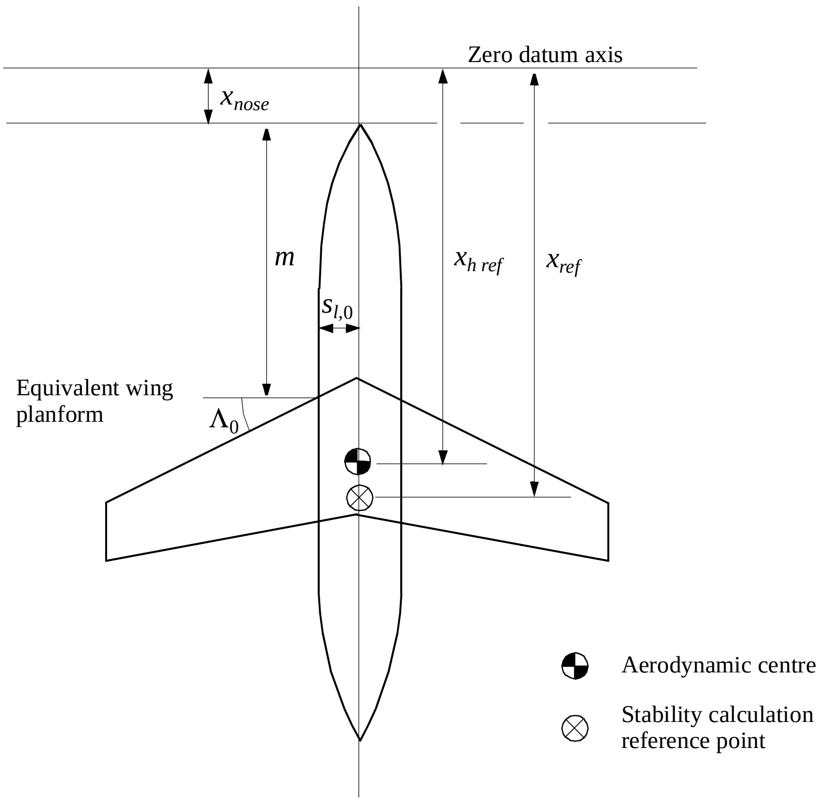
$$\bar{\bar{x}} = c_0\left(\frac{1+2\lambda}{12}\right)A\tan\Lambda_0 \ . \tag{3.14}$$

The aerodynamic properties a and $\bar{x}/\bar{\bar{c}}$ are found from Item No.70011 using the equivalent wing values of $A\tan\Lambda_{½}$, βA and λ.

For use in Equation (2.2), the fuselage width and height, d and h, are measured at the leading edge of c_r.

4. GENERAL REFERENCE POSITION

Normally, aircraft stability calculations will be based on a reference chord, c_{ref}, which is different from $\bar{\bar{c}}$, and will be referred to a reference point which is not located at the leading edge of the aerodynamic mean chord of the equivalent wing. Thus, it will usually be necessary to convert the calculated value of $x_h / \bar{\bar{c}}$ to the required reference point and chord.



Sketch 4.1

If the reference point for stability calculations is a chordwise distance x_{ref} from an arbitrary, transverse zero-datum axis, and the fuselage nose is a chordwise distance x_{nose} from the zero-datum (see Sketch (4.1)), then the distance of the aerodynamic centre from the stability calculation reference point, based on the reference chord c_{ref}, is given by the equation

$$\frac{x_{h\,ref} - x_{ref}}{c_{ref}} = \frac{m - s_{l,\,0}\tan\Lambda_0 + x_{nose} + \bar{\bar{x}}}{c_{ref}} + \left(\frac{x_h}{\bar{\bar{c}}}\right)\frac{\bar{\bar{c}}}{c_{ref}} - \frac{x_{ref}}{c_{ref}}. \tag{4.1}$$

A negative value of $(x_{h\,ref} - x_{ref})/c_{ref}$ implies that the aerodynamic centre is forward of the stability calculation reference point, a positive value implies that it is aft.

5. APPLICABILITY AND ACCURACY

5.1 Applicability

The method calculates the aerodynamic-centre position of conventional aircraft wing-fuselage combinations, with flaps undeployed, for the linear part of the lift-curve slope where the rate of change of pitching moment with lift is also essentially linear. The method applies when the flow over the configuration is wholly subsonic and fully attached, but it is only applicable for Mach numbers where the variation of the lift-curve slope of the configuration with Mach number is adequately represented by subsonic similarity parameters, because this is assumed in Item No. 70011 from which the lift-curve slope is determined. Wing-fuselage combinations with aspect ratios greater than 5 and equivalent wing sweep angles between 0 and 45° are covered by the method, but it should be used only for closed or nearly-closed fuselage shapes and not for fuselages with severely tapered or truncated afterbodies where flow separation might occur.

The method has only been tested for two aircraft with leading-edge cranks, the cranks occurring approximately $0.3(s - s_{l,\,0})$ out from the fuselage side, with a reduction in leading-edge sweep of approximately 20°. Caution should therefore be exercised if the method is applied to configurations with more severe leading-edge cranks. The method has been tested for fifteen configurations with trailing-edge cranks, the cranks being located between $0.2(s - s_{l,\,0})$ and $0.35(s - s_{l,\,0})$ from the fuselage side, with increases in wing trailing-edge sweep angle of between 10° and 25°.

The effects of wing height and wing-fuselage setting angle are small in comparison with those of fuselage length and diameter (see Derivation 2 for example) and are not taken into account.

The term $\Delta x_h / \bar{\bar{c}}$ which allows for the effect of the fuselage is applied as a correction to a wing-alone aerodynamic centre position and may therefore be used independently of the equivalent wing aerodynamic centre data. It may, for example, be combined with experimentally derived wing-alone data to study the effect of adding a fuselage or with experimentally derived wing-fuselage data to study the effect of adding a fuselage or with experimentally derived wing-fuselage data to study the effect of changes to the fuselage geometry of an existing configuration.

5.2 Accuracy

Comparison of the predictions of the method with experimental data from Derivations 5 to 14 indicates that the method can be expected to predict $x_h / \bar{\bar{c}}$ to within ± 0.03. Thirty-four configurations were studied, covering the parameter ranges shown in Table 5.1.

TABLE 5.1
Parameter Variation of Data

A	6	to	12	d/b	0.08	to	0.14
$\Lambda_{1/2}$	0	to	45°	d/c_r	0.4	to	0.9
$A \tan \Lambda_{1/2}$	0	to	7.5	m/c_r	1.0	to	3.5
λ	0.2	to	1.0	n/c_r	1.5	to	3.0

See also remarks concerning crank geometry in Section 5.1.

For seven configurations experimental data were available for both the wing alone and the wing-fuselage combination. Comparison of the experimental value of $x_h / \bar{\bar{c}}$ with the value obtained by combining $\Delta x_h / \bar{\bar{c}}$ with the experimental value of the wing-alone aerodynamic centre position suggests that the error associated with the term $\Delta x_h / \bar{\bar{c}}$ is within ± 0.02. The larger error band associated with the purely theoretical prediction of aerodynamic centre position probably reflects the error between the aerodynamic centre positions of the equivalent and true wings and is consistent with the conclusion of Item No.70011 which quotes the mean error in predicted values of wing-alone aerodynamic centre as being within $\pm 0.03\bar{\bar{c}}$.

6. DERIVATION

Estimation Method

1. SCHLICHTING, H. — Calculation of the influence of a body on the position of the aerodynamic centre of aircraft with sweptback wings. RAE tech. Note Aero. 1879, 1947.

2. ANSCOMBE, A; RANEY, D.J. — Low-speed tunnel investigation of the effect of the body on C_{m0} and aerodynamic centre of unswept wing-body combinations. RAE Rep. Aero. 2323, 1949.

3. WEBER, J.; KIRBY, D.A.; KETTLE, D.J. — An extension of Multhopp's method of calculating the spanwise loading of wing-fuselage combinations. ARC R & M 2872, 1956.

4. – — Unpublished data on theoretical[3] spanwise load-grading of wings with and without a fuselage. Hawker Siddeley Aviation Ltd, Hatfield, 1976.

Supporting Data

5. SIVELLS, J.C.; SPOONER, S.H. — Investigation in the Langley 19-foot pressure tunnel of two wings of NACA 65-210 and 64-210 airfoil sections with various type flaps. NACA tech. Note 1579, 1947.

6. PRATT, G.L.; SHEILDS, E.R. — Low-speed longitudinal characteristics of a 45° sweptback wing of aspect ratio 8 with high-lift and stall-control devices at Reynolds numbers from 1,500,000 to 4,800,000. NACA RM L51J04 (TIL 3038), 1951.

7. FINK, M.P.; FREEMAN, D.C.; GREER, H.D. — Full-scale wind-tunnel investigation of the static longitudinal and lateral characteristics of a light single-engine airplane. NASA tech. Note D-5700, 1970.

8. LOVELL, D.A. — A wind-tunnel investigation of the effects of flap span and deflection angle, wing planform and a body on the high lift performance of a 28° swept wing. RAE tech. Rep. 76030, 1976.

9. – — Results of wind-tunnel tests. Aircraft Research Association Ltd.

10. – — Unpublished wind-tunnel data from British Aircraft Corporation Ltd, Weybridge.

11. – — Unpublished wind-tunnel data from Hawker Siddeley Aviation Ltd, Hatfield.

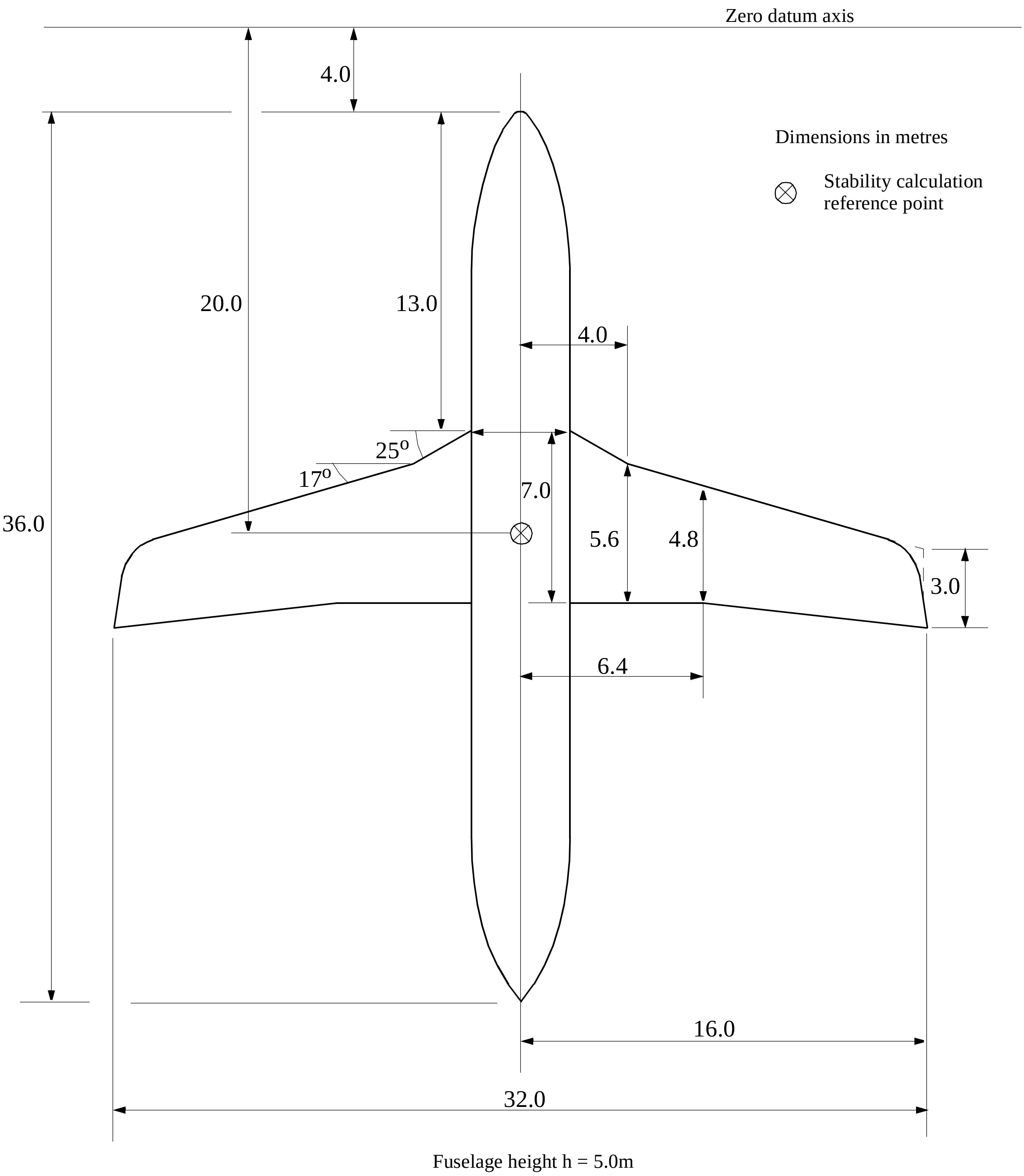
12. – — Unpublished wind-tunnel data from Hawker Siddeley Aviation Ltd, Woodford.

13. – — Unpublished wind-tunnel data from Aerospatiale, Tolouse.

14. – — Unpublished wind-tunnel data from BAe, Aircraft Group, Hatfield.

7. EXAMPLE

Find the position of the aerodynamic centre of the wing-fuselage planform shown in Sketch 7.1 for a Mach number of 0.48, based on a reference chord of 4.5 m and referred to a point 20 m aft of a zero-datum axis which is 4 m ahead of the fuselage nose.



Sketch 7.1

STEP 1 Determine Equivalent Wing Geometry

From the sketch, extrapolation of the wing leading-edge to maximum span gives a tip chord, c_t, of 3.0 m.

For the dimensions shown in the sketch the area of the exposed planform of the true wing is 125.04 m^2.

Thus from Equation (3.1)

$$c_r = \frac{S_e}{s - s_{l,0}} - c_t = \frac{125.04}{16.0 - 2.0} - 3.0 = 5.931 \text{ m}.$$

As there is a crank in the wing leading-edge, $N = 1$, and the distance of the leading edge of c_r from the fuselage nose in given by Equation (3.2), *i.e.*

$$m = x_f + \sum_{i=1}^{N} (\tan \Lambda_{l,\,i} - \tan \Lambda_{l,\,i+1}) \frac{(s_{l,\,i} - s_{l,0})(s - s_{l,\,i})}{(s - s_{l,0})}$$

$$= 13.0 + (\tan 25° - \tan 17°) \frac{(4.0 - 2.0)(16.0 - 4.0)}{(16.0 - 2.0)} = 13.275 \text{ m}.$$

The half-chord sweep of the equivalent wing is obtained from Equation (3.4), *i.e.*

$$\tan \Lambda_{½} = \sum_{i=1}^{N} (\tan \Lambda_{l,\,i} - \tan \Lambda_{l,\,i+1}) \left[\frac{s_{l,\,i} - s_{l,0}}{s - s_{l,0}}\right]^2 + \tan \Lambda_{l,\,N+1} + \frac{c_t - c_r}{2(s - s_{l,0})}$$

$$= (\tan 25° - \tan 17°) \left[\frac{4.0 - 2.0}{16.0 - 2.0}\right]^2 + \tan 17° + \frac{3.0 - 5.931}{2(16.0 - 2.0)} = 0.2043.$$

The remaining geometric parameters to be calculated in determining the position of the aerodynamic centre are evaluated from Equations (3.6) to (3.14) as follows.

$$n = l - m - c_r = 36.0 - 13.275 - 5.931 = 16.794 \text{ m},$$

$$c_0 = \frac{s\, c_r - s_{l,0}\, c_t}{s - s_{l,0}} = \frac{16.0 \times 5.931 - 2.0 \times 3.0}{16.0 - 2.0} = 6.350 \text{ m},$$

$$\lambda = \frac{c_t}{c_0} = \frac{3.0}{6.350} = 0.472,$$

$$£ = c_0 \frac{(1 + \lambda)}{2} = \frac{6.350(1 + 0.472)}{2} = 4.674 \text{ m},$$

$$! = 2c_0 \frac{(1 + \lambda + \lambda^2)}{3(1 + \lambda)} = \frac{2 \times 6.350(1 + 0.472 + 0.472^2)}{3(1 + 0.472)} = 4.874 \text{ m},$$

$$S = b\,\bar{c} = 32.0 \times 4.674 = 149.6 \text{ m}^2,$$

$$A = \frac{b^2}{S} = \frac{32.0^2}{149.6} = 6.845,$$

$$\tan\Lambda_0 = \tan\Lambda_{½} + \frac{2(1-\lambda)}{A(1+\lambda)} = 0.2043 + \frac{2(1-0.472)}{6.845(1+0.472)} = 0.3091,$$

$$\bar{\bar{x}} = c_0\left(\frac{1+2\lambda}{12}\right)A\tan\Lambda_0 = 6.350\left(\frac{1+2\times0.472}{12}\right)6.845\times0.3091 = 2.177 \text{ m}.$$

The fuselage width and height at the leading edge of c_r are respectively

$$d = 4.0 \text{ m},$$

and $$h = 5.0 \text{ m}.$$

STEP 2 Determine a and $\bar{x}/\bar{\bar{c}}$ for the Equivalent Wing

The free-stream Mach number is 0.48, so

$$\beta = (1-M^2)^{½} = (1-0.48^2)^{½} = 0.8773 .$$

Therefore $$\beta A = 0.8773\times6.845 = 6.005 .$$

From STEP 1,

$$A\tan\Lambda_{½} = 6.845\times0.2043 = 1.398 .$$

From Item No. 70011, for $\beta A = 6.005$, $A\tan\Lambda_{½} = 1.398$ and $\lambda = 0.472$.

$$\frac{a}{A} = 0.712,$$

so $$a = 0.712\times6.845 = 4.874 \text{ per radian},$$

and $$\frac{\bar{x}}{\bar{\bar{c}}} = 0.243 .$$

STEP 3 Determine the Fuselage Effect

Using the values from STEP 1

$$\frac{m}{c_r} = \frac{13.275}{5.931} = 2.238$$

and $$\frac{n}{c_r} = \frac{16.794}{5.931} = 2.832 .$$

So, using Figure 1 for these values of m/c_r and n/c_r ,

$$F = 4.86 .$$

Using previously calculated values,

$$\beta\frac{d}{c_r} = 0.8773 \times \frac{4.0}{5.931} = 0.592 \, .$$

From Figure 2 this gives a value for G of

$$G = 1.081 \quad .$$

As $d / b = 4.0 / 32.0 = 0.125$, it is necessary to evaluate K_1 from a cross-plot of values at $d / b = 0.08$, 0.12 and 0.16. As determined previously, $A\tan\Lambda_{½} = 1.398$ and $\lambda = 0.472$, so the values of K_1 obtained from Figure 3 are 0.0045, 0.0195 and 0.0360. Therefore from the cross-plot the value appropriate to $d / b = 0.125$ is

$$K_1 = 0.0205 \, .$$

The value of K_2 is obtained from Figure 4 for $A\tan\Lambda_{½} = 1.398$ and $\beta A = 6.005$, giving

$$K_2 = 0.0039 \, .$$

Therefore the effect of the fuselage on aerodynamic centre position is, using Equation (2.2),

$$\begin{aligned}\frac{\Delta x_h}{\bar{\bar{c}}} &= \frac{c_r d^2 F G}{\bar{\bar{c}}\, a S}\left[1 + 0.15\left(\frac{h}{d} - 1\right)\right] - (K_1 + \lambda K_2) \\ &= \frac{5.931 \times 4.0^2 \times 4.86 \times 1.081}{4.874 \times 4.874 \times 149.6}\left[1 + 0.15\left(\frac{5.0}{4.0} - 1\right)\right] - (0.0205 + 0.472 \times 0.0039) \\ &= 0.123 \quad .\end{aligned}$$

STEP 4 Determine the Aerodynamic Centre Position of the Wing-Fuselage Combination

The position of the aerodynamic centre relative to the leading edge of the aerodynamic mean chord of the equivalent wing is obtained from Equation (2.1). So

$$\frac{x_h}{\bar{\bar{c}}} = \frac{\bar{x}}{\bar{\bar{c}}} - \frac{\Delta x_h}{\bar{\bar{c}}} = 0.243 - 0.123 = 0.120 \, .$$

The aerodynamic centre can now be referred to the required stability calculation reference point and based on the required reference chord by means of Equation (4.1). Thus

$$\begin{aligned}\frac{x_{h\,ref} - x_{ref}}{c_{ref}} &= \frac{m - s_{l,\,0}\tan\Lambda_0 + x_{nose} + \bar{\bar{x}}}{c_{ref}} + \left(\frac{x_h}{\bar{\bar{c}}}\right)\frac{\bar{\bar{c}}}{c_{ref}} - \frac{x_{ref}}{c_{ref}} \\ &= \frac{13.275 - 2.0 \times 0.3091 + 4.0 + 2.177}{4.5} + (0.120)\frac{4.874}{4.5} - \frac{20.0}{4.5} \\ &= -\,0.129 \quad .\end{aligned}$$

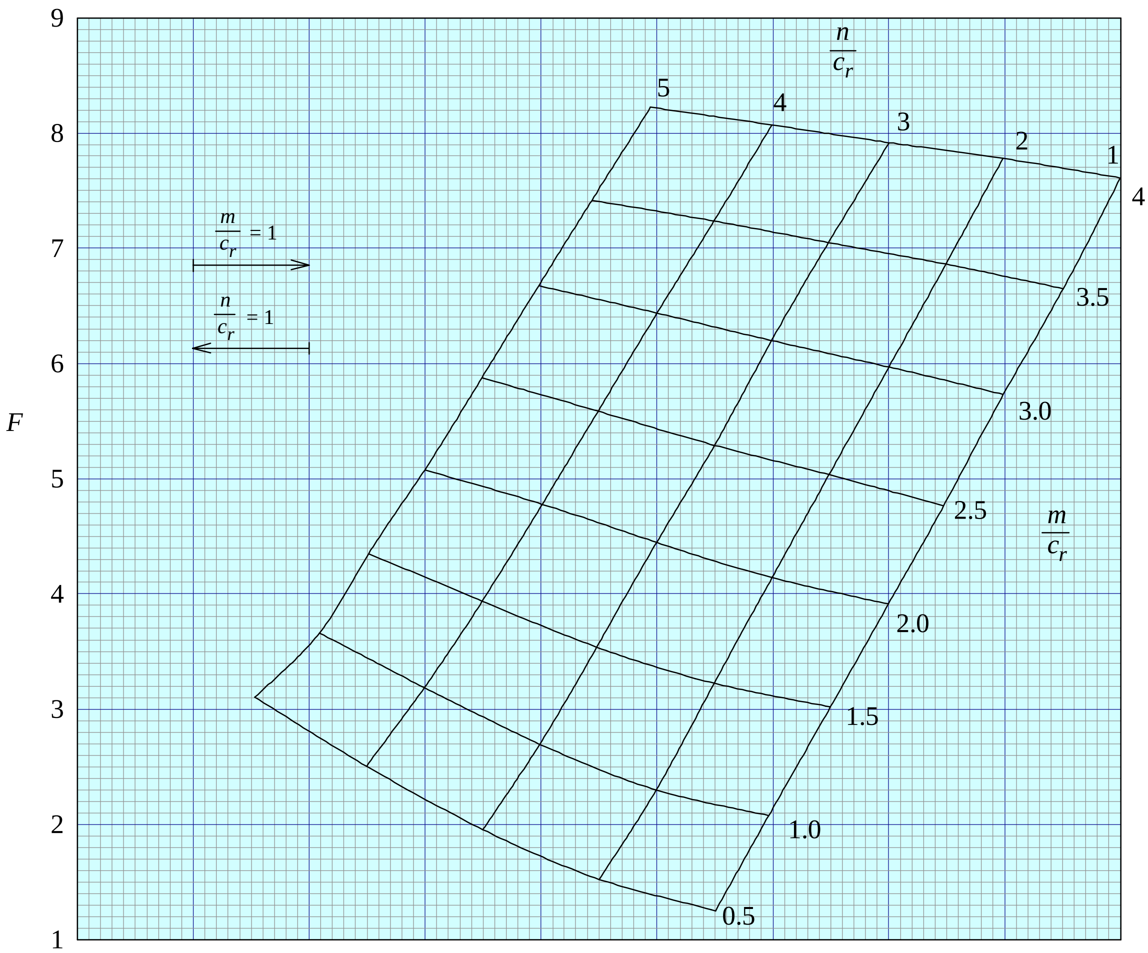


FIGURE 1 EFFECT OF FOREBODY AND AFTERBODY LENGTHS

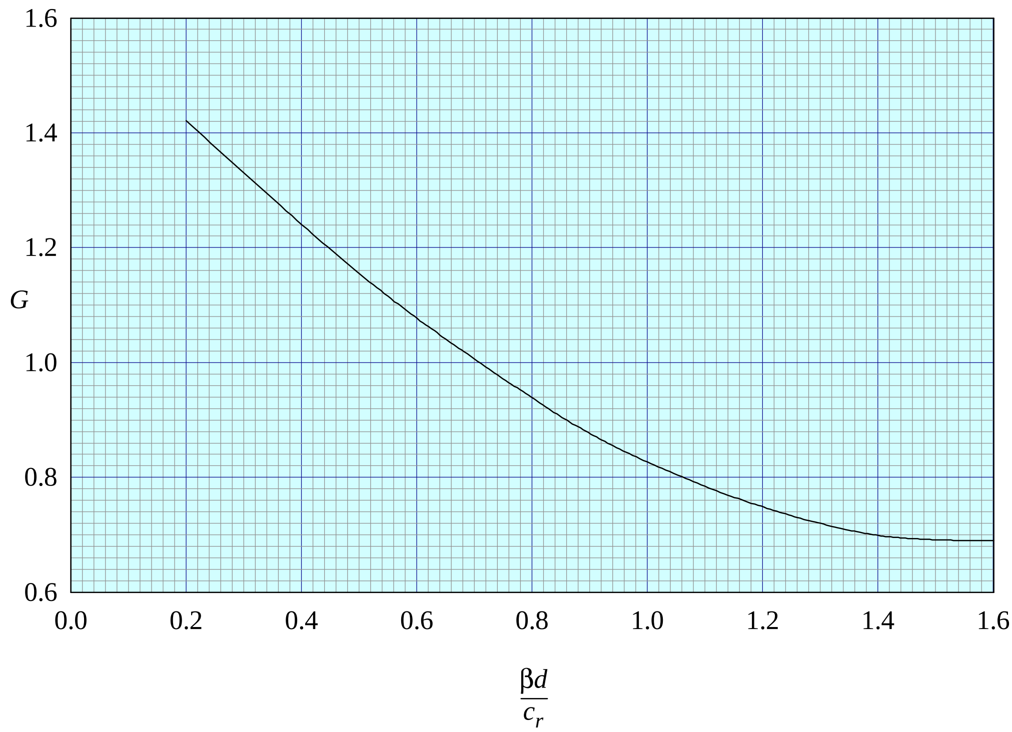
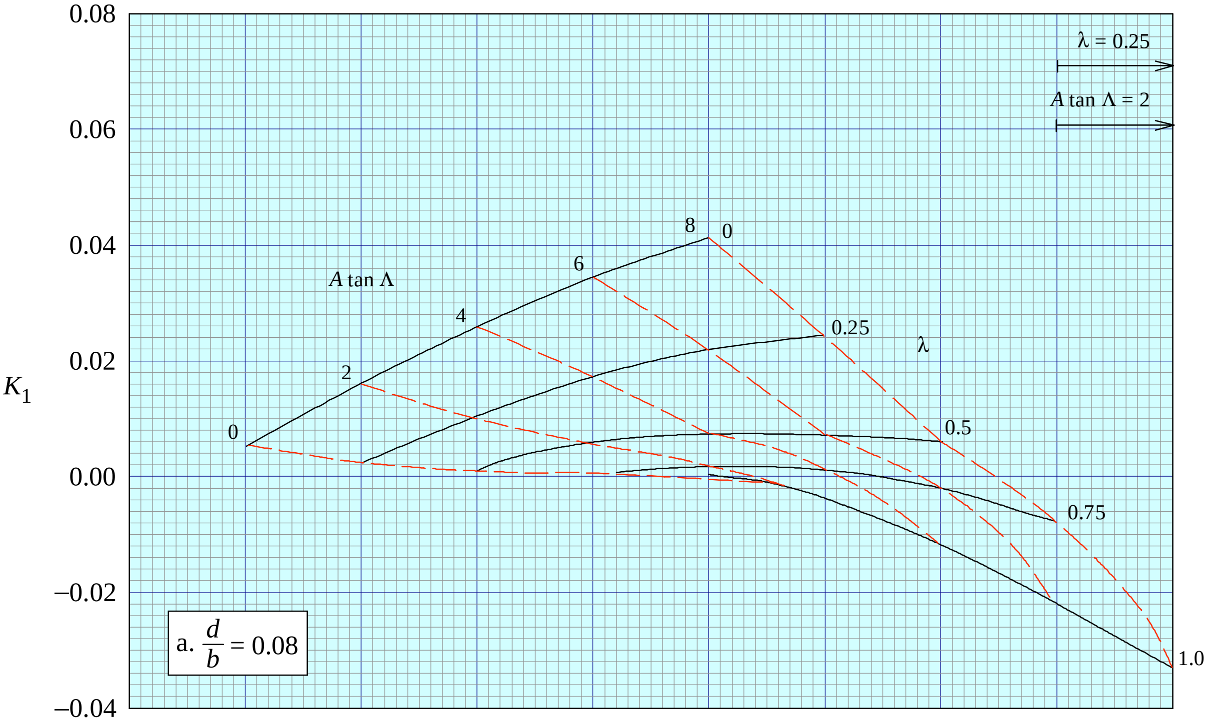


FIGURE 2 EFFECT OF FUSELAGE WIDTH



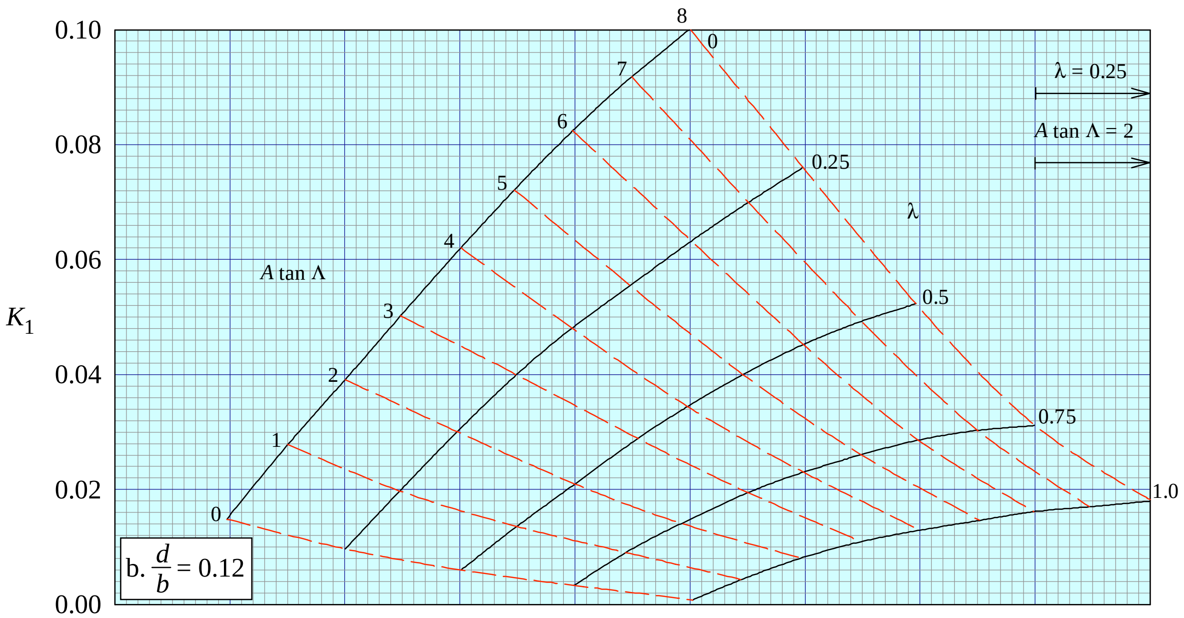


FIGURE 3 EFFECT OF WING SWEEPBACK

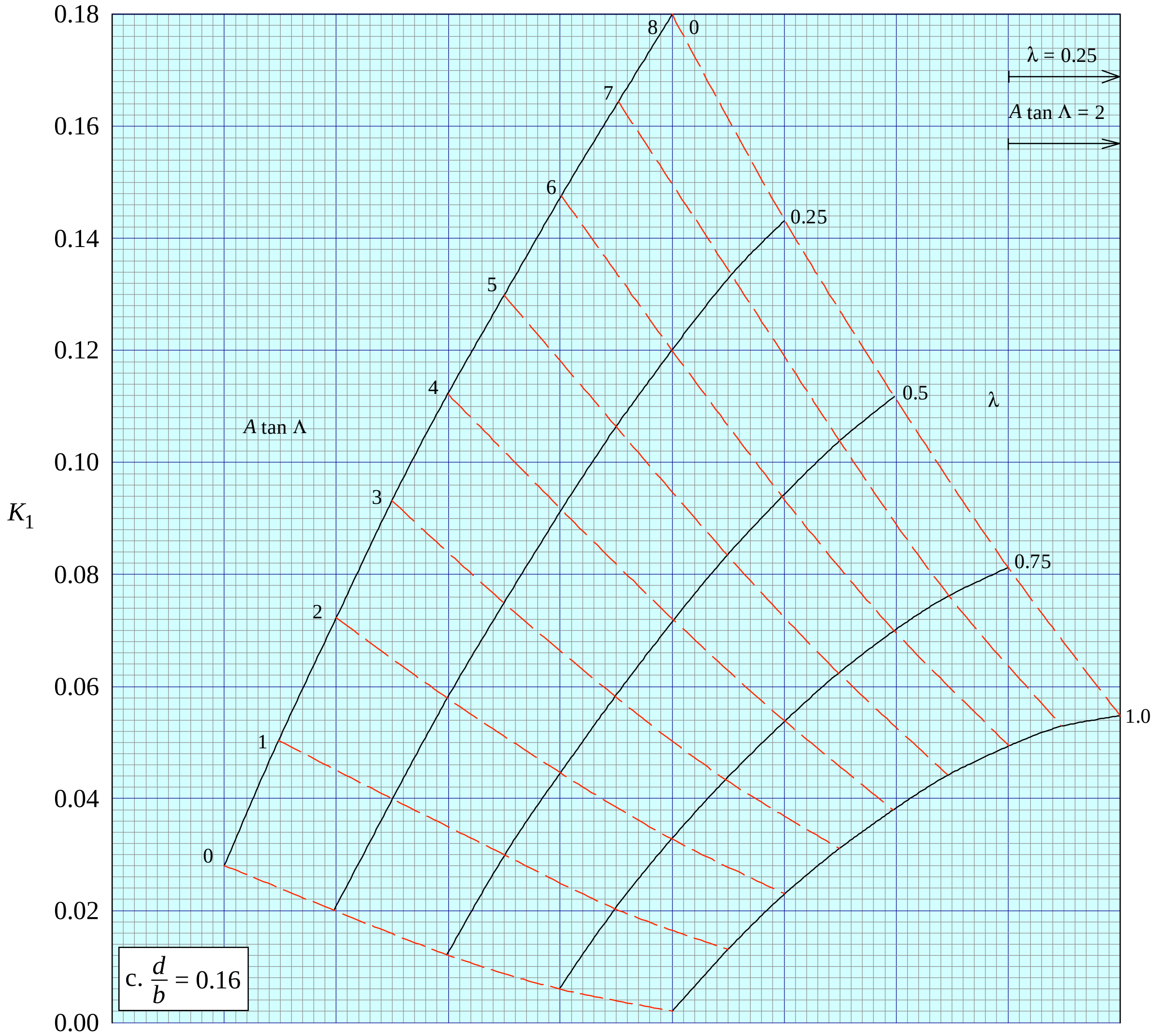


FIGURE 3 (concluded)

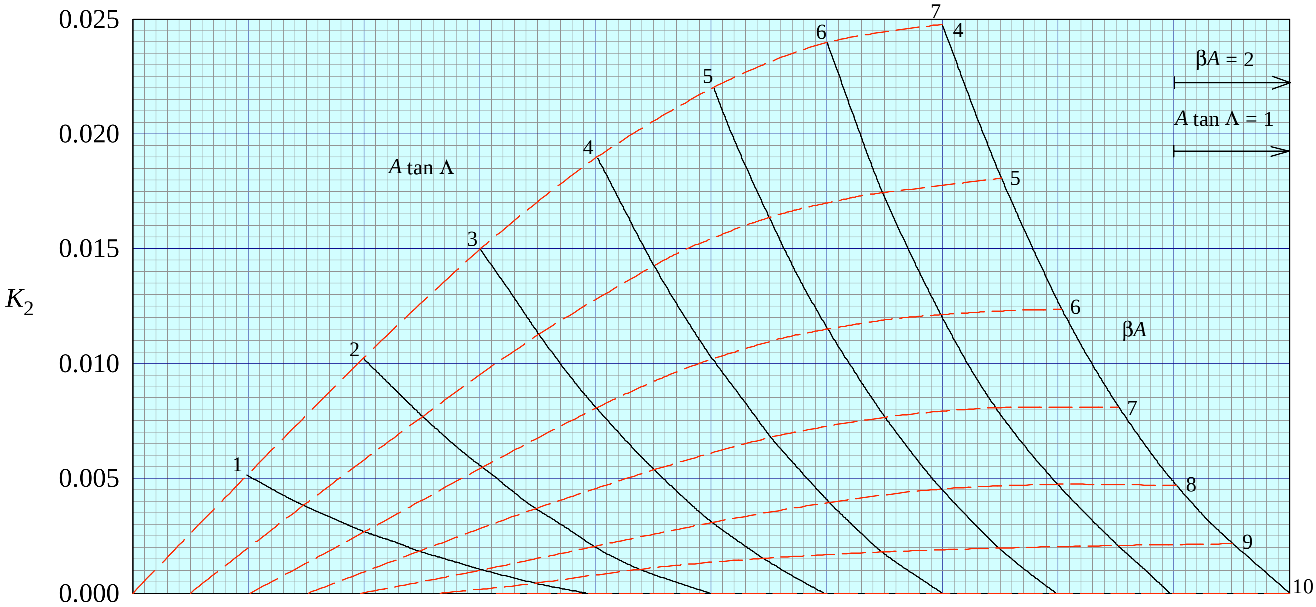


FIGURE 4 CORRECTION TO WING SWEEPBACK EFFECT

THE PREPARATION OF THIS DATA ITEM

The work on this particular Data Item, which supersedes Item No. Aero A.08.01.01, was monitored and guided by the Aerodynamics Committee, which first met in 1942 and now has the following membership.

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The work on this Item was carried out in the Aircraft Motion Group of ESDU under the supervision of Mr P.D. Chappell, Group Head. The member of staff who undertook the technical work involved in the initial assessment of the available information and the construction and subsequent development of the Item was:

Mr R.W. Gilbey – Engineer

Particular assistance in the preparation of this Item was received from Mr J.A. Jupp and Mr A.K. Blackwell from Hawker Siddeley Aviation Ltd, Hatfield, and from Mr M.R. Smith and Mr J.W. Gilbert from the British Aircraft Corporation Ltd, Weybridge.