AEROFOIL AND WING PITCHING MOMENT COEFFICIENT AT ZERO ANGLE OF ATTACK DUE TO DEPLOYMENT OF TRAILING-EDGE SINGLE-SLOTTED FLAPS AT LOW SPEEDS

1. NOTATION AND UNITS

		SI	*British*
A	aspect ratio, $2s/\bar{c}$		
$(a_1)_0$	basic aerofoil lift-curve slope in incompressible flow	rad^{-1}	rad^{-1}
C_L	lift coefficient; $(lift)/qc$ for aerofoil, $(lift)/qS$ for wing		
C_{L0}	C_L at zero angle of attack for aerofoil, based on c		
ΔC_{L0t}	increment in lift coefficient at zero angle of attack due to deployment of trailing-edge single-slotted flap on aerofoil, based on c (see Item No. 94030)		
$\Delta C'_{L0t}$	increment in lift coefficient at zero angle of attack due to deployment of trailing-edge single-slotted flap on aerofoil, based on c' (see Item No. 94030)		
$\Delta C'_{L1}$	increment in lift coefficient associated with deployment of trailing-edge single-slotted flap on aerofoil with lift-curve slope of 2π, based on c', Figure 2 (from Item No. 94030)		
C_m	pitching moment coefficient; $(pitching\ moment)/qc^2$ for aerofoil, $(pitching\ moment)/qS\bar{\bar{c}}$ for wing, referenced to $c/4$ for aerofoil and $\bar{\bar{c}}/4$ for wing, see Sketch 1.1		
C_{m0}	C_m for aerofoil zero lift, based on c^2, see Section 7.1		
C_{m0i}	inviscid value of C_{m0}		
$C_{m\alpha 0}$	pitching moment coefficient at zero angle of attack for aerofoil, based on c^2 and referenced to $c/4$, approximated as C_{m0}, see Section 7.1		
$C_{mw\alpha 0}$	pitching moment coefficient at zero angle of attack for wing, based on $S\bar{\bar{c}}$ and referenced to $\bar{\bar{c}}/4$		
$\Delta C_{mt\alpha 0}$	increment in pitching moment coefficient at zero angle of attack due to deployment of trailing-edge single-slotted flap on aerofoil, based on c^2 and referenced to $c/4$, see Equation (3.1)		
$\Delta C_{mtw\alpha 0}$	increment in pitching moment coefficient at zero angle of attack due to deployment of trailing-edge single-slotted flap on wing, based on $S\bar{\bar{c}}$ and referenced to $\bar{\bar{c}}/4$, see Equation (3.6)		

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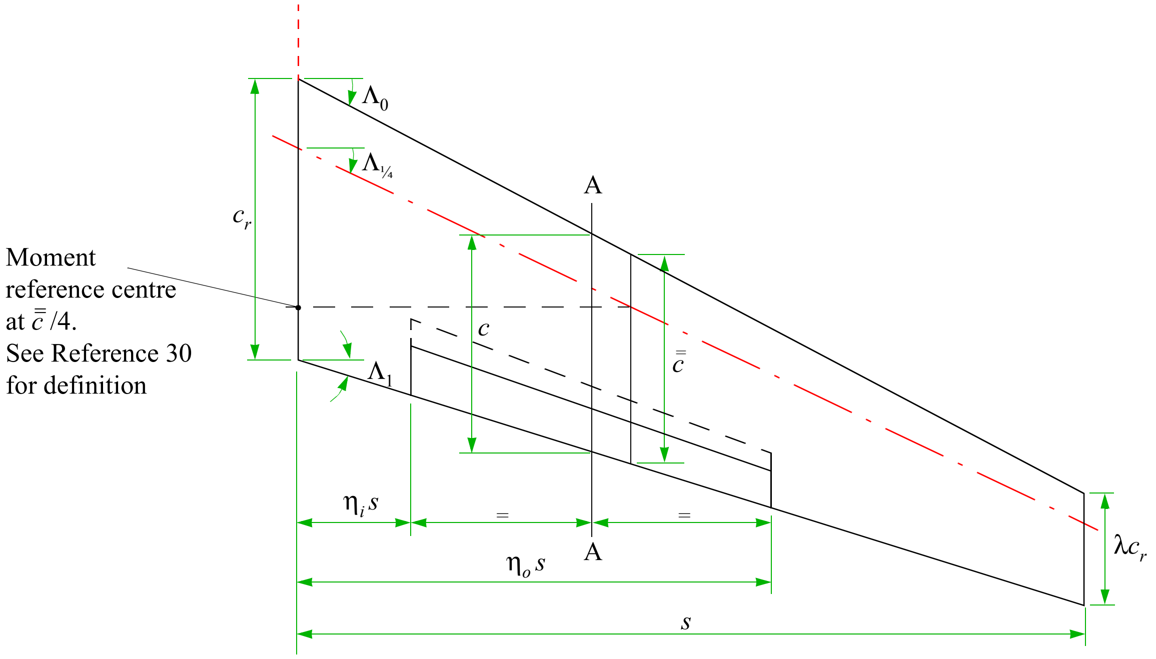
c	basic (plain) aerofoil chord or wing chord at flap mid-span (*i.e.* chord with high-lift devices undeployed), see Sketches 1.1 and 1.2	m	ft
c'	extended aerofoil chord, *i.e.* chord with trailing-edge single-slotted flap deployed, see Sketch 1.2	m	ft
$\bar{c}$	wing geometric mean chord	m	ft
$\bar{\bar{c}}$	wing aerodynamic mean chord, see Item No. 76003	m	ft
c_r	wing root (centre-line) chord, see Sketch 1.1	m	ft
c_{t1}	chord of trailing-edge single-slotted flap, see Sketch 1.2	m	ft
Δc_{t1}	increment in c_{t1}, see Sketch 1.2	m	ft
c'_{t1}	extended chord of trailing-edge single-slotted flap, see Sketch 1.2	m	ft
F	viscous correction factor used in calculation of C_{m0}, see Equation (7.1).		
h_2	centre of incremental lift at zero angle of attack due to trailing-edge single-slotted flap deployment on aerofoil section expressed as fraction of basic aerofoil chord, measured positive aft from its quarter-chord position		
h'_2	empirical centre of incremental lift at zero angle of attack due to trailing-edge single-slotted flap angular deflection on aerofoil aft from extended aerofoil quarter-chord position, see Equation (3.4)		
h'_{2T}	theoretical value on which h'_2 is based, see Equation (3.3) and Figure 3		
J_{t1}	correlation factor for trailing-edge single-slotted flap, Figure 1 (from Item No. 94030)		
K	part-span factor; pitching moment coefficient increment due to part-span trailing-edge single-slotted flaps extending symmetrically from wing centre-line divided by pitching moment coefficient increment due to full-span trailing-edge single-slotted flaps at same deflection angle, Figure 4.		
K_f	flap type correlation factor, (= 1.0)		
$K_{f\Lambda}$	flap type correlation factor for wing sweep, (= 1.0)		
K_i	value of K corresponding to $\eta = \eta_i$, required in Equation (3.6)		
K_o	value of K corresponding to $\eta = \eta_o$, required in Equation (3.6)		
K_Λ	part-span factor dependent on wing sweep effect, Equation (3.9) and Figures 5a to 5f		
$K_{\Lambda i}$	value of K_Λ corresponding to $\eta = \eta_i$, required in Equation (3.6)		

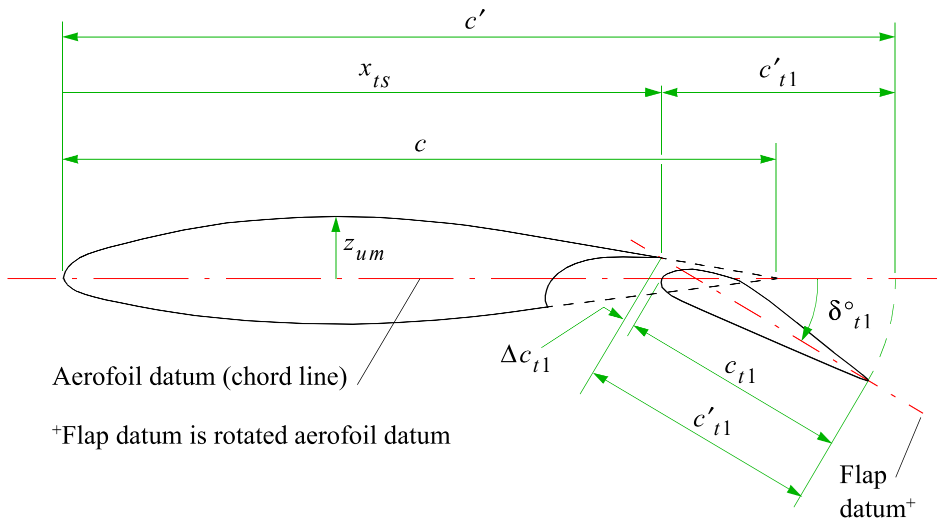
$K_{\Lambda o}$	value of K_{Λ} corresponding to $\eta = \eta_o$, required in Equation (3.6)		
M	free-stream Mach number		
p	parameter in Equation (3.9) for K_{Λ}, see Equation (3.10)		
q	free-stream kinetic pressure	N/m^2	lbf/ft^2
R_c	aerofoil Reynolds number, based on free-stream conditions and c		
$R_{\bar{\bar{c}}}$	wing Reynolds number, based on free-stream conditions and $\bar{\bar{c}}$		
S	wing planform area, $2s\bar{c}$	m^2	ft^2
s	wing semi-span, see Sketch 1.1	m	ft
t	maximum thickness of aerofoil	m	ft
x_{ts}	chordwise location of flap-shroud trailing edge, see Sketch 1.2	m	ft
z_{um}	maximum upper-surface ordinate of basic aerofoil, see Sketch 1.2	m	ft
$\delta_{t1}°$	deflection of trailing-edge single-slotted flap, positive trailing-edge down, see Sketch 1.2	deg	deg
η	spanwise distance from wing centre-line as fraction of semi-span		
η_i	value of η at inboard limit of flap, see Sketch 1.1		
η_o	value of η at outboard limit of flap, see Sketch 1.1		
Λ_0	wing leading-edge sweep angle, see Sketch 1.1	deg	deg
$\Lambda_{1/4}$	wing quarter-chord sweep angle, see Sketch 1.1	deg	deg
$\Lambda_{1/2}$	wing half-chord sweep angle	deg	deg
Λ_1	wing trailing-edge sweep angle, see Sketch 1.1	deg	deg
λ	wing taper ratio (tip chord/root chord)		

Subscripts

$(\ )_{expt}$	denotes experimental value
$(\ )_{pred}$	denotes predicted value



Sketch 1.1 Wing notation (flaps undeployed)



Sketch 1.2 Deployed single-slotted flap notation (at section AA)

2. INTRODUCTION

This Item provides a method to obtain the increment in pitching moment coefficient at zero angle of attack due to deployment of trailing-edge single-slotted flaps at low speeds, either on an aerofoil or on a wing.

For aerofoils the method predicts the centre of lift position, h_2, due to single-slotted flap deployment, based on the thin-aerofoil theory of Derivation 26 and modified to obtain correlation with the experimental data of Derivations 3, 4, 5, 7, 8, 11, 12, 14, 15 and 21. This is combined with the increment in aerofoil lift coefficient calculated from Item No. 94030 (Derivation 2) to estimate the pitching moment coefficient increment.

For wings with full-span flaps, factors, dependent on planform geometry, are applied to the pitching moment coefficient increment on a section that is representative of the wing, to allow for three-dimensional effects. Derivations 27 and 28 are used as the basis for these factors, with some adjustment to the simple theoretical assumptions. For wings with part-span flaps, additional factors are introduced that are dependent on taper ratio, aspect ratio, sweep and spanwise extent of the flap.

Section 3 describes the prediction method and Section 4 discusses Mach number and Reynolds number effects. The applicability and accuracy of the method are addressed in Section 5. The Derivation and References are given in Section 6. Section 7 provides worked examples illustrating the use of the Item for an aerofoil and a wing.

3. PREDICTION METHOD

The method for aerofoils requires the use of Item No. 94030 to determine the lift increment characteristics of the aerofoil/flap combination from which to derive the pitching moment coefficient increment.

For wings, the *streamwise* section, flap geometries and angles at the mid-span of the flap panel are taken to be representative of the wing/flap system, see Sketches 1.1 and 1.2. The method again requires the use of Item No. 94030 to determine the lift increment characteristics of the representative section/flap combination from which to derive the section pitching moment coefficient increment. By this means the effects of spanwise variation are averaged out. Empirical corrections allow for the effects of wing planform geometry and the spanwise extent of the flaps.

3.1 Aerofoil Pitching Moment Coefficient Increment $\Delta C_{mt\alpha 0}$

The increment in pitching moment coefficient at zero angle of attack, due to deployment of a single-slotted flap on an aerofoil, is obtained from

$$\Delta C_{mt\alpha 0} = -\Delta C'_{L0t}\, h'_2\, (c'/c)^2 - \Delta C'_{L0t}\, (c'/c)(c'/c-1)/4 - C_{L0}(c'/c-1)/4 + C_{m\alpha 0}(c'/c-1)\ . \quad (3.1)$$

Here, $\Delta C'_{L0t}$ is the increment in lift coefficient at zero angle of attack due to deployment of a single-slotted flap on an aerofoil, based on the extended chord, and is evaluated as

$$\Delta C'_{L0t} \;=\; J_{t1} \Delta C'_{L1}\, (a_1)_0 / 2\pi\ , \quad (3.2)$$

where J_{t1} and $\Delta C'_{L1}$ are given in Figures 1 and 2 respectively, reproduced from Item No. 94030, and $(a_1)_0$ is the basic aerofoil lift-curve slope in incompressible flow, which may be obtained from Item No. Wings 01.01.05 (Derivation 1), and $\delta_{t1}°$ is the flap deflection angle in degrees.

The centre of the lift increment at zero angle of attack , h'_2 , is expressed as a fraction of the extended chord and measured positive aft from the quarter-chord point of the extended aerofoil chord. It is derived empirically from its theoretical value in Derivation 26 for a hinged plate on a thin aerofoil and adjusted to allow for chord extension by replacing c_{t1}/c with c'_{t1}/c' to give

$$h'_{2T} = \frac{0.25[1-(2c'_{t1}/c'-1)^2]^{1/2}\,[1-(2c'_{t1}/c'-1)]}{\left\{\pi - \cos^{-1}(2c'_{t1}/c'-1) + [1-(2c'_{t1}/c'-1)^2]^{1/2}\right\}} \,. \tag{3.3}$$

Values of h'_{2T} determined from Equation (3.3) are given in Figure 3 as a function of c'_{t1}/c' .

An incremental adjustment to obtain correlation with experimental data gives

$$h'_2 = h'_{2T} - 4(z_{um}/c)^{1.5}(x_{ts}/c-1) \,, \tag{3.4}$$

where x_{ts}/c is the chordwise location of the flap shroud trailing edge as a fraction of the basic aerofoil chord and z_{um}/c is the ratio of the maximum upper-surface ordinate to the aerofoil chord, see Sketch 1.2.

The final terms in Equation (3.1) involving C_{L0} and $C_{m\alpha0}$, the lift and pitching moment coefficients at zero angle of attack for the basic aerofoil, provide an approximation to the effect of extension of the aerofoil without flap angular rotation. (Note that the term involving $C_{m\alpha0}$ is always small compared to the first two terms in Equation (3.1) and it is sufficient to use C_{m0} in place of $C_{m\alpha0}$, see Section 7.1.) The method must not be used to obtain pitching moment increments due to extension without rotation, since those values are critically dependent on detailed geometry not accounted for in this method. As shown in Tables 5.1 and 5.2, the minimum validated flap deflection angle is $\delta_{t1}° = 10°$.

The centre of incremental lift at zero angle of attack due to flap deployment, expressed as a fraction of the *basic* aerofoil chord and referred to its quarter-chord position, is obtained from

$$h_2 = \frac{-\Delta C_{mt\alpha0}}{\Delta C'_{L0t}\,(c'/c)} \,. \tag{3.5}$$

3.2 Wing Pitching Moment Coefficient Increment $\Delta C_{mtw\alpha0}$

For a wing at zero angle of attack the increment in pitching moment coefficient due to single-slotted flap deployment is

$$\Delta C_{mtw\alpha0} = K_f(K_o - K_i)\Delta C_{mt\alpha0} + K_{f\Lambda}(K_{\Lambda o} - K_{\Lambda i})(A/2)\Delta C'_{L0t}\,(c'/c)\tan\Lambda_{1/4} \,, \tag{3.6}$$

where A is the wing aspect ratio, $\Lambda_{1/4}$ is the wing quarter-chord sweep angle, and $\Delta C'_{L0t}$ and $\Delta C_{mt\alpha0}$ are now calculated from Equations (3.2) and (3.1), respectively, for the representative streamwise section of the wing, taken at flap mid-span.

The part-span factors K_i and K_o are obtained from Figure 4 as functions of wing taper ratio and the inboard and outboard limits of the flap, $\eta = \eta_i$ and η_o, respectively.

The flap-type correlation factors for single-slotted flaps have been derived from the data of Derivations 6, 13, 17, 19, 20 and 22 to 25 to be

$$K_f = 1.0 \tag{3.7}$$

and $$K_{f\Lambda} = 1.0 . \tag{3.8}$$

The part-span wing-sweep factors, $K_{\Lambda i}$ and $K_{\Lambda o}$, are obtained for single-slotted flaps from Figures 5a to 5f as functions of the extended chord ratio, c'/c, and the inboard and outboard limits of the flap, $\eta = \eta_i$ and η_o respectively, for a range of values of wing taper ratio. Note that for all cases with a full-span flap or an unswept quarter-chord line the second term in Equation (3.6) has a value of zero.

The data for K_Λ given in Figures 5a to 5f for $\lambda = 0$, 0.1, 0.2, 0.4, 0.6 and 1 were obtained from Derivation 27 for extended flaps in the form

$$K_\Lambda = \frac{-3(1+\lambda)}{4(1+\lambda+\lambda^2)}\left\{\left[0.5\eta^2 - 0.333(1-\lambda)\eta^3\right]\left[(c'/c)(1-p)+p\right] - \left[0.5 - 0.333(1-\lambda)\right]p\right\}, \tag{3.9}$$

where λ is the wing taper ratio

and $$p = \frac{(c'/c)[\eta - 0.5(1-\lambda)\eta^2]}{0.5(1+\lambda) - [\eta - 0.5(1-\lambda)\eta^2](1-c'/c)} . \tag{3.10}$$

4. EFFECTS OF MACH NUMBER AND REYNOLDS NUMBER

4.1 Mach Number Effects

High local Mach numbers will occur at low free-stream Mach number as a result of high angle deployment of slotted flaps. Significant Mach number effects will occur at free-stream Mach numbers greater than about 0.2, at large values of $\delta_{t1}°$, and at progressively smaller values of this angle as Mach number is increased. None of the data considered for this Item was for a Mach number greater than 0.25.

4.2 Reynolds Number Effects

Values of $\Delta C_{mt\alpha 0}$ or $\Delta C_{mtw\alpha 0}$ are derived with an aerofoil lift-curve slope, a_{10} (obtained from Item No. Wings 01.01.05), which is dependent on Reynolds number. For the data used in the derivation of this Item no additional effect of Reynolds number on $\Delta C_{mt\alpha 0}$ or $\Delta C_{mtw\alpha 0}$ was found over the ranges of Reynolds number shown in Tables 5.1 and 5.2.

5. APPLICABILITY AND ACCURACY

5.1 Applicability

5.1.1 Aerofoils

The method given in this Item for estimating the position of the centre of the lift increment and the increment in pitching moment coefficient at zero angle of attack due to deployment of a trailing-edge single-slotted flap applies only to aerofoils without the deployment of a leading-edge device.

Table 5.1 summarises the parameter ranges covered by the experimental data, obtained from Derivations 3, 4, 5, 7, 8, 11, 12, 14, 15 and 21, correlated by Equation (3.5).

TABLE 5.1 Parameter ranges for test data for trailing-edge single-slotted flaps on aerofoils used in the method of Section 3.1

Parameter	*Range*
t/c	0.10 to 0.30
z_{um}/c	0.060 to 0.165
c_{t1}/c	0.25 to 0.40
c'/c	1.04 to 1.32
$\Delta c_{t1}/c$	–0.054 to 0.037
x_{ts}/c	0.72 to 1
$\delta_{t1}°$	10° to 60°
$R_c \times 10^{-6}$	1.0 to 9.0
M	$\le$ 0.24

5.1.2 Wings

The method given in this Item for estimating the increment in pitching moment coefficient, at zero angle of attack, due to deployment of a trailing-edge single-slotted flap on a wing, has been shown to be applicable to straight-tapered wings covering a wide range of planform parameters. Table 5.2 summarises the parameter ranges covered by the experimental data that were obtained from Derivations 6, 9, 10, 13, 16 to 20, 22 to 25 and used in the development of the method.

For a wing where c_{t1}/c is not constant, the flap should be divided into several spanwise portions, a calculation made separately for each portion, using its mid-span geometry, and the results summed to provide a total value of $\Delta C_{mtw\alpha 0}$. The number of portions required will depend on how rapidly the ratio c_{t1}/c varies across the span.

No wings with cranked leading or trailing edges or curved tips were included in the analysis. It is suggested that for such wings the planform parameters λ and $\Lambda_{1/4}$ be calculated for the equivalent straight-tapered planform as defined in Item No. 76003 (Reference 30). Care should be taken with the definition of c_{t1}/c and the user of the final result should be aware of the non-validated use of the method for such wings.

The method has only been validated for wings without leading-edge devices.

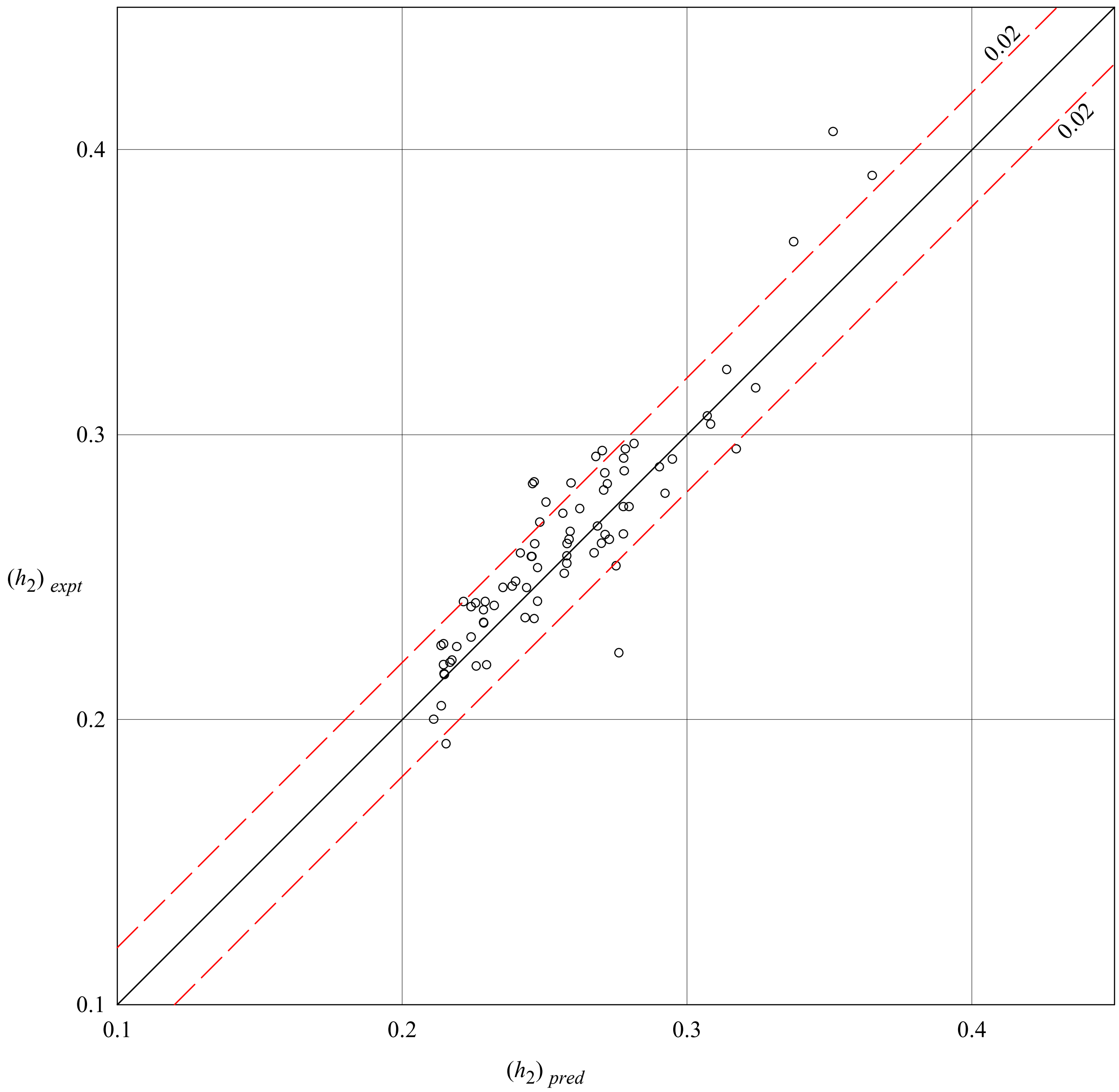
TABLE 5.2 Parameter ranges for test data for trailing-edge single-slotted flaps on wings used in the method of Section 3.2

Parameter	*Range*		
A	3.7	to	9.0
$A\tan\Lambda_0$	0	to	5.5
$A\tan\Lambda_{1/2}$	–0.4	to	4.7
Λ_0	0	to	47º
Λ_1	–12º	to	37º
λ	0.2	to	1.0
c_{t1}/c	0.2	to	0.50
c'/c	1.0	to	1.42
$\Delta c_{t1}/c$	–0.088	to	0.014
x_{ts}/c	0.72	to	1
$\delta_{t1}{}^\circ$	10º	to	64º
η_i	0	to	0.80
η_o	0.20	to	1
$R_{\bar{\bar{c}}} \times 10^{-6}$	0.61	to	7.0
M		$\leq$	0.25

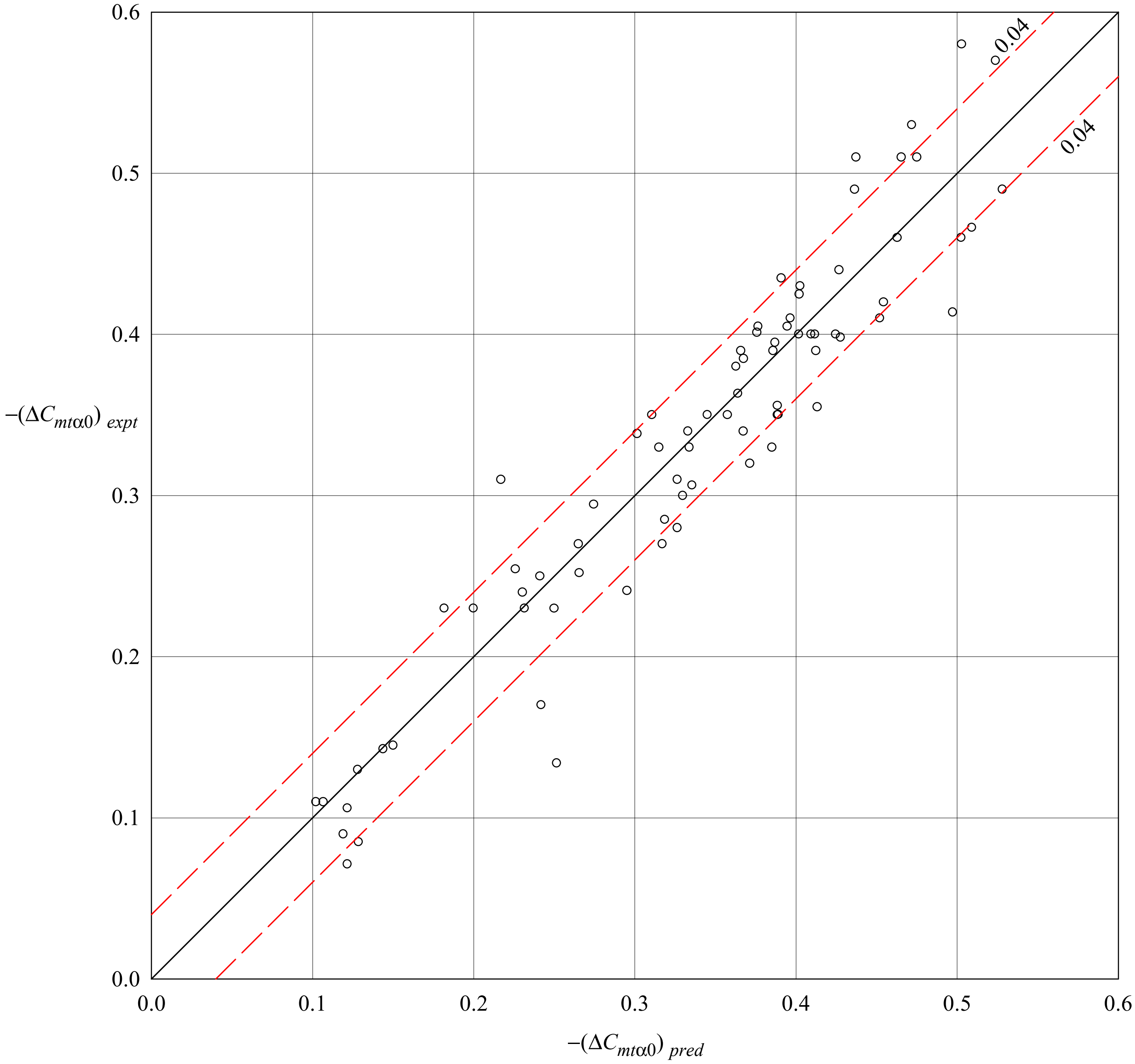
5.2 Accuracy

5.2.1 Aerofoils

Sketch 5.1 shows the comparison between predicted and experimental values of h_2, the centre of the lift increment due to deployment of single-slotted flaps on an aerofoil, for data from Derivations 3, 4, 5, 7, 8, 11, 12, 14, 15 and 21; the rms error is 0.016 and 83% of the data are correlated to within ± 0.02. Sketch 5.2 shows the corresponding comparison between predicted and experimental values of pitching moment coefficient increments, where the rms error is 0.038 and 73% of the data are correlated to within ± 0.04.



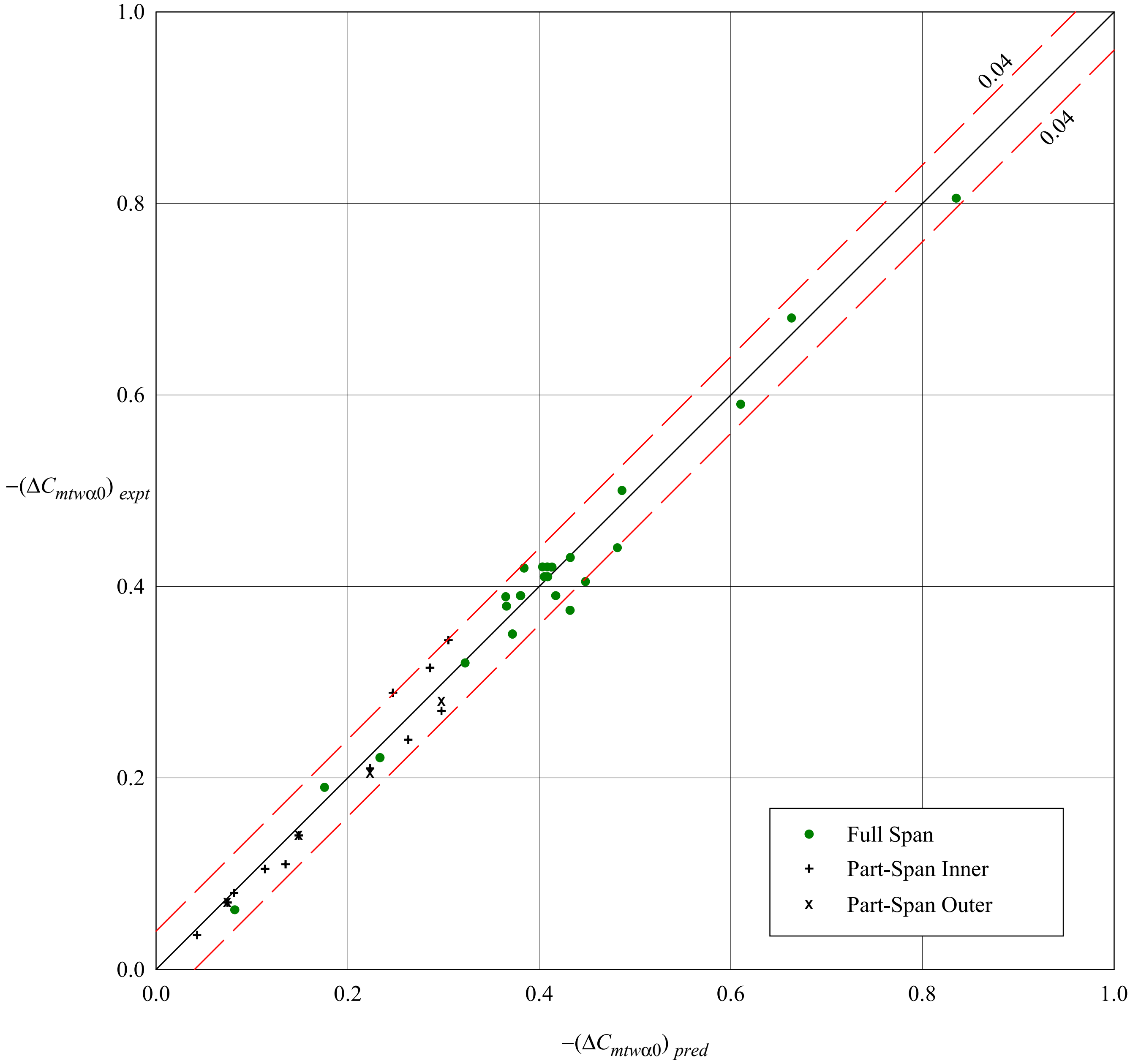
Sketch 5.1 Comparison of predicted and experimental values of h_2 for deployment of single-slotted flaps



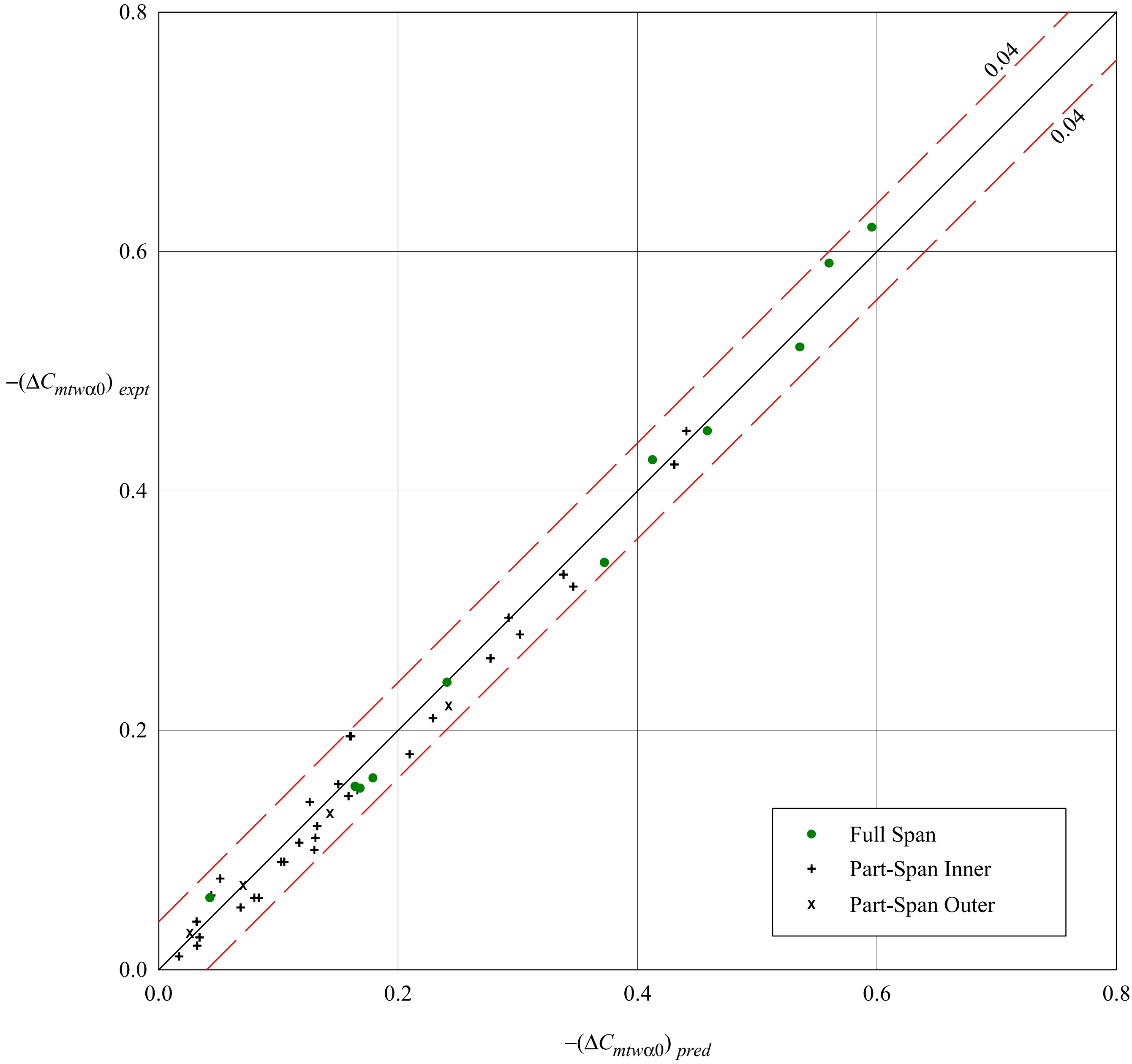
Sketch 5.2 Comparison of predicted and experimental values of $\Delta C_{mt\alpha0}$ for deployment of single-slotted flaps on aerofoils

5.2.2 Wings

The comparison between predicted and experimental values of the pitching moment coefficient increment, $\Delta C_{mtw\alpha0}$, due to deployment of both full-span and part-span single-slotted flaps is shown on Sketch 5.3 for unswept wings and on Sketch 5.4 for swept wings, for data from Derivations 6, 9, 10, 13, 16 to 20, 22 to 25. In the two sketches 98% of the data are within ± 0.04 and the rms error is 0.02.



Sketch 5.3 Comparison of predicted and experimental values of $\Delta C_{mtw\alpha0}$ for deployment of single-slotted flaps on unswept wings



Sketch 5.4 Comparison of predicted and experimental values of $\Delta C_{mtw\alpha 0}$ for deployment of single-slotted flaps on swept wings

6. DERIVATION AND REFERENCES

6.1 Derivation

The Derivation lists selected sources of information that have been used in the preparation of this Item.

6.1.1 ESDU Data Items

1.	ESDU	Slope of lift curve for two-dimensional flow. ESDU International, Item No. Wings 01.01.05, 1955.
2.	ESDU	Increments in aerofoil lift coefficient at zero angle of attack and in maximum lift coefficient due to deployment of a single-slotted trailing-edge flap, with or without a leading-edge high-lift device, at low speeds. ESDU International, Item No. 94030, 1995.

6.1.2 Wind-tunnel test reports

3.	WENZINGER, C.J. HARRIS, T.A.	Wind-tunnel investigation of an NACA 23012 airfoil with various arrangements of slotted flaps. NACA Report 664, 1938.
4.	WENZINGER, C.J. GAUVAIN, W.E.	Wind-tunnel investigation of an NACA airfoil with a slotted flap and three types of auxiliary flap. NACA Report 679, 1939.
5.	WENZINGER, C.J. HARRIS, T.A.	Wind-tunnel investigation of an NACA 23021 airfoil with various arrangements of slotted flaps. NACA Report 677, 1939.
6.	HOUSE, R.O.	The effects of partial span slotted flaps on the aerodynamic characteristics of a rectangular and tapered NACA 23012 wing. NACA tech. Note 719, 1939.
7.	RECANT, I.G.	Wind-tunnel investigation of an NACA 23030 airfoil with various arrangements of slotted flaps. NACA tech. Note 755, 1940.
8.	LOWRY, J.G.	Wind-tunnel investigation of an NACA 23012 airfoil with several arrangements of slotted flaps with extended lips. NACA tech. Note 808, 1941.
9.	BLACKBURN AIRCRAFT	Wind-tunnel tests on moderately large chord flaps with single and multiple slots. Blackburn Aircraft Limited, Report W.T. 85/42, 1943.
10.	BLACKBURN AIRCRAFT	Preliminary experiments on a combination of large chord slotted and split flaps. Blackburn Aircraft limited, Report W.T. 86/43, 1943.
11.	HOLTZCLAW, R.S.	Wind-tunnel investigation of the effects of spoilers on the characteristics of a low drag airfoil equipped with a 0.25-chord slotted flap. NACA MR A5G23 (TIL 1625), 1945.

12.	LOFTIN, L.K. RICE, F.K.	Two-dimensional wind-tunnel investigation of two NACA low drag airfoil sections equipped with slotted flaps. NACA WR L5L11 (TIL 1453), 1946.
13.	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 Report 942, 1947.
14.	CAHILL, J.F.	Two-dimensional wind-tunnel investigation of four types of high-lift flap on a NACA 65-210 airfoil section NACA tech. Note 1191, 1947.
15.	RACISZ, S.F.	Investigation of NACA 65(112)A111 (approx.) airfoil with 0.35-chord slotted flap at Reynolds numbers up to 25 million. NACA tech. Note 1463, 1947.
16.	FISCHEL, J. SCHNEITER, L.E.	High speed wind tunnel investigation of high lift and aileron control characteristics of a NACA 65-210 semi-span wing. NACA tech. Note 1473, 1947.
17.	SCHNEITER, L.E. VOGLER, R.D.	Wind-tunnel investigation at low speeds of various plug aileron and lift flap configurations on a 42° sweptback semi-span wing. NACA RM L8K19 (TIL 2058), 1948.
18.	FISCHEL, J. VOGLER, R.D.	High lift and lateral control characteristics of a NACA 65_2-215 semi-span wing equipped with plug and retractable ailerons and full span flap. NACA tech. Note 1872, 1949.
19.	SPOONER, S.H. MOLLENBERG, E.F.	Positioning investigation of single slotted flaps on a 47.7° sweptback wing at Reynolds numbers of 4.0×10^6. NACA RM L50H29(TIB 2507), 1950.
20.	NAESETH, R.L.	Low speed aerodynamic characteristics of a 45° sweptback wing with double slotted flaps. NACA RM L56A10 (TIL 5052), 1956.
21.	LJUNGSTROM, B.L.G.	Two-dimensional wind-tunnel experiments with single and double-slotted flaps. Aeronaut. Res. Inst., Sweden. FFA AU-1083, 1975.
22.	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 TR 76030, 1976.
23.	PAULSON, J.W.	Wind-tunnel investigation of a Fowler flap and spoiler for an advanced general aviation wing. NASA tech. Note D-8236, 1976.
24.	MORGAN, H.L. PAULSON, J.W.	Aerodynamic characteristics of wing-body configuration with two advanced general aviation airfoil sections and simple flap systems. NASA tech. Note D-8524, 1977.
25.	HAHNE, D.E. JORDAN, F.L.	Full-scale semi-span tests of a business-jet wing with a natural laminar flow aerofoil. NASA tech. Paper 3133, 1991.

6.1.3 Theory

26.	GLAUERT, H.	Theoretical relationships for an aerofoil with hinged flap. ARC R&M 1095, 1927.
27.	DENT, M.M. CURTIS, M.F.	A method of estimating the effect of flaps on pitching moment and lift on tailless aircraft. RAE Report No. Aero 1861, 1943.
28.	YOUNG, A.D.	The aerodynamic characteristics of flaps. ARC R&M 2622, (RAE Report No. Aero 2185), 1947.

6.1.4 References

The References are sources of information supplementary to that used in the derivation of this Item.

29.	ESDU	Aerodynamic characteristics of aerofoils in compressible inviscid airflow at subcritical Mach numbers. ESDU International, Item No. 72024, 1972.
30.	ESDU	Geometrical properties of cranked and straight tapered wing planforms. ESDU International, Item No. 76003, 1976.
31.	ESDU	Wing pitching moment at zero lift at subcritical Mach numbers. ESDU International, Item No. 87001, 1987.
32.	ESDU	Slope of aerofoil lift curve for subsonic two-dimensional flow. ESDU International, Item No. 97020, 1997.
33.	ESDU	Aerofoil incidence for zero lift in subsonic two-dimensional flow. ESDU International, Item No. 98011, 1998.

7. EXAMPLES

7.1 Example 1: Pitching Moment Increment due to a Trailing-edge Single-slotted Flap on an Aerofoil

Estimate the increment in pitching moment coefficient at zero angle of attack due to the deployment of a trailing-edge single-slotted flap installed on a modified NACA 65_2-215 section as shown in Sketch 7.1. The modifications produced a linear profile rearwards from 75% chord and 65% chord on the upper and lower surfaces, respectively.

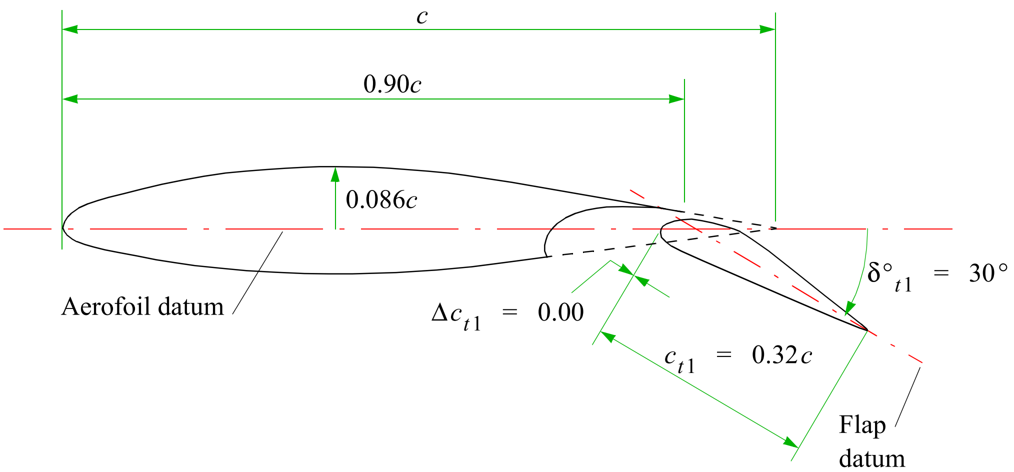
The required geometrical parameters are as follows.

Aerofoil	*Flap*		
$z_{um}/c = 0.086$	$x_{ts}/c = 0.90$		
	$c_{t1}/c = 0.32$	$\delta_{t1}° = 30°$	$\Delta c_{t1}/c = 0.00$

The flow conditions are $M = 0.2$ and $R_c = 4.5 \times 10^6$, both of which are within the ranges of Table 5.1.

The inviscid value, C_{m0i}, of the pitching moment coefficient for aerofoil zero lift, which may be calculated by the method in Item No. 72024 (Reference 29) is taken as –0.031 for $M = 0.2$.

The angle of attack for zero lift, which may be calculated by the method in Item No. 98011 (Reference 33), is taken as –1.004° for the given flow conditions.



Sketch 7.1 Flap Geometry

(1) Obtain C_{L0} and $C_{m\alpha 0}$

For the modified NACA 65_2–215 section, from Item No. W.01.01.05 (Derivation 1) for boundary-layer transition at the leading edge

$$(a_1)_0 = 5.62 \text{ rad}^{-1} .$$

(Although a more accurate estimate of the lift-curve slope is available by the method of Item No. 97020, the value from Item No Wings 01.01.05 is used because the method of that Item was employed in the original correlation of flap lift coefficient increments.)

The angle of attack for zero lift is given as –1.004°

so that $C_{L0} = -5.62 \times (-1.004 / 57.3) = 0.10$.

As remarked in Section 3.1, the term in Equation (3.1) involving $C_{m\alpha 0}$ is always small compared with the first two terms so that it is acceptable to assume that for the basic aerofoil the pitching moment coefficient at zero angle of attack is equal to the pitching moment coefficient at zero lift, *i.e.* $C_{m\alpha 0} = C_{m0}$. This assumption is exact for all cases in which the aerodynamic centre is at the quarter-chord point.

The inviscid value of C_{m0} at $M = 0.2$ is given as $C_{m0i} = -0.031$. Figure 1 of Item No. 87001 (Reference 31) provides a viscous correction factor *F*, which can be approximated by the equation

$$F = 1 - 0.29\left[\sin\left(\frac{-C_{m0i}}{0.29}\,\frac{\pi}{2}\right)\right]^{0.7}, \text{ where the angle is measured in radians,} \quad (7.1)$$

so that

$$F = 1 - 0.29\left[\sin\left(\frac{0.031}{0.29}\,\frac{\pi}{2}\right)\right]^{0.7}$$

$$= 0.917 ,$$

and hence, through the assumption noted above,

$$C_{m\alpha 0} = C_{m0} = F\,C_{m0i} = 0.917 \times (-0.031)$$

$$= -0.028 .$$

(2) Obtain c'/c and c'_{t1}/c'

From the definitions of Sketch 1.2, the dimensions of Sketch 7.1 give

$$c'_{t1}/c = c_{t1}/c + \Delta c_{t1}/c$$

$$= 0.32 + 0.00$$

$$= 0.32$$

and

$$c'/c = x_{ts}/c + c'_{t1}/c$$

$$= 0.90 + 0.32$$

$$= 1.22$$

so that

$$\begin{aligned} c'_{t1}/c' &= (c'_{t1}/c)/(c'/c) \\ &= 0.32/1.22 \\ &= 0.262\ . \end{aligned}$$

(3) Determine $\Delta C'_{L0t}$

From Equation (3.2)

$$\Delta C'_{L0t} = J_{t1}\Delta C'_{L1}\,(a_1)_0/2\pi$$

in which, from

Figure 1 with $\delta_{t1}° = 30°$,

$$J_{t1} = 1.17$$

and, from Figure 2 with $\delta_{t1}° = 30°$ and $c'_{t1}/c' = 0.262$,

$$\Delta C'_{L1} = 1.26\ .$$

Therefore,

$$\begin{aligned} \Delta C'_{L0t} &= 1.17 \times 1.26 \times 5.62/(2\pi) \\ &= 1.319\ . \end{aligned}$$

(4) Determine h'_2

From Equation (3.3), with the inverse cosine evaluated in radians,

$$\begin{aligned} h'_{2T} &= \frac{0.25[1-(2c'_{t1}/c'-1)^2]^{1/2}[1-(2c'_{t1}/c'-1)]}{\left\{\pi - \cos^{-1}(2c'_{t1}/c'-1) + [1-(2c'_{t1}/c'-1)^2]^{1/2}\right\}} \\ &= \frac{0.25[1-(2\times 0.262-1)^2]^{1/2}[1-(2\times 0.262-1)]}{\left\{\pi - \cos^{-1}(2\times 0.262-1) + [1-(2\times 0.262-1)^2]^{1/2}\right\}} \\ &= 0.1661, \end{aligned}$$

Alternatively, the value of h'_{2T} could be read from Figure 3.

From Equation (3.4)

$$\begin{aligned} h'_2 &= h'_{2T} - 4(z_{um}/c)^{1.5}(x_{ts}/c - 1) \\ &= 0.1661 - 4 \times (0.086)^{1.5} \times (0.90 - 1) \\ &= 0.1762\ . \end{aligned}$$

(5) Determine $\Delta C_{mt\alpha 0}$

From Equation (3.1)

$$\begin{aligned} \Delta C_{mt\alpha 0} &= -\Delta C'_{L0t}\, h'_2\, (c'/c)^2 - \Delta C'_{L0t}\,(c'/c)(c'/c - 1)/4 \\ &\qquad - C_{L0}(c'/c - 1)/4 + C_{m\alpha 0}(c'/c - 1) \\ &= -1.319 \times 0.1762 \times 1.22^2 - 1.319 \times 1.22 \times (1.22 - 1)/4 \\ &\qquad - 0.10 \times (1.22 - 1)/4 - 0.028 \times (1.22 - 1) \\ &= -0.3459 - 0.0885 - 0.0055 - 0.0062 \\ &= -0.4461 \\ &\approx -\mathbf{0.446}\ . \end{aligned}$$

The centre of incremental lift expressed as a fraction of the basic aerofoil chord and referred to its quarter-chord position is obtained by evaluation of Equation (3.5) from which

$$\begin{aligned} h_2 &= \frac{-\Delta C_{mt\alpha 0}}{\Delta C_{L0t}(c'/c)} \\ &= \frac{0.446}{1.319 \times 1.22} \\ &= 0.277. \end{aligned}$$

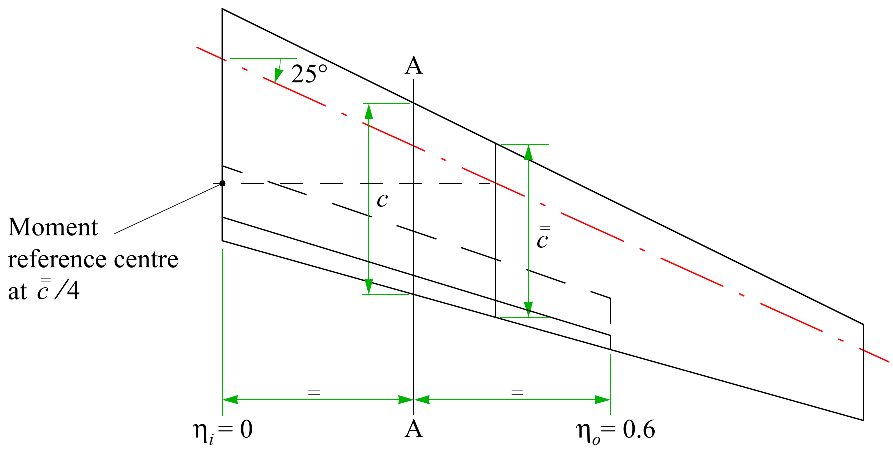
7.2 Example 2: Pitching Moment Increment due to a Trailing-edge Single-slotted Flap on a Wing

Estimate the increment in pitching moment coefficient at zero angle of attack for a Reynolds number $R_{\bar{\bar{c}}} = 4.5 \times 10^6$ and a free-stream Mach number $M = 0.2$ for a wing with a part-span trailing-edge single-slotted flap as shown in Sketch 7.2. The wing has the planform parameter values

$$A = 8,\ \Lambda_{1/4} = 25° \text{ and } \lambda = 0.4$$

and across the whole span the streamwise section is the modified NACA 65_2–215 profile which was used in Example 1.

The flap has the same streamwise geometrical parameters as those used in Example 7.1 and extends from the wing centre-line to 60% of the wing semi-span.



Sketch 7.2 Wing planform (flap undeployed)

The derived sweep angles $\Lambda_0 = 27.5°$ and $\Lambda_1 = 17°$ (see Item No. 76003, Reference 30), the parameter $A \tan\Lambda_0 = 4.16$, the Mach number and the Reynolds number all lie within the ranges shown on Table 5.2.

Determine $\Delta C_{mtw\alpha 0}$

In addition to the incremental coefficients

$$\Delta C_{mt\alpha 0} = -0.4461 .$$

and

$$\Delta C'_{L0t} = 1.319$$

for the aerofoil section from Example 1, various factors are needed to determine $\Delta C_{mt\alpha 0}$ from Equation (3.6)

The flap type correlation factors K_f and $K_{f\Lambda}$ are given by Equations (3.7) and (3.8) as

$$K_f = K_{f\Lambda} = 1.0 .$$

From Figure 4 for $\eta_i = 0$

$$K_i = 0$$

and for $\eta_o = 0.6$ and $\lambda = 0.4$

$$K_o = 0.788 .$$

From Figure 5d for $\lambda = 0.4$, $c'/c = 1.22$ and $\eta_i = 0$

$$K_{\Lambda i} = 0$$

and for $\eta_o = 0.6$

$$K_{\Lambda o} = 0.0526\,.$$

Therefore, from Equation (3.6)

$$\begin{aligned}\Delta C_{mtw\alpha 0} &= K_f(K_o - K_i)\Delta C_{mt\alpha 0} + K_{f\Lambda}(K_{\Lambda o} - K_{\Lambda i})(A/2)\Delta C'_{L0t}\,(c'/c)\tan\Lambda_{1/4} \\ &= 1.0 \times (0.788 - 0) \times (-0.4461) + 1.0 \times (0.0526 - 0) \times (8/2) \times 1.319 \times 1.22 \times \tan 25° \\ &= -\,0.3515 + 0.1579 \\ &= -\,0.1936 \\ &\approx \mathbf{-\,0.194}\,.\end{aligned}$$

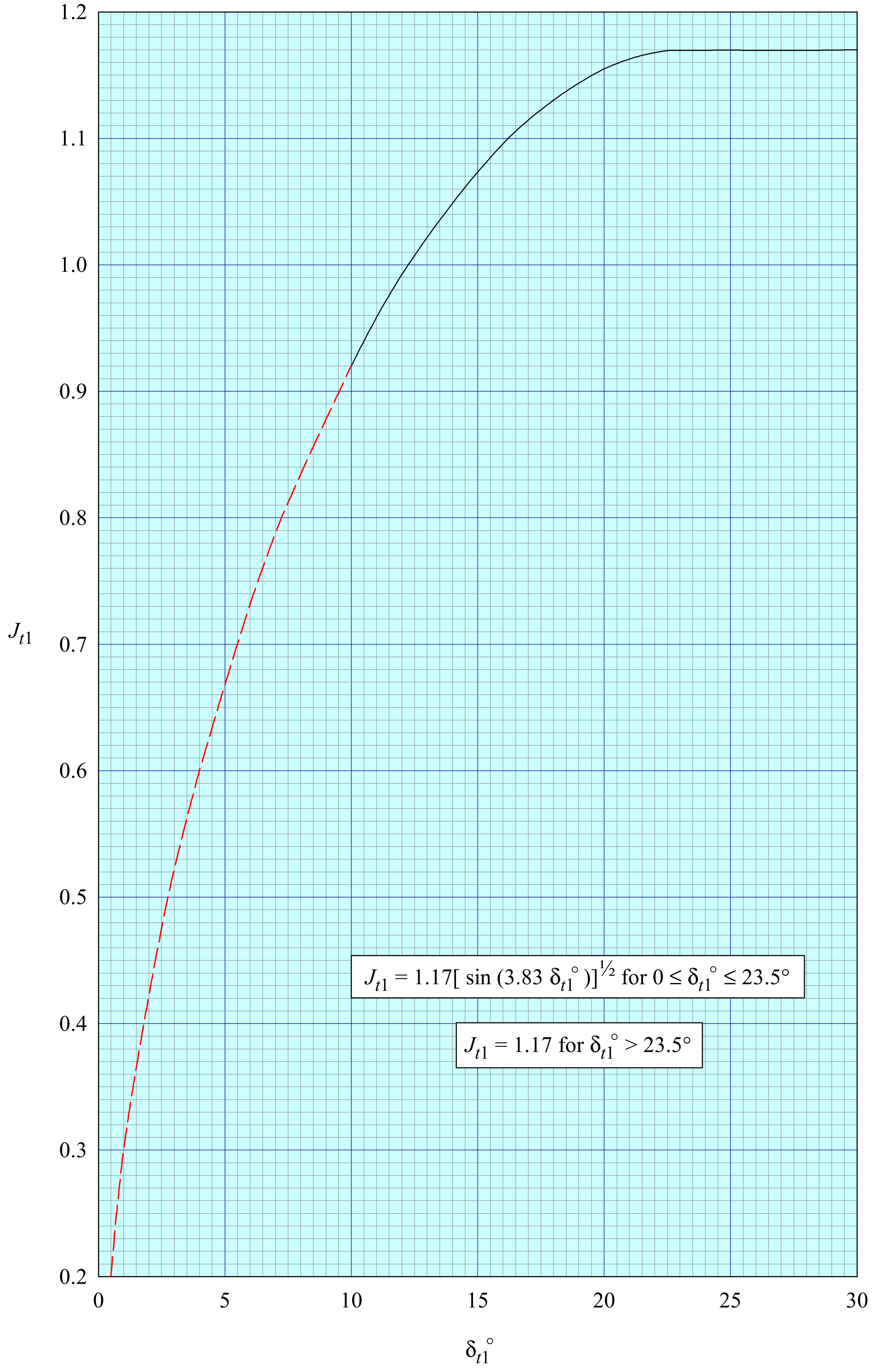


FIGURE 1 CORRELATION FACTOR FOR SINGLE-SLOTTED FLAP

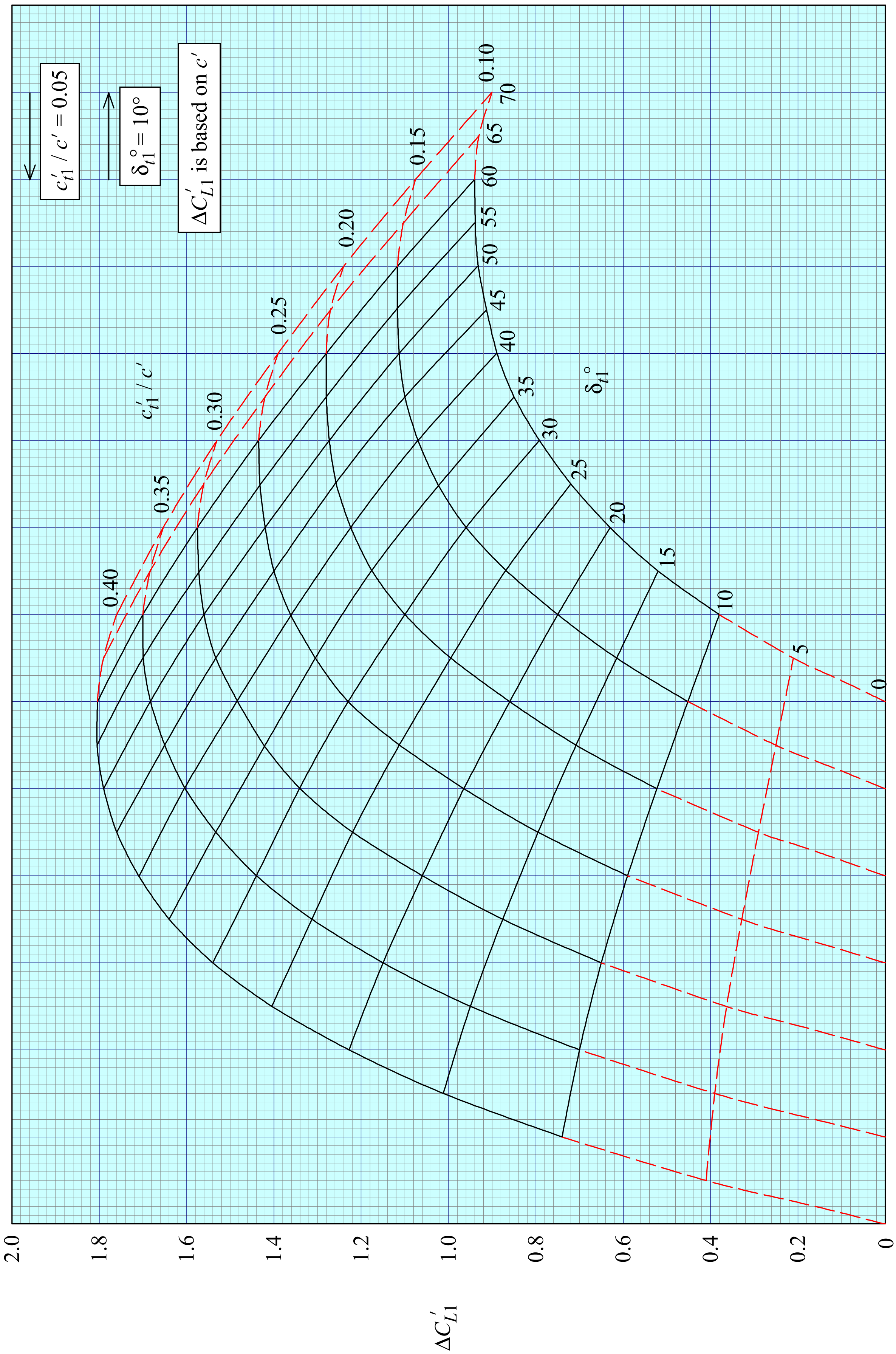


FIGURE 2 LIFT COEFFICIENT INCREMENT DUE TO SINGLE-SLOTTED FLAP DEPLOYMENT ON AEROFOIL

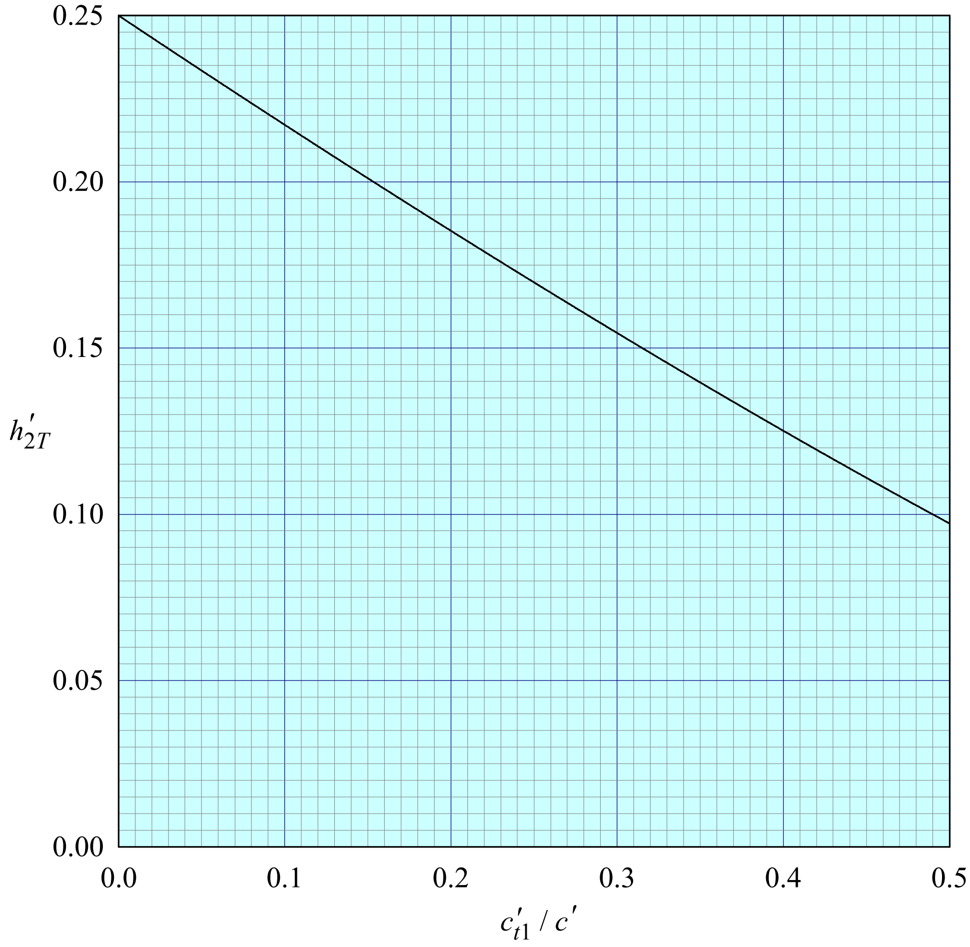


FIGURE 3 THEORETICAL VALUE OF CENTRE OF INCREMENTAL LIFT ON AEROFOIL DUE TO FLAP DEPLOYMENT

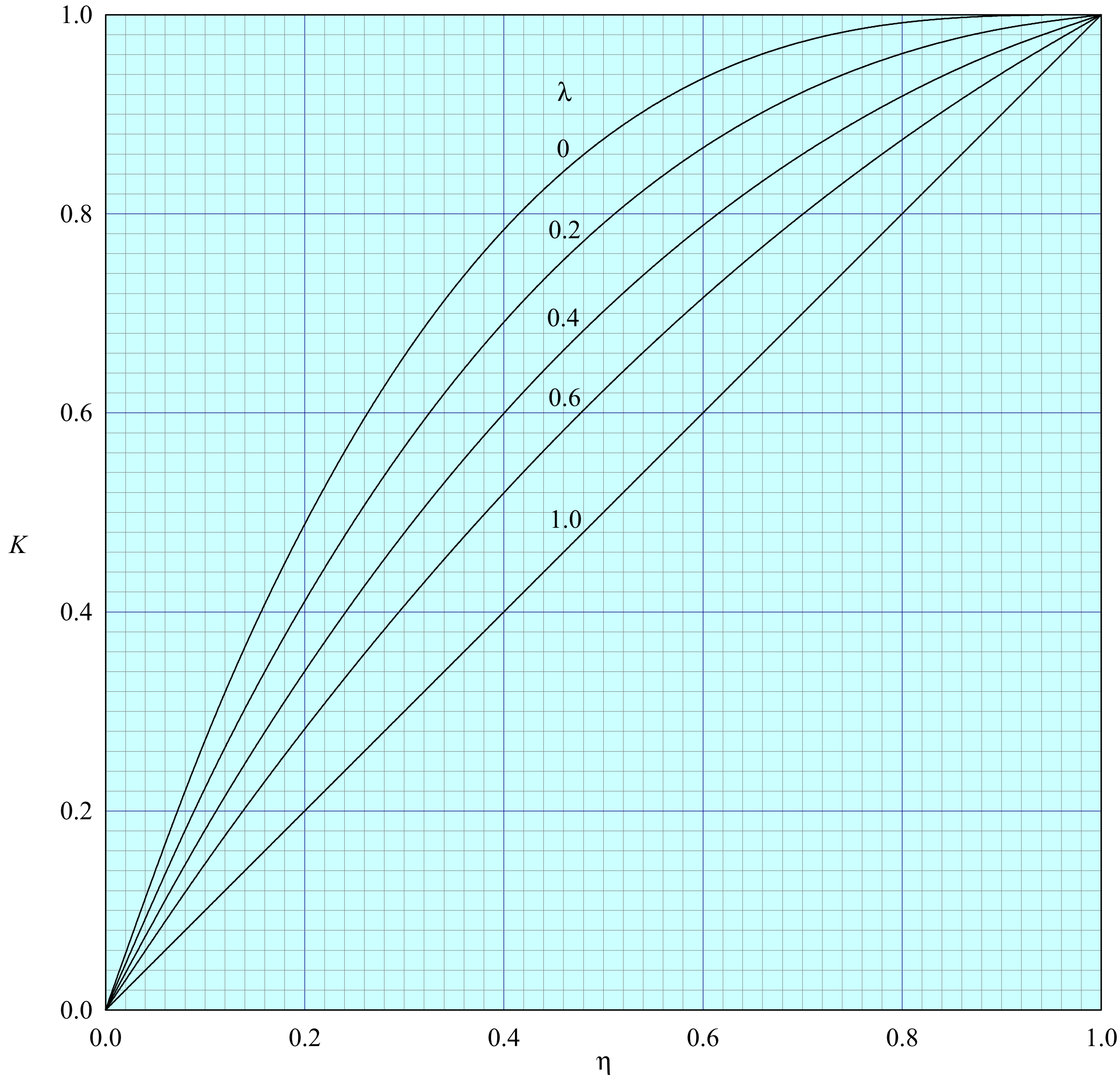


FIGURE 4 PART-SPAN FACTOR *K* FOR SINGLE-SLOTTED FLAPS

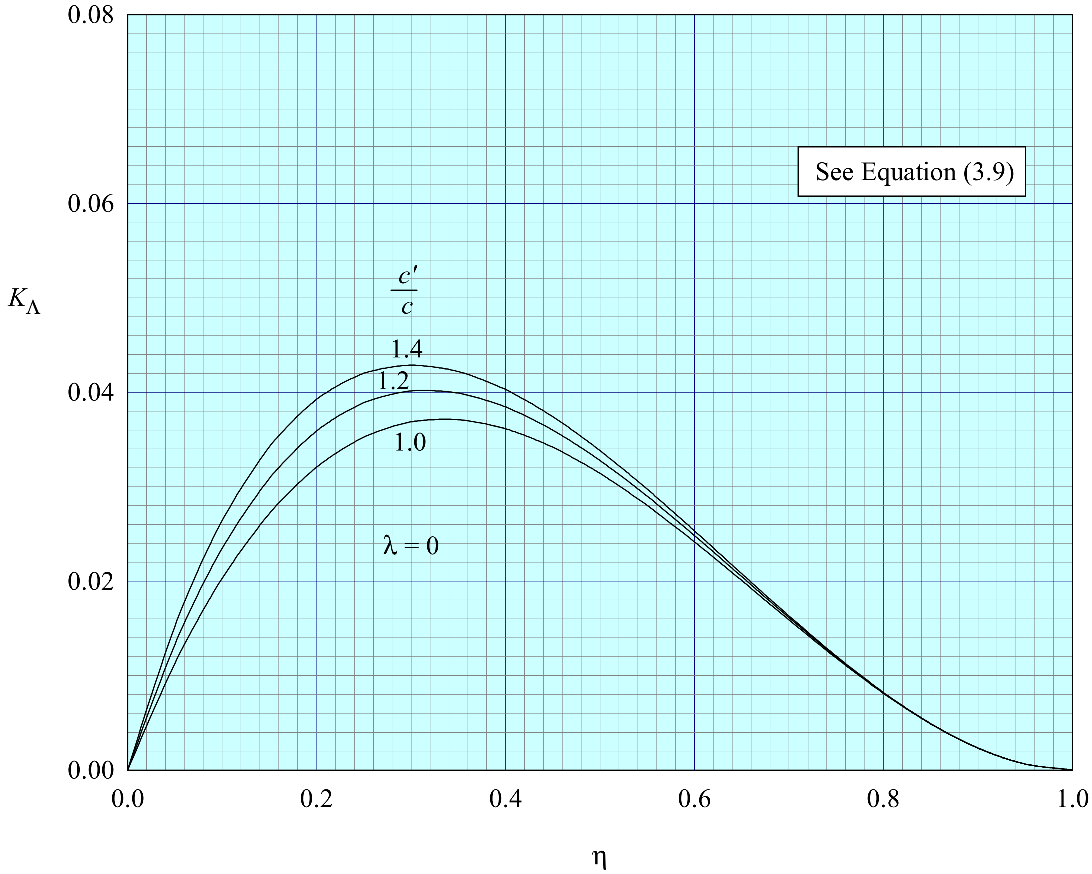


FIGURE 5a PART-SPAN FACTOR K_Λ FOR SINGLE-SLOTTED FLAPS

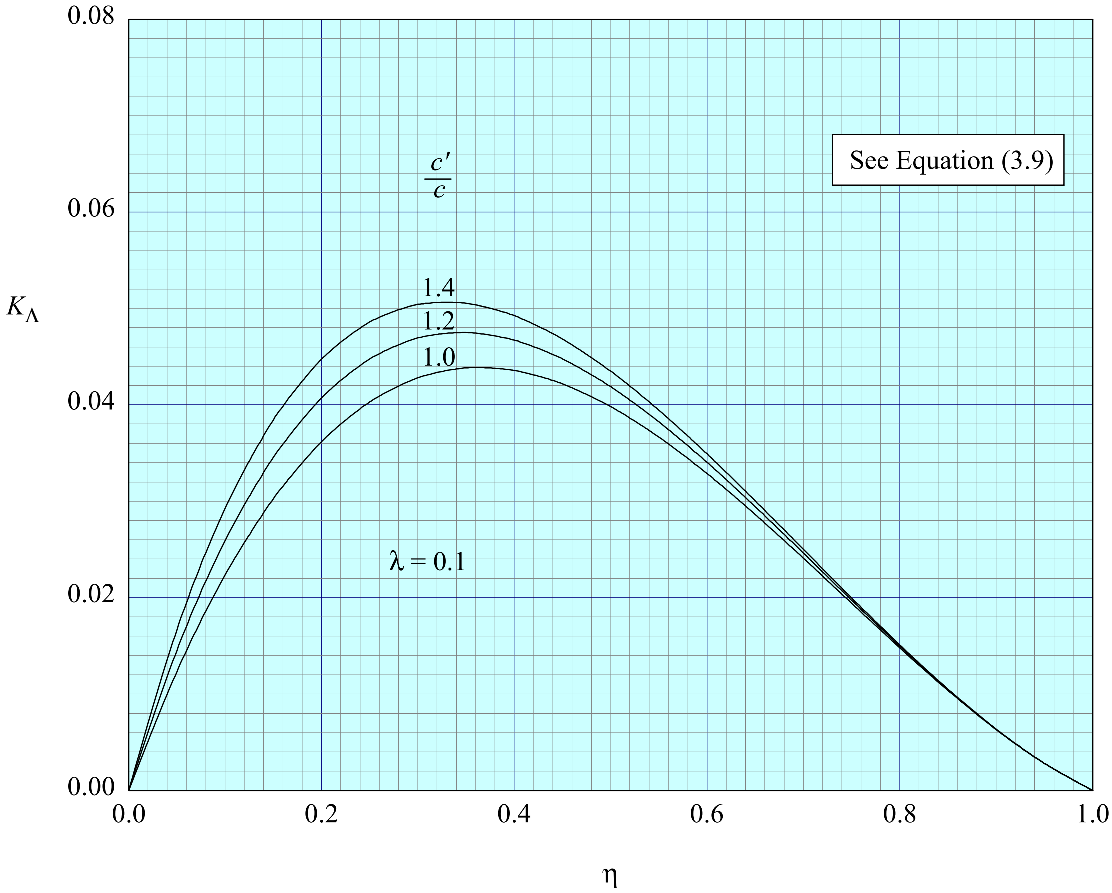


FIGURE 5b PART-SPAN FACTOR K_Λ FOR SINGLE-SLOTTED FLAPS

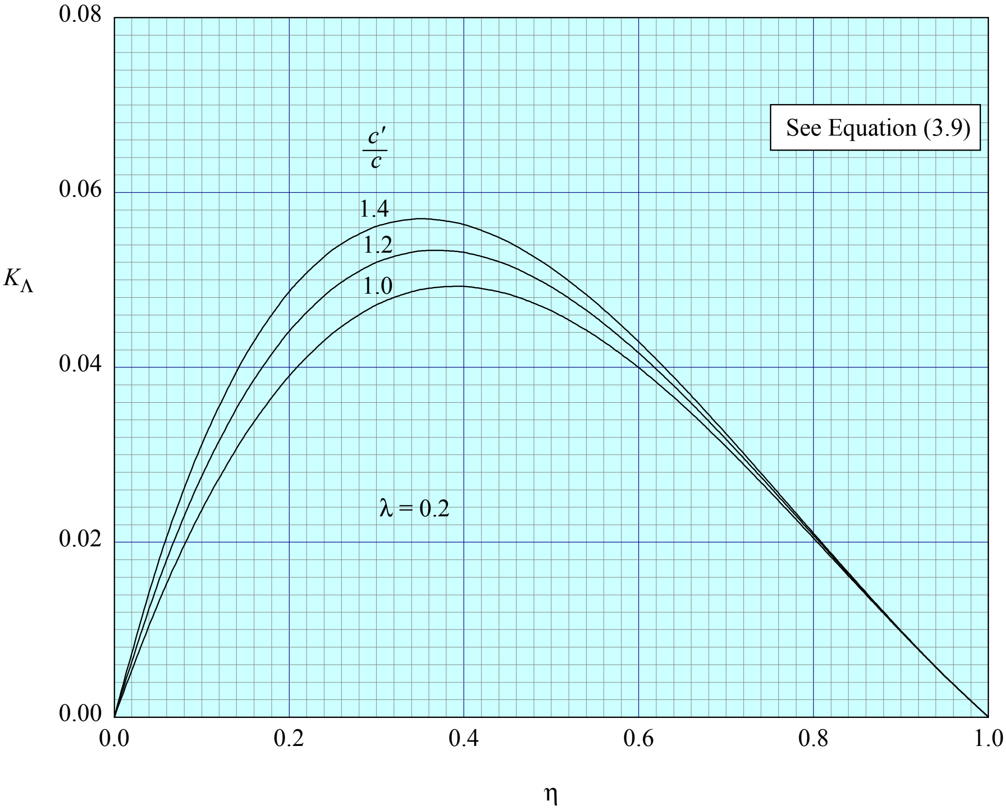


FIGURE 5c PART-SPAN FACTOR K_Λ FOR SINGLE-SLOTTED FLAPS

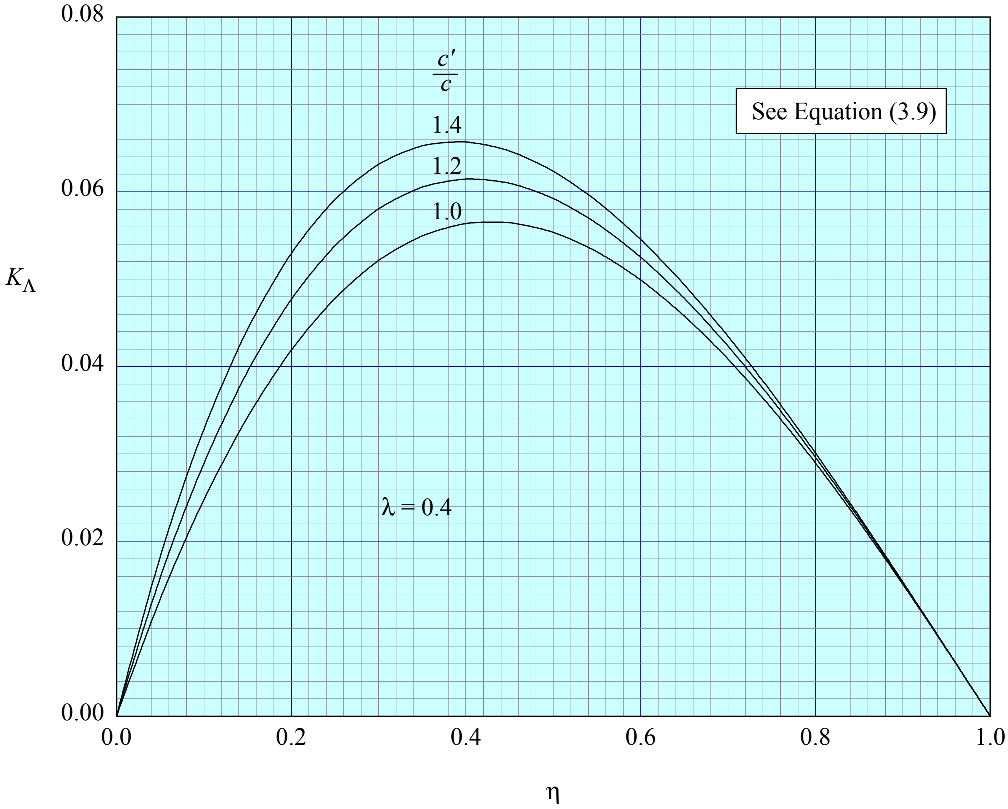


FIGURE 5d PART-SPAN FACTOR K_Λ FOR SINGLE-SLOTTED FLAPS

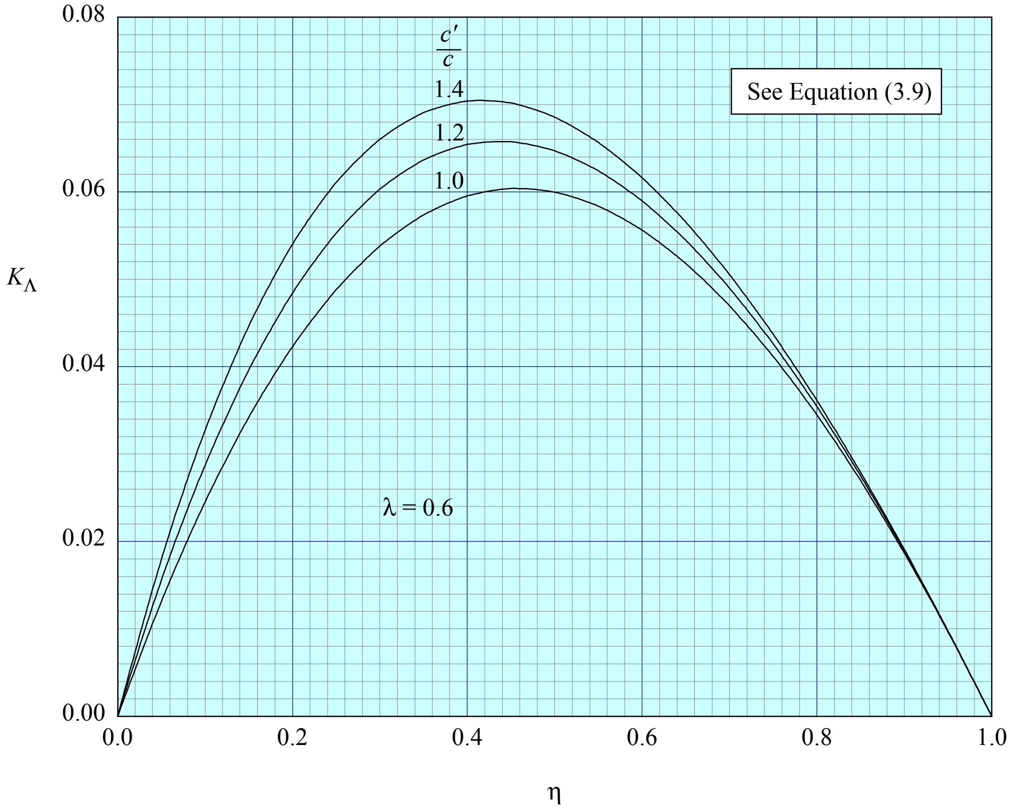


FIGURE 5e PART-SPAN FACTOR K_Λ FOR SINGLE-SLOTTED FLAPS

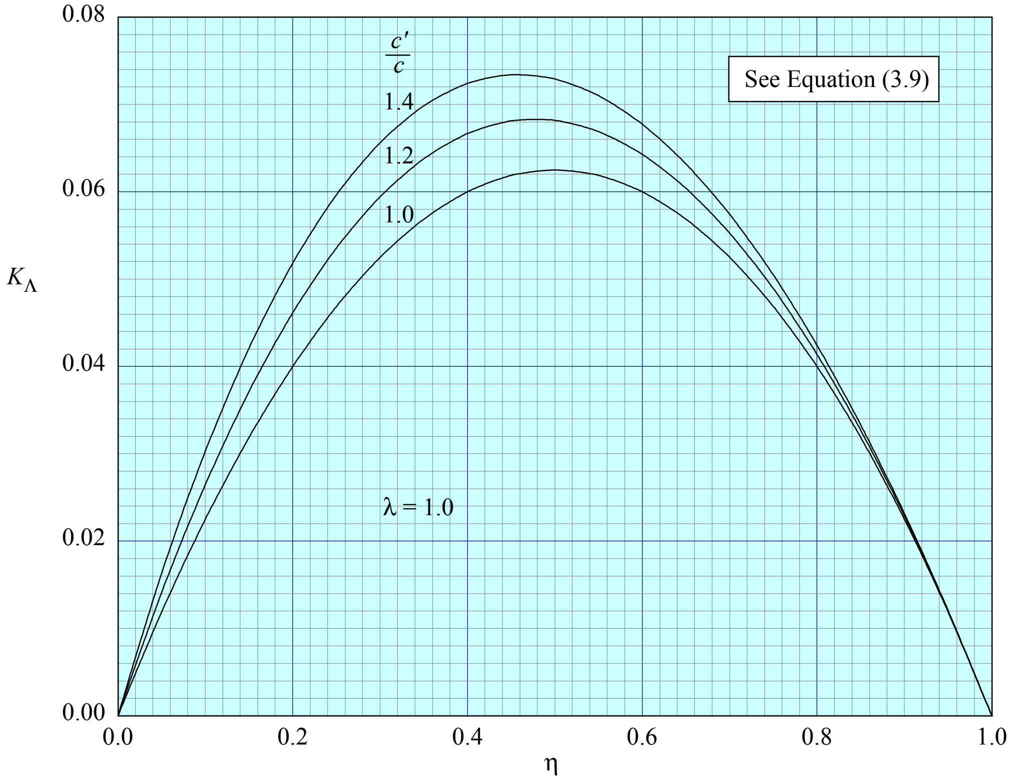


FIGURE 5f PART-SPAN FACTOR K_Λ FOR SINGLE-SLOTTED FLAPS

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THE PREPARATION OF THIS DATA ITEM

The work on this particular Data Item which supersedes, in part, Item Nos Aero F.08.01.01 and 02 was monitored and guided by the Aerodynamics Committee which first met in 1942 and now has the following membership:

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The technical work involved in the assessment of the available information and the development and subsequent construction of the Data Item method was carried out under contract to ESDU by Mr J.R.J. Dovey.

The person with overall responsibility for the work in this subject area is Mr R.W. Gilbey, Head of Aircraft Aerodynamics Group.