



ESDU 99031

Issued October 1999
With Amendments A and B
November 2009

Computer program for estimation of lift curve to maximum lift for wing-fuselage combinations with high-lift devices at low speeds

Associated software: ESDUpacs A9931, B9931

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THE PREPARATION OF THIS DATA ITEM

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Mr S.F. Wood	— BAE SYSTEMS, Woodford.

* Corresponding Member

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 by Mr R.W. Gilbey, Head of the Aircraft Aerodynamics Group.

The computer programs were prepared under contract to ESDU by Mr C. King.

COMPUTER PROGRAM FOR ESTIMATION OF LIFT CURVE TO MAXIMUM LIFT FOR WING-FUSELAGE COMBINATIONS WITH HIGH-LIFT DEVICES AT LOW SPEEDS

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COMPUTER PROGRAM FOR ESTIMATION OF LIFT CURVE TO MAXIMUM LIFT FOR WING-FUSELAGE COMBINATIONS WITH HIGH-LIFT DEVICES AT LOW SPEEDS

The program, ESDUpac A9931, has been written in ANSI FORTRAN 77 for use on machines using PC/MS DOS. An auxiliary program, ESDUpac B9931, is also provided.

SUMMARY

The construction of the input file for an ESDUpac that predicts the lift curve from the zero-lift angle of attack to the maximum lift coefficient is described. The program applies to aerofoils, wings and wing-fuselage combinations at low speeds with leading-edge and trailing-edge high-lift devices deployed. Sketches from the Data Items from which prediction methods have been taken are reproduced to aid the user in determining geometric parameters but the underlying methods are not repeated.

General information on the availability and use of Data Items on high-lift devices within the Aerodynamics Series is to be found in Item No. 97002²⁴.

Issued October 1999
With Amendments A and B, November 2009 – 112 pages
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1. NOTATION AND UNITS

The aerofoil or wing with no high-lift devices deployed is described in this Item as the *basic* aerofoil or *basic* wing.

		SI	British
A	aspect ratio		
a, b	construction lengths (see Sketch 1.6c)	m	ft
a_1	lift-curve slope of basic wing	rad ⁻¹	rad ⁻¹
$(a_1)_0$	lift-curve slope of basic aerofoil	rad ⁻¹	rad ⁻¹
$CALC$	input file entry to select calculation procedure (see Table 3.1 or Sketch 3.1)		
C_L	lift coefficient; $(lift)/qS$ for wing, $(lift)/qc$ for aerofoil		
C_{Lm}	maximum lift coefficient of aerofoil with high-lift devices deployed		
C_{LmB}	maximum lift coefficient of basic aerofoil (high-lift devices not deployed)		
ΔC_{Lm}	increment in aerofoil maximum lift coefficient due to deployment of high-lift devices		
ΔC_{Lml}	increment in aerofoil maximum lift coefficient due to deployment of leading-edge device		
ΔC_{Lmt}	increment in aerofoil maximum lift coefficient due to deployment of trailing-edge flap		
C_{LmaxB}	maximum lift coefficient of basic wing (high-lift devices not deployed)		
C_{LmaxBf}	maximum lift coefficient of basic wing-fuselage combination (high-lift devices not deployed)		
C_{Lmaxw}	maximum lift coefficient of wing with high-lift devices deployed		
C_{Lmaxwf}	maximum lift coefficient of wing-fuselage combination with high-lift devices deployed		
ΔC_{Lmax}	increment in wing maximum lift coefficient due to deployment of high-lift devices		
ΔC_{Lmaxf}	increment in maximum lift coefficient due to fuselage		
ΔC_{Lmaxl}	increment in wing maximum lift coefficient due to deployment of leading-edge devices		

ΔC_{Lmax}	increment in wing maximum lift coefficient due to deployment of trailing-edge flaps		
C_{L0}	aerofoil lift coefficient at zero angle of attack with high-lift devices deployed		
ΔC_{L0}	increment in aerofoil lift coefficient at zero angle of attack due to deployment of high-lift devices		
ΔC_{L0l}	increment in aerofoil lift coefficient at zero angle of attack due to deployment of leading-edge device		
ΔC_{L0t}	increment in aerofoil lift coefficient at zero angle of attack due to deployment of trailing-edge flap		
C_{L0B}	lift coefficient at zero angle of attack for basic aerofoil (high-lift devices not deployed)		
C_{L0Bf}	lift coefficient at zero angle of attack for basic wing-fuselage combination (high-lift devices not deployed)		
C_{L0Bw}	lift-coefficient at zero angle of attack for basic wing (high-lift devices not deployed)		
C_{L0w}	lift coefficient of wing at zero angle of attack with high-lift devices deployed		
C_{L0wf}	lift coefficient of wing-fuselage combination at zero angle of attack with high-lift devices deployed		
ΔC_{L0w}	increment in wing lift coefficient at zero angle of attack due to deployment of high-lift devices		
ΔC_{L0lw}	increment in wing lift coefficient at zero angle of attack due to deployment of leading-edge devices		
ΔC_{L0tw}	increment in wing lift coefficient at zero angle of attack due to deployment of trailing-edge flaps		
c	chord of basic aerofoil, local chord of basic wing	m	ft
c'	extended chord, i.e. chord with high-lift devices deployed	m	ft
$\bar{c}$	aerodynamic mean chord of basic wing	m	ft
c_l	chord of leading-edge device (see Sketches 1.4 and 1.6)	m	ft
Δc_l	chord extension due to deployment of leading-edge device (see Sketches 1.4 to 1.6)	m	ft
c_l'	extended chord of leading-edge device (see Sketches 1.4 to 1.6)	m	ft
c_p	wing chord at $\eta = \eta_p$ (see Sketch 1.2a)	m	ft
c_r	wing root (centre-line) chord	m	ft

c_t	chord of trailing-edge flap (see Sketches 1.7 and 1.8)	m	ft
Δc_t	chord extension due to deployment of trailing-edge flap (see Sketches 1.7 and 1.8)	m	ft
c_{t1}, c_{t2}, c_{t3}	chord of first, second, third element of slotted trailing-edge flap system (see Sketch 1.9)	m	ft
$c'_{t1}, c'_{t2}, c'_{t3}$	extended chord of first, second, third element of slotted trailing-edge flap system (see Sketch 1.9)	m	ft
$\Delta c_{t1}, \Delta c_{t2}, \Delta c_{t3}$	chord extension due to deployment of first, second, third element of slotted trailing-edge flap system (see Sketch 1.9)	m	ft
G_l	slat or vented Krüger gap; distance between device trailing edge and section surface, measured normal to surface (see Sketches 1.6a and 1.6b)	m	ft
H	fuselage height (see Sketch 1.10)	m	ft
H_l	height of slat or vented Krüger flap trailing-edge above section chord line (see Sketches 1.6a and 1.6b)	m	ft
h	height of quarter-chord point of wing root (centre-line) chord above fuselage longitudinal centre-line (see Sketch 1.10)	m	ft
INT	input file entry to allow or prohibit possibility of fuselage/trailing-edge flap interference (see Section 4.1)		
i_W	angle between chord line of wing root (centre-line) section and fuselage longitudinal centre-line (see Sketch 1.10)	deg	deg
L_l	overlap between deployed slat or vented Krüger flap trailing-edge and fixed-section nose (see Sketches 1.6a and 1.6b)	m	ft
M	free-stream Mach number		
M_k	values of free-stream Mach number in input file, $1 \leq k \leq N_{MR}$		
$N_l, (N_l)_j$	input file entry to indicate type of leading-edge device (see Table 3.2), for multiple panels $j = 1$ to N_{nl}		
N_M	number of Mach number values for calculation in ESDUpac B9931		
N_{MR}	number of pairs of Mach number and Reynolds number values for calculation in ESDUpac A9931		
N_{nl}	number of leading-edge device panels on each half-wing		
N_{nt}	number of trailing-edge flap panels on each half-wing		

$N_t, (N_t)_j$	input file entry to indicate type of trailing-edge flap (see Table 3.2), for multiple panels $j = 1$ to N_{nt}		
$N_{\delta l}$	number of deflection angle settings of leading-edge devices		
$N_{\delta t}$	number of deflection angle settings of trailing-edge flaps		
n	denotes n^{th} -chord line ($0 \leq n \leq 1$), $n = 0$ indicates leading edge, $n = 1$ indicates trailing edge		
q	free-stream kinetic pressure	N/m ²	lbf/ft ²
R_c	Reynolds number based on c		
$R_{\bar{c}}$	Reynolds number based on $\bar{c}$		
R_k	values of Reynolds number in input file, based on c for aerofoil and $\bar{c}$ for wing, $1 \leq k \leq N_{MR}$		
S	wing planform area	m ²	ft ²
s	semispan	m	ft
TES	input file entry to indicate presence or absence of slotted trailing-edge flap for basic aerofoil or wing (any panel) (see Section 4.1)		
$TYPE$	input file entry to indicate aerofoil or wing section category (see Section 4.1)		
t	section maximum thickness	m	ft
t_b	section thickness at trailing-edge	m	ft
W	fuselage width (see Sketch 1.10)	m	ft
x	general chordwise station measured streamwise from leading edge of section, positive aft, (see Sketch 1.1d)	m	ft
x_{cm}	chordwise location of z_{cm} (see Sketch 1.1d)	m	ft
x_l	chordwise location of undeployed slat trailing edge (see Sketch 1.6a)	m	ft
x_{lm}	chordwise location of z_{lm} (see Sketch 1.8)	m	ft
x_n	chordwise location of fixed-section nose (see Sketch 1.6a)	m	ft
x_{tr}	chordwise location of boundary-layer transition; same location assumed on both surfaces	m	ft
x_{ts}	chordwise location of flap-shroud trailing edge (see Sketch 1.9)	m	ft
x_{um}	chordwise location of z_{um} (see Sketch 1.1d)	m	ft

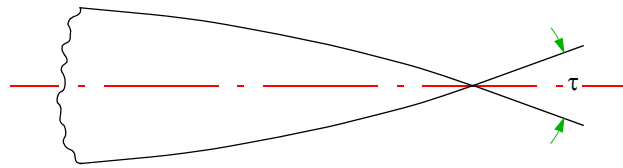
x_τ	chordwise location of trailing edge of c_l' for deployed Krüger flaps and sealed slats (see Sketch 1.5)	m	ft
z	general section ordinate, positive above chord line	m	ft
z_{cm}	maximum height of camber line of basic aerofoil or wing section (see Sketch 1.1d)	m	ft
z_{c50}	camber-line ordinate at $x/c = 0.5$	m	ft
z_{c92}	camber-line ordinate at $x/c = 0.92$	m	ft
z_h	vertical location of hinge line for drooped leading edge (see Sketch 1.4)	m	ft
z_l	lower-surface ordinate, negative below chord line	m	ft
$(z_l/c)_\xi$	value of z_l/c at $x = \xi c/100$		
z_{lm}	maximum (negative) lower-surface ordinate (see Sketch 1.8)	m	ft
z_u	upper-surface ordinate, positive above chord line	m	ft
$(z_u/c)_\xi$	value of z_u/c at $x = \xi c/100$		
z_{um}	maximum upper-surface ordinate (see Sketch 1.1d)	m	ft
α	angle of attack of aerofoil or wing root (centre-line) chord	deg	deg
α_{CLm}	angle of attack for C_{Lm}	deg	deg
α_{CLmaxw}	angle of attack for C_{Lmaxw}	deg	deg
$\alpha_{CLmaxwf}$	angle of attack for C_{Lmaxwf}	deg	deg
α_{0Bw}	zero-lift angle of attack for basic wing (high-lift devices not deployed)	deg	deg
α_{0wf}	zero-lift angle of attack for wing-fuselage combination with high-lift devices deployed	deg	deg
δ_l	deflection of leading-edge device (see Sketches 1.4 to 1.6)	deg	deg
δ_t	deflection of trailing-edge flap, positive trailing-edge down (see Sketches 1.7 and 1.8)	deg	deg
$\delta_{t1}, \delta_{t2}, \delta_{t3}$	deflection of first, second, third element of slotted trailing-edge flap system relative to datum of preceding element, positive trailing-edge down (see Sketch 1.9)	deg	deg
$\delta_{2/3}$	twist angle of wing chord at $\eta = 2/3$, positive nose up	deg	deg
η	spanwise distance from wing root (centre-line) chord as fraction of semi-span		

$(\eta_{il})_j, (\eta_{ol})_j$	values of η for inboard, outboard limits of j 'th leading-edge device panel, $1 \leq j \leq N_{nl}$		
$(\eta_{it})_j, (\eta_{ot})_j$	values of η for inboard, outboard limits of j 'th trailing-edge flap panel, $1 \leq j \leq N_{nt}$		
η_p	value of η for section of basic wing with peak loading due to angle of attack (see Sketch 1.2)		
Λ_h	hinge-line sweep of trailing-edge flaps, (see Sketch 1.3b)	deg	deg
Λ_n	sweep of n^{th} -chord line of basic wing	deg	deg
λ	wing taper ratio, tip chord/root (centre-line) chord		
ξ	dimensionless coordinate, $\xi = 100x/c$		
ρ_l	leading-edge radius (see Sketches 1.4 to 1.6)	m	ft
τ	trailing-edge angle (see Sketch 1.1a)	deg	deg
τ_a	trailing-edge angle (see Sketch 1.1b)	deg	deg
τ_{au}	trailing-edge angle (see Sketch 1.1c)	deg	deg
τ_u	aerofoil upper-surface angle (see Sketch 1.1d)	deg	deg
ϕ_t	angle between aerofoil datum and tangent to upper surface at trailing edge (see Sketch 1.7a)	deg	deg

Subscripts

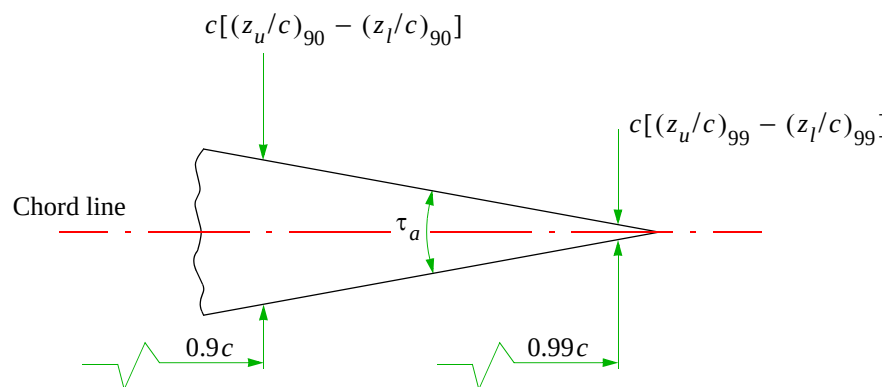
i, k	dummy integer variables
j	as in (), denotes values associated with device panels $1 \leq j \leq N_{nl}$, $1 \leq j \leq N_{nt}$. Section properties are taken at panel mid-span
r	as in (), denotes values associated with root (centre-line) section
$2/3$	as in (), denotes values associated with section at $\eta = 2/3$

Taken from Item Nos
Aero W.01.01.05
72024



(a)

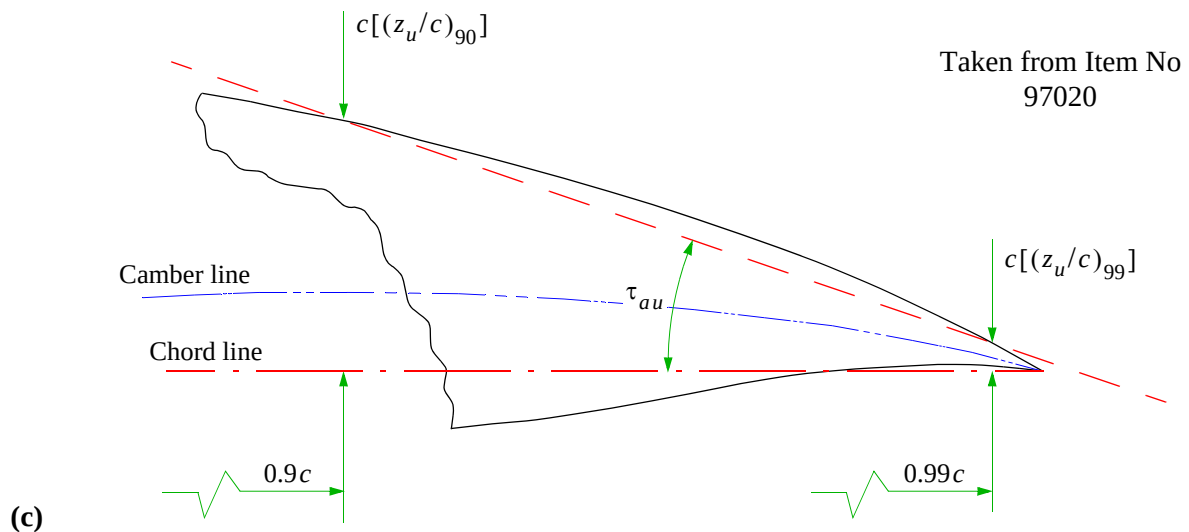
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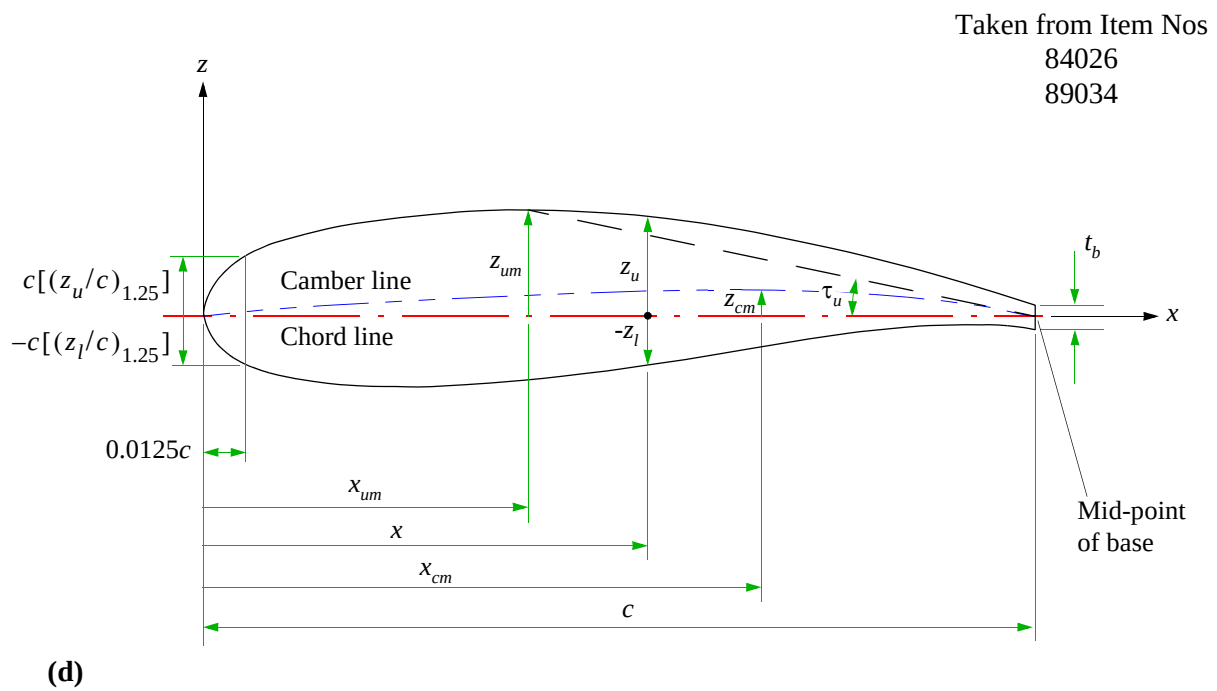
(b)

$$\tan(\tau_a/2) = \{[(z_u/c)_{90} - (z_l/c)_{90}] - [(z_u/c)_{99} - (z_l/c)_{99}]\} / 0.18$$

Sketch 1.1 Aerofoil geometry (*continuing*)

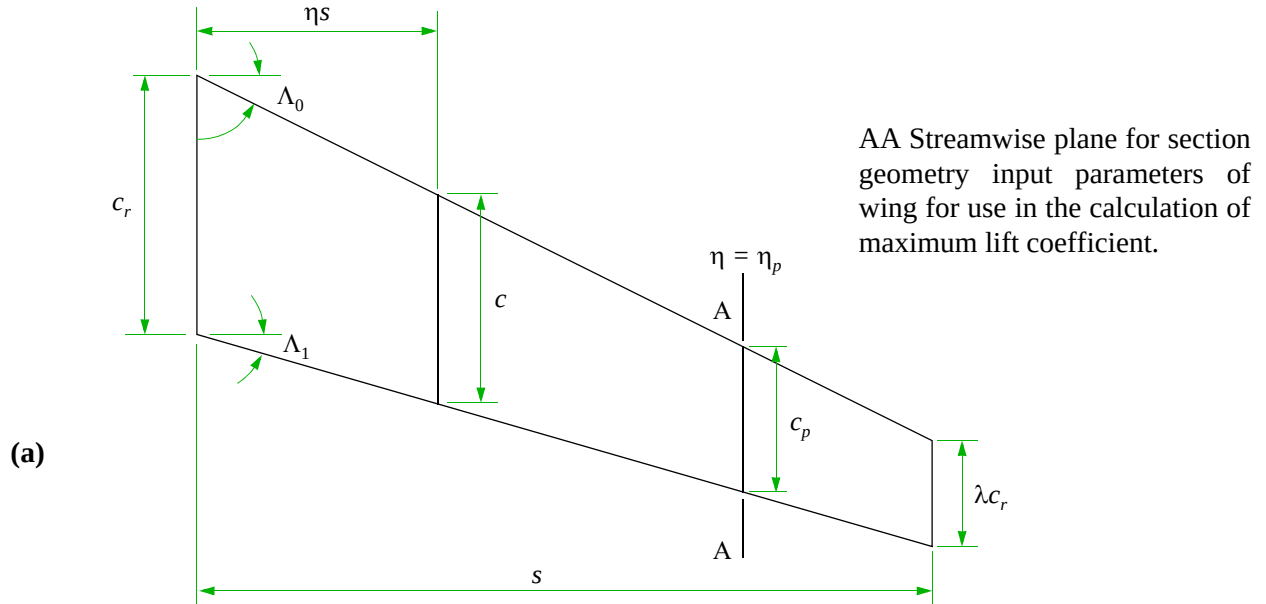


$$\tan \tau_{au} = [(z_u/c)_{90} - (z_u/c)_{99}] / 0.09$$

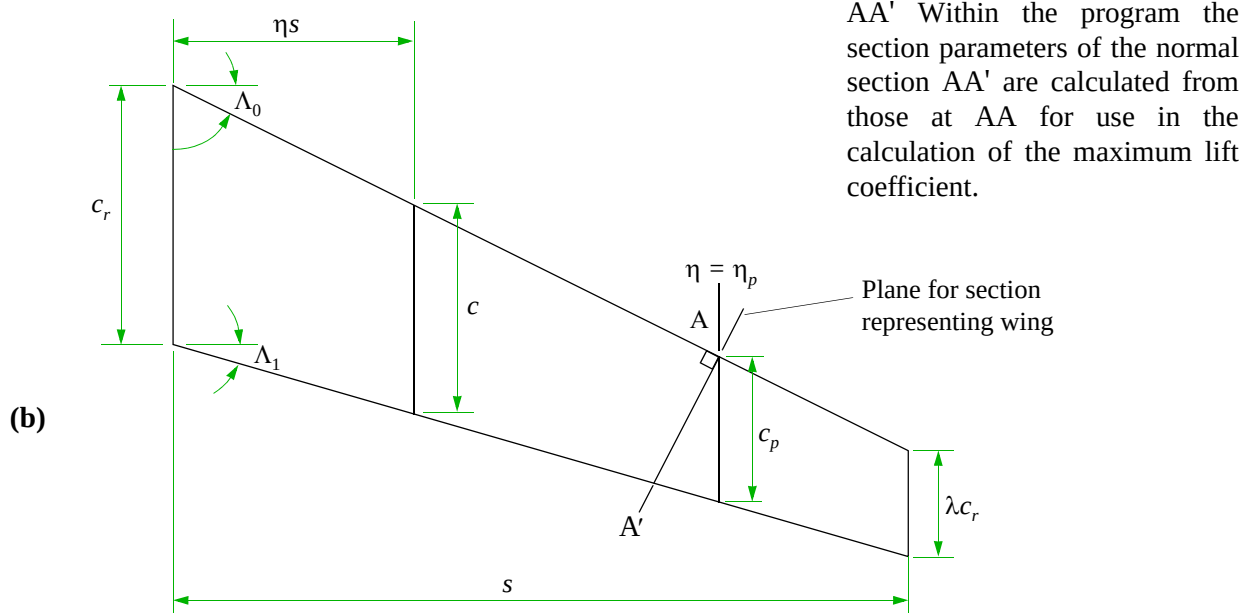


Sketch 1.1 (concluded)

Taken from Item No.
89034



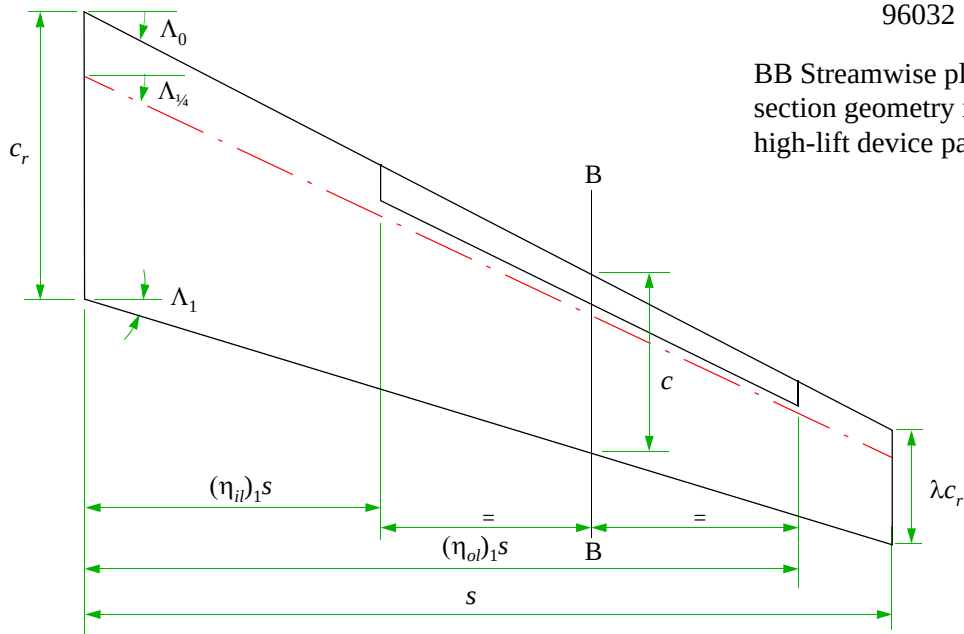
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Sketch 1.2 Wing notation, with indication of station $\eta = \eta_p$

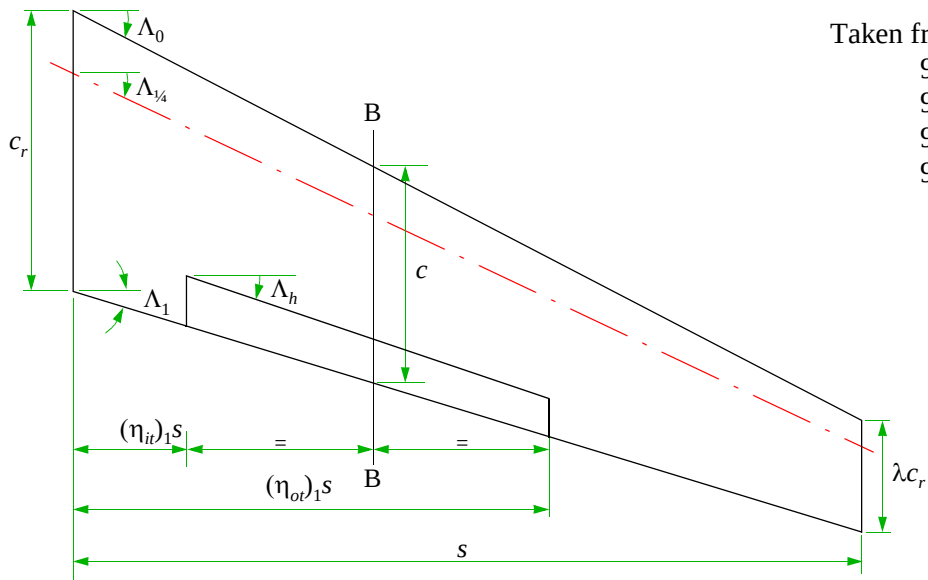
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96032

BB Streamwise planes for
section geometry input of
high-lift device panels.



(a) Leading-edge device

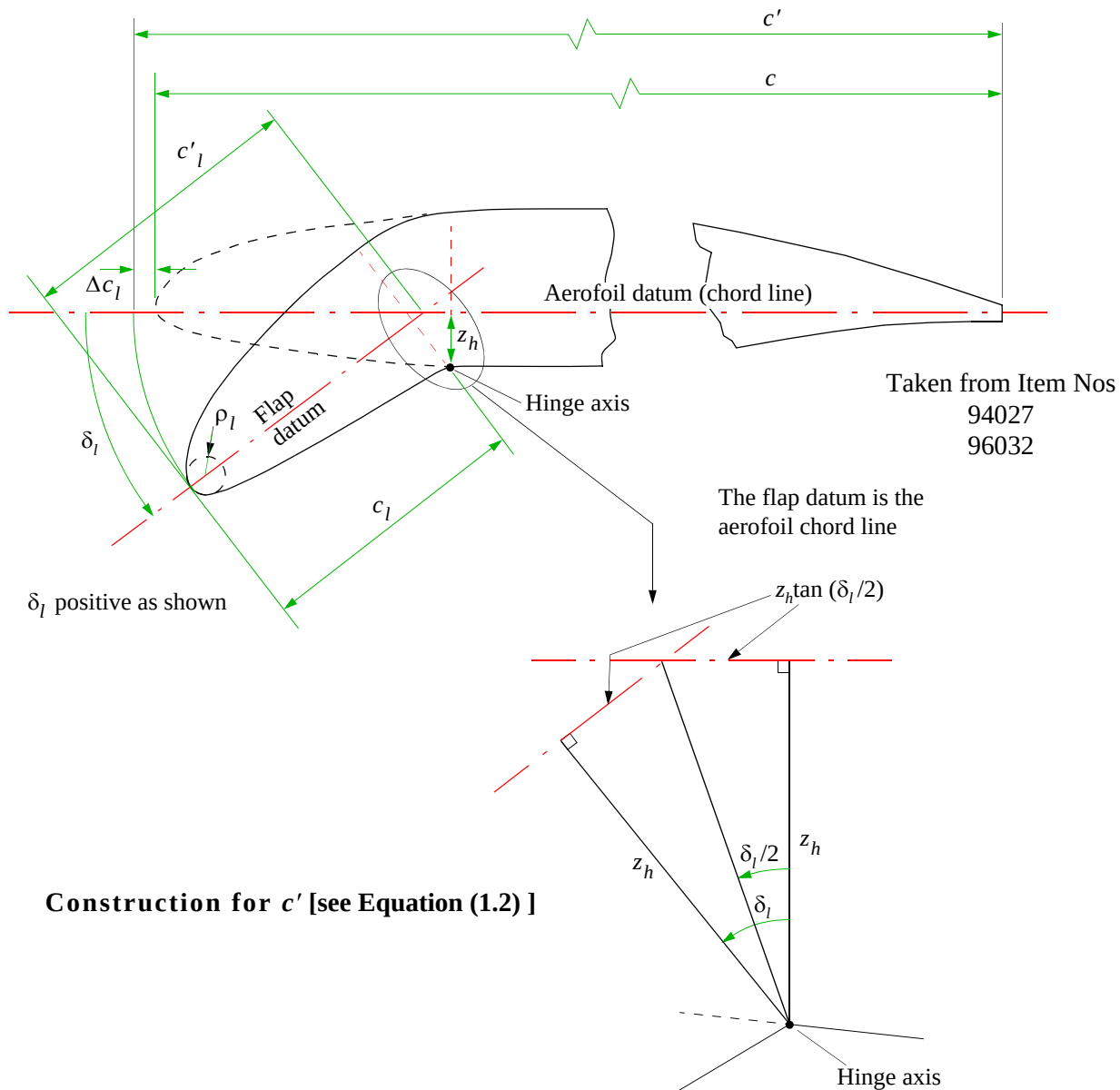
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Note that for trailing-edge slotted flaps it is normally sufficiently accurate to take Λ_h to be the sweep of the leading edge of the flap system in its deployed position or the sweep of the trailing edge of the flap shroud.

(b) Trailing-edge flap

Sketch 1.3 Wing notation with high lift devices undeployed (single panel)



From the sketch,

$$c'_l = c_l + z_h \tan(\delta_l/2) \quad (1.1)$$

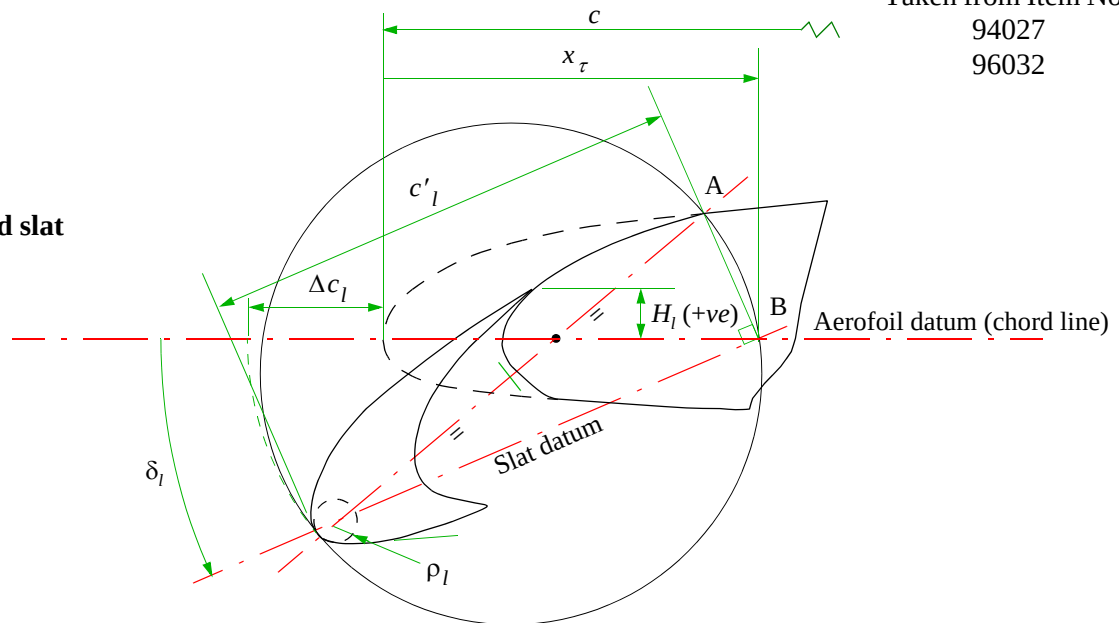
and
$$c' = c + 2z_h \tan(\delta_l/2). \quad (1.2)$$

Equations (1.1) and (1.2) relate to the specific type of device shown in the sketch, *i.e.* a device deployed by rotation around a lower-surface hinge. Many variations in design are possible and the corresponding values of c'_l and c' appropriate to any other arrangement used would have to be determined.

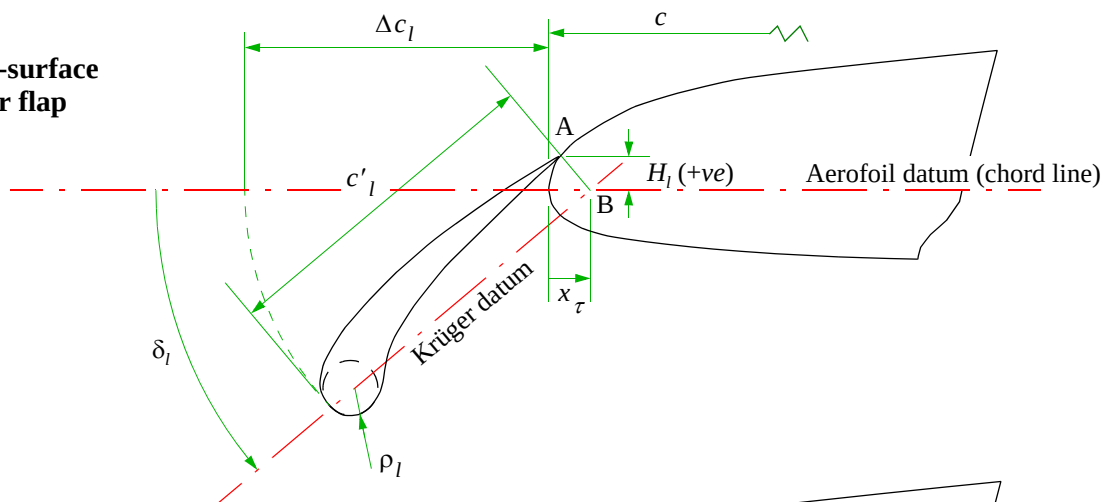
Sketch 1.4 Plain leading-edge flaps and drooped leading edges

Taken from Item Nos
94027
96032

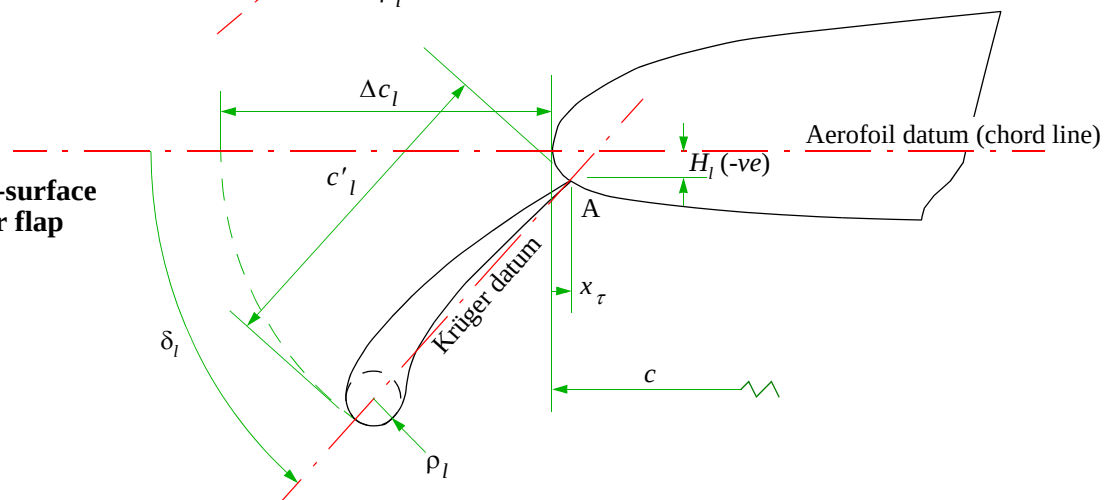
(a) Sealed slat



(b) Upper-surface Krüger flap



(c) Lower-surface Krüger flap



Sketch 1.5 Krüger flaps and sealed slats (continuing for definitions of A and B)

Krüger flaps and sealed slats have no slot and are therefore quite similar, in terms of their operation, to plain leading-edge flaps; the method for predicting ΔC_{L0l} is likewise similar. Care is, however, required for the definition of equivalent flap chord c'_l and deflection angle δ_l . Sketches 1.5(a) to 1.5(c) show how they are defined for sealed slats and two forms of Krüger flap, termed upper-surface and lower-surface Krüger flaps.

For the sealed slat and upper-surface Krüger flap the equivalent leading-edge flap is taken to be related to that part of the aerofoil and flap forward of the point at which the section first departs from the original upper-surface profile due to the deployment of the flap, *i.e.* points A on Sketches 1.5(a) and 1.5(b). The flap chord and deflection consistent with this are shown on those sketches. There is of necessity a small difference in the definitions for the case of the lower-surface Krüger flap, see Sketch 1.5(c).

Sealed slats and upper-surface Krüger flaps (Sketches 1.5(a) and 1.5(b))

For sealed slats and upper-surface Krüger flaps the values of c'_l and δ_l for the equivalent plain flap are obtained as follows. A straight line is drawn from the leading edge, passing through the centre of the flap leading-edge radius to point A, the point of departure from the original aerofoil surface. A circle centred on the mid-point of this line intersects the basic aerofoil chord at point B. The straight line joining B to the flap leading edge and passing through the centre of the flap leading-edge radius defines the equivalent plain flap chord c'_l . The angle between that chord and the aerofoil chord defines δ_l .

Lower-surface Krüger flap (Sketch 1.5(c))

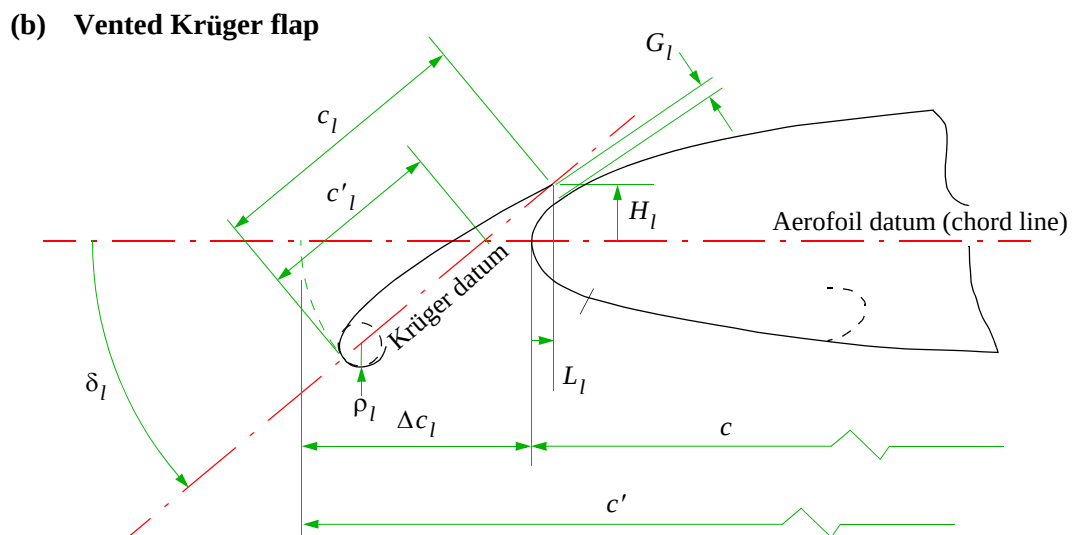
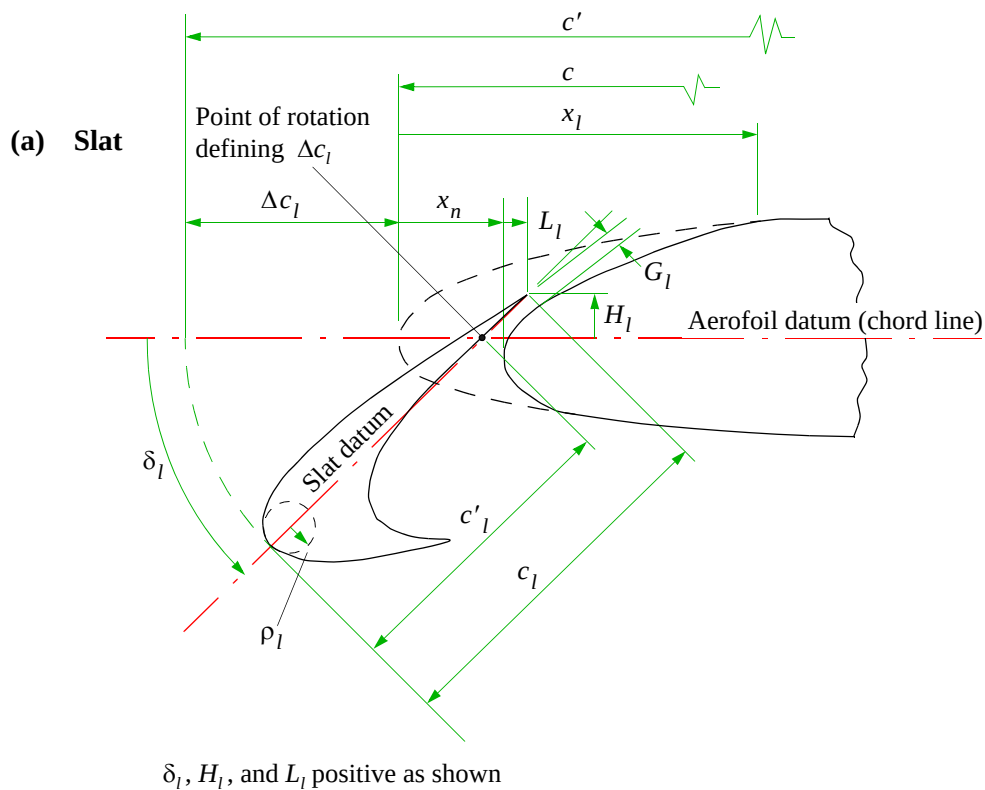
For a lower-surface Krüger flap, where the flap trailing-edge is on the aerofoil lower surface, the chord c'_l is taken as the length of the line drawn from A to the leading edge of the flap, passing through the centre of the flap leading-edge radius. The angle between that chord and the aerofoil chord defines δ_l .

In Sketches 1.5(a) to 1.5(c)

$$c' = c + \Delta c_l = c + c'_l - x_\tau \quad (1.3)$$

Sketch 1.5 Krüger flaps and sealed slats (concluded)

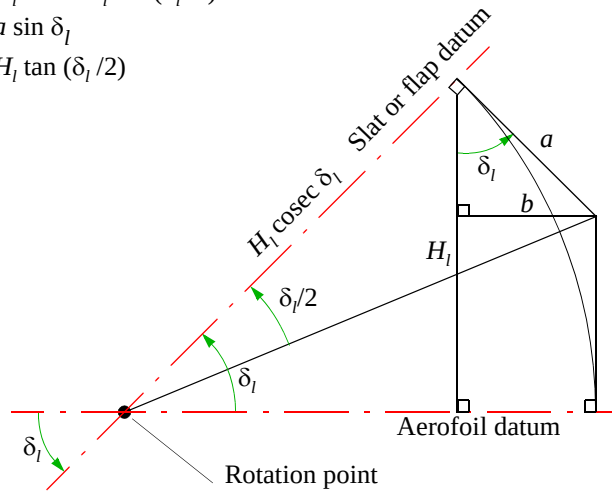
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Sketch 1.6 Slats and vented Krüger flaps (*continuing*)

$$\begin{aligned} a &= H_l \operatorname{cosec} \delta_l \tan (\delta_l / 2) \\ b &= a \sin \delta_l \\ &= H_l \tan (\delta_l / 2) \end{aligned}$$

(c) **Construction for c'**
[see Equations (1.5a) and (1.5b)]



The deflection, δ_l , of both slats and vented Krüger flaps is defined using the angle between the datum chords of the slat (or flap) and the aerofoil. The slat (or flap) datum chord is defined using the line passing through the centre of the leading-edge radius and the slat (or flap) trailing edge. The extended chord, c' , and the chord extension, Δc_l , are defined by rotating the slat (or flap) about the intersection of the slat (or flap) and aerofoil datum chords, as shown in the sketch.

In Sketches 1.6(a) and 1.6(b), for a slat and vented Krüger flap

$$c'_l = c_l - H_l \operatorname{cosec} \delta_l. \quad (1.4)$$

For a slat, the geometry in Sketch 1.6(a) gives

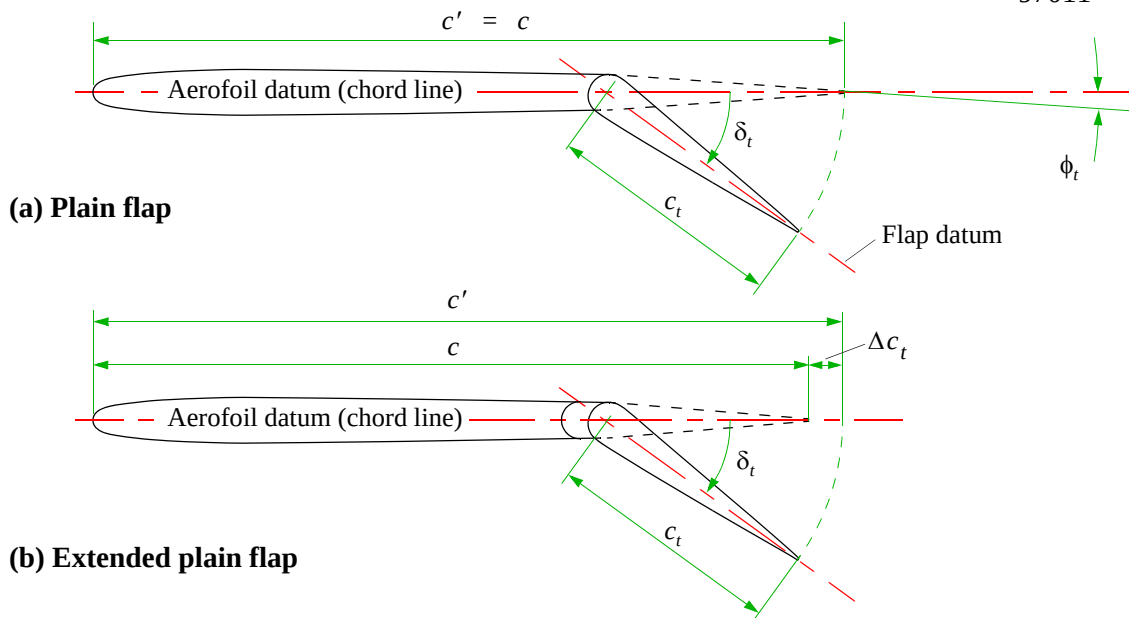
$$\begin{aligned} c' &= c + \Delta c_l \\ &= c + c_l - x_n - L_l - b \\ &= c + c_l - x_n - L_l - H_l \tan (\delta_l / 2). \end{aligned} \quad (1.5a)$$

For a vented Krüger flap, the geometry in Sketch 1.6(b) gives

$$\begin{aligned} c' &= c + \Delta c_l \\ &= c + c_l - L_l - b \\ &= c + c_l - L_l - H_l \tan (\delta_l / 2). \end{aligned} \quad (1.5b)$$

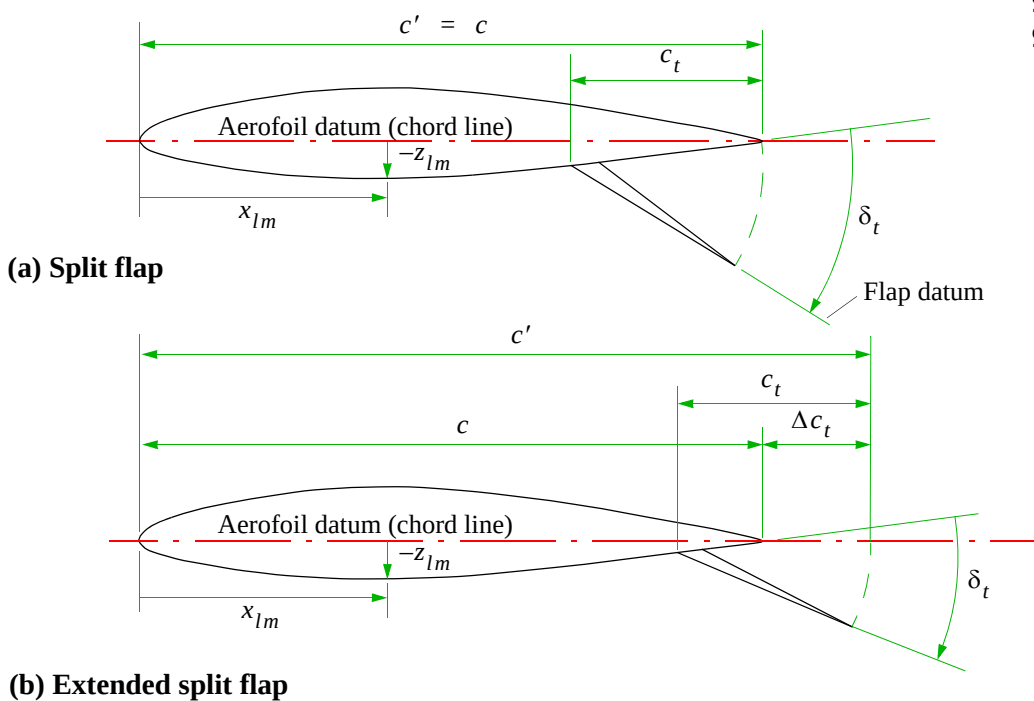
Sketch 1.6 Slats and vented Krüger flaps (concluded)

Taken from Item Nos
94028
97011



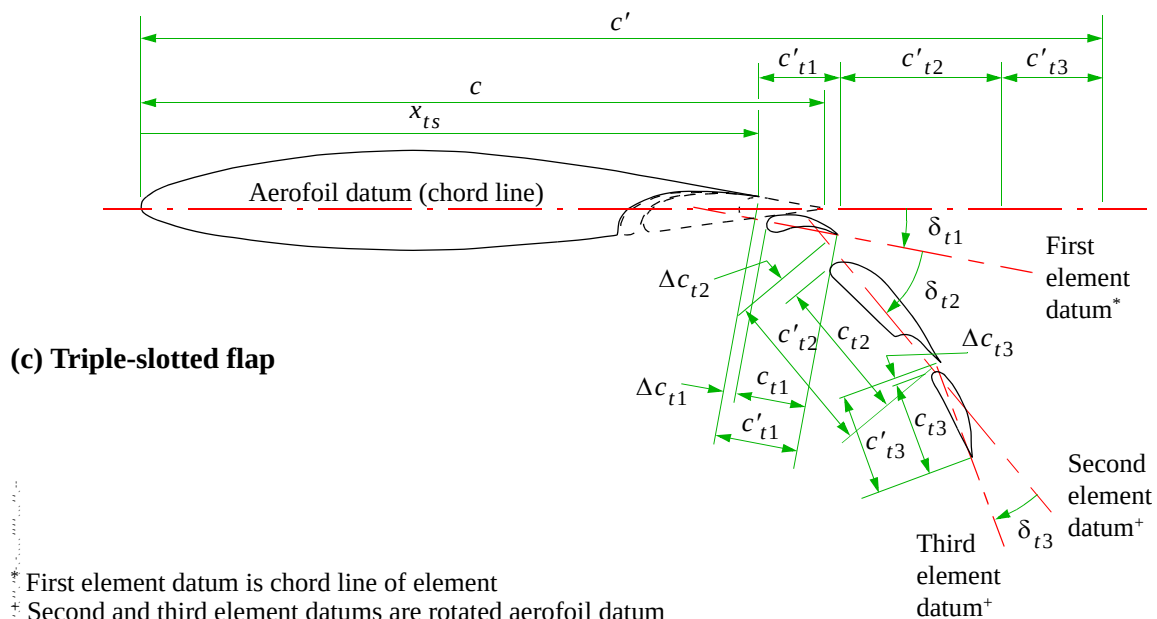
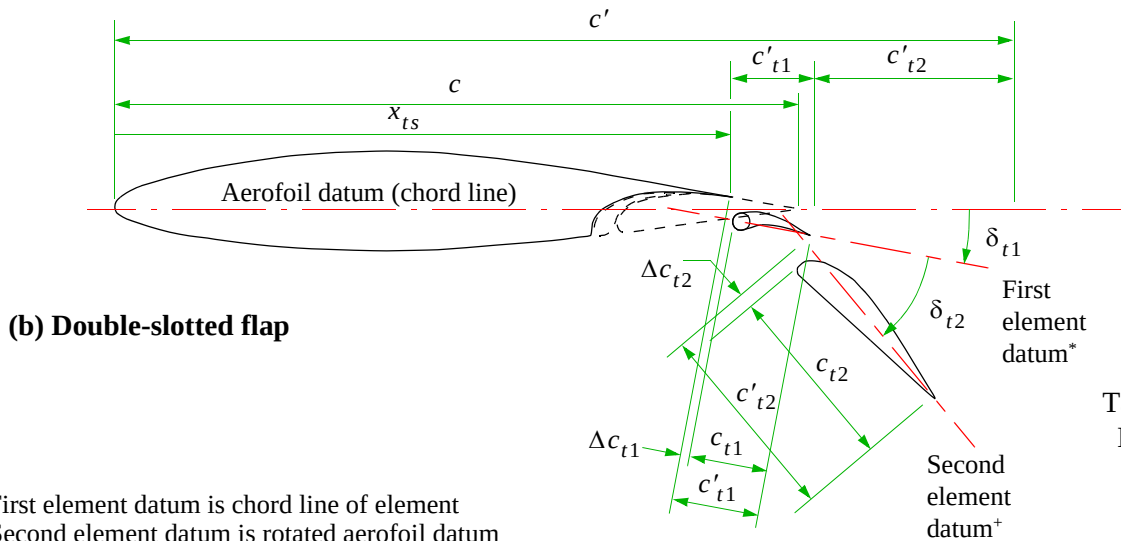
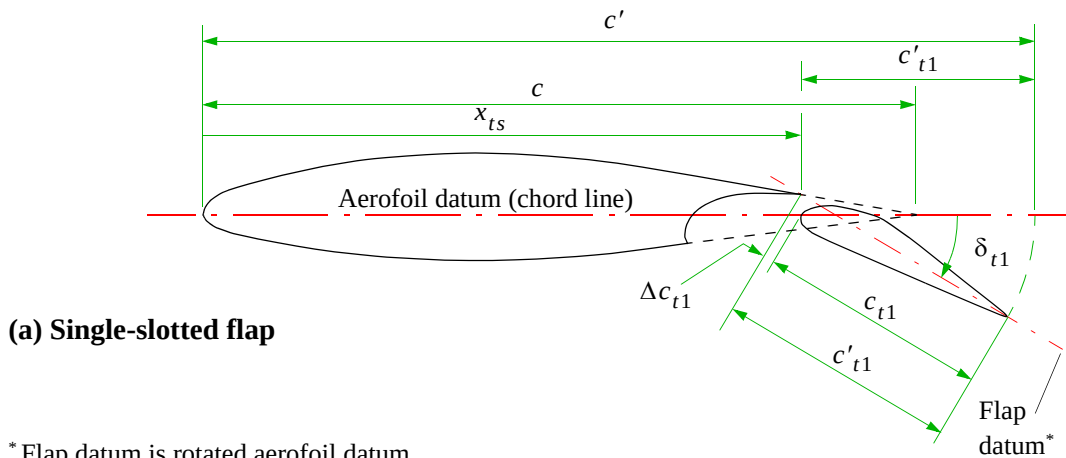
Sketch 1.7 Plain flap

Taken from Item Nos
94029
97009



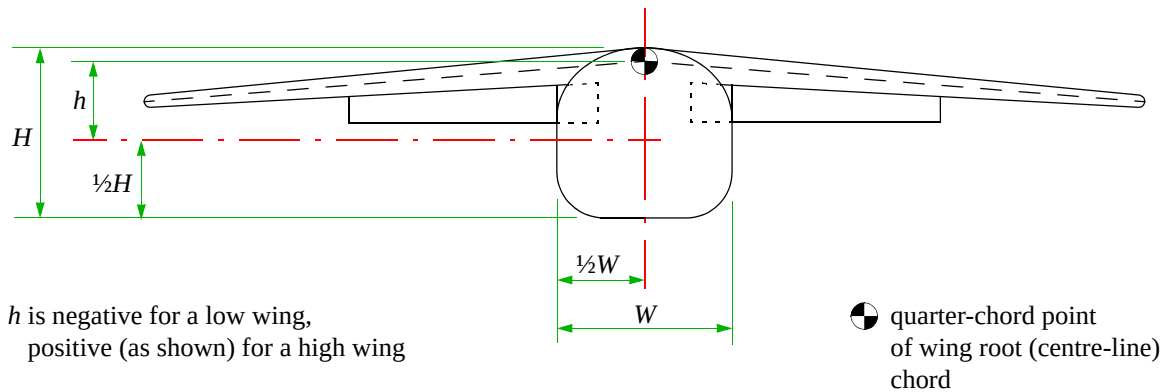
Sketch 1.8 Split flap

Taken from Item Nos
93019
94030

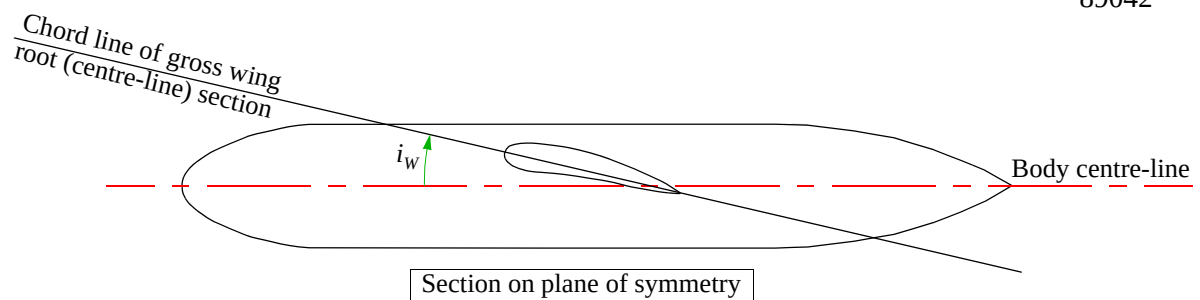


Sketch 1.9 Slotted flap

Taken from Item No
97003



Taken from Item No
89042



Sketch 1.10 Wing-fuselage geometry

2. INTRODUCTION

Details are provided of a program, ESDUpac A9931, which calculates the maximum lift coefficient, the lift coefficient at zero angle of attack and the lift curve of aerofoil sections and straight-tapered wings with high-lift devices deployed at low speeds. The basic wing (no devices deployed) may have camber and twist. The program includes methods that allow for the presence of a fuselage. For completeness, calculations can be made for the aerofoil and wing when there are no high-lift devices deployed but the prediction of the shape of the non-linear part of the lift curve should be treated with caution because it employs a technique developed for use when high-lift devices are deployed*. The method is intended for use only for free-stream Mach numbers up to about $M = 0.25$. No allowance is made for ground effect.

An auxiliary program, ESDUpac B9931, allows the prior calculation of the spanwise position of the section of the basic wing that has the peak loading due to angle of attack. It is necessary to know that section because full runs of the main program for wings and wing-fuselage combinations require its geometry.

A companion program described in Item No. 93015⁸ allows calculations of the maximum lift coefficient to be made for a basic wing for subcritical Mach numbers. The program introduced in Item No. 93015 also permits aerofoil characteristics to be calculated for Mach numbers up to 0.4. However, neither the lift coefficient at zero angle of attack nor the general shape of the lift curve is estimated.

The methods employed within the program to determine the maximum lift coefficient and the lift coefficient at zero angle of attack are those given in Item Nos 84026³ and 94027¹⁰ to 94031¹⁴ for aerofoils with and without high-lift devices and in Item Nos 89034⁴ and 91014⁶, 92031⁷, 93019⁹, 95021¹⁵, 96032¹⁷, 97009¹⁹ and 97011²⁰ for wings with and without high-lift devices. The method of Item No. 96003¹⁶ is used to calculate the lift curve. The calculation of the zero-lift incidence, the lift-curve slope and the lift coefficient at zero angle of attack for a basic aerofoil or wing section are made using Item Nos 72024², 97020²¹ and 98011²², in preference to using the more approximate methods given in other Items before Item Nos 97020 and 98011 were available. This provides greater consistency throughout the program although the changes in magnitude are small and of little consequence in the overall construction of the lift curve. Some Items employ a two-dimensional lift-curve slope from Item No. Aero W.01.01.05¹ in the estimation of device effectiveness and the method of that Item is retained in such cases to preserve the integrity of the original correlations.

The presence of a fuselage is modelled via its effect on the zero-lift angle of attack, using the method of Item No. 89042⁵, and through its influence on the performance of high-lift devices if they are fitted close to the fuselage side. The method of Item No. 97003¹⁸ is used to determine the fuselage effect on trailing-edge flaps. The influence of the fuselage on the performance of leading-edge devices is treated by assuming that they extend to the fuselage centre-line. For the fuselage effect on maximum lift coefficient the user must enter an increment derived from experiment or based on experience as there is no Data Item method for the prediction of that contribution, which is thought to be small.

Section 3 highlights some particular features of the programs. Section 4 describes the input files required by the programs A9931 and B9931 and Section 5 discusses the output. Section 7 provides examples of input and output files for specific cases. Section 6 lists the Derivation and References. Appendix A presents a flow chart that gives an alternative description of the construction of input files for A9931.

* For basic aerofoils and wings the angle of attack for the onset of non-linear behaviour may be obtained from Item No. 88030²³ and is dependent on section geometry and, for wings, planform geometry. At present there is no Data Item that addresses the lift curve beyond that angle of attack, but it is likely that the subsequent non-linear behaviour up to maximum lift is also dependent on section and planform geometry. These effects are masked by the additional effects associated with high-lift devices when they are deployed and are not represented in the simple fairing that models the lift curve as the maximum lift coefficient is approached. However, comparisons with a few data have shown that the use of the fairing developed for cases where devices are deployed gives an improvement over a linear assumption and, in the absence of a more detailed alternative, this is used for basic aerofoils and wings to provide an initial estimate of the curve in the non-linear range.

3. PROGRAM FEATURES

For aerofoil calculations, the program A9931 reads in all the required geometric and flow data from a single directed input file. For the case of a wing or a wing-fuselage combination it is usually necessary to run the auxiliary program B9931 first in order to determine η_p , the spanwise location of the peak loading due to angle of attack for the basic wing. Strictly, this varies with Mach number but for $M \leq 0.25$ the movement is sufficiently small to be neglected. It is suggested that users run B9931 for a Mach number in the middle of their range of interest. With η_p known, the user can then identify the section geometry at that station and enter it in the input file for A9931.

There is one special feature in the entry of the basic-section geometry in that the user may specify whether the section is of 'conventional' or 'modern' type, the latter description denoting rear-loaded sections which terminate with a small trailing-edge base thickness, incorporate large rear camber, and have a higher maximum lift coefficient than 'conventional' sections with the same $(z_u/c)^{1.25}$ and $\tan \tau_u$, see Item No. 84026³. Alternatively, the program can be set to make the decision automatically through a test based on the geometry over the rear 10% of the section chord. If, at $x/c = 0.9$, $(z_u - z_l)/z_u > 0.64$ and if, at the trailing-edge, $t_b/c > 0.004$, the section is taken as 'modern'. (These criteria were adopted in Item No. 93015⁸.) If the user selects the category a warning message is output should the choice be contrary to that which the program would have made.

Where geometric details of the aerofoil, wing, fuselage or high-lift system are required the user is referred to sketches that identify the necessary parameters. The sketches identify the source Data Item(s) from which they have been reproduced, with additional annotation in some cases.

For the basic wing, linear and non-linear spanwise variations of geometric twist and camber are allowable, provided that the twist variation and the spanwise variation of section zero-lift angle due to camber are monotonic and give a modest effective tip twist angle, of less than 10° say. For such cases, geometric data are required only for the wing root (centre-line) section and the wing section at $\eta = 2/3$, see Item No. 89034⁴.

Calculations may be made at a number of Mach number and Reynolds number pairs. At each pair there may be one or more settings of the leading-edge device. At each leading-edge device setting there may be one or more trailing-edge flap settings.

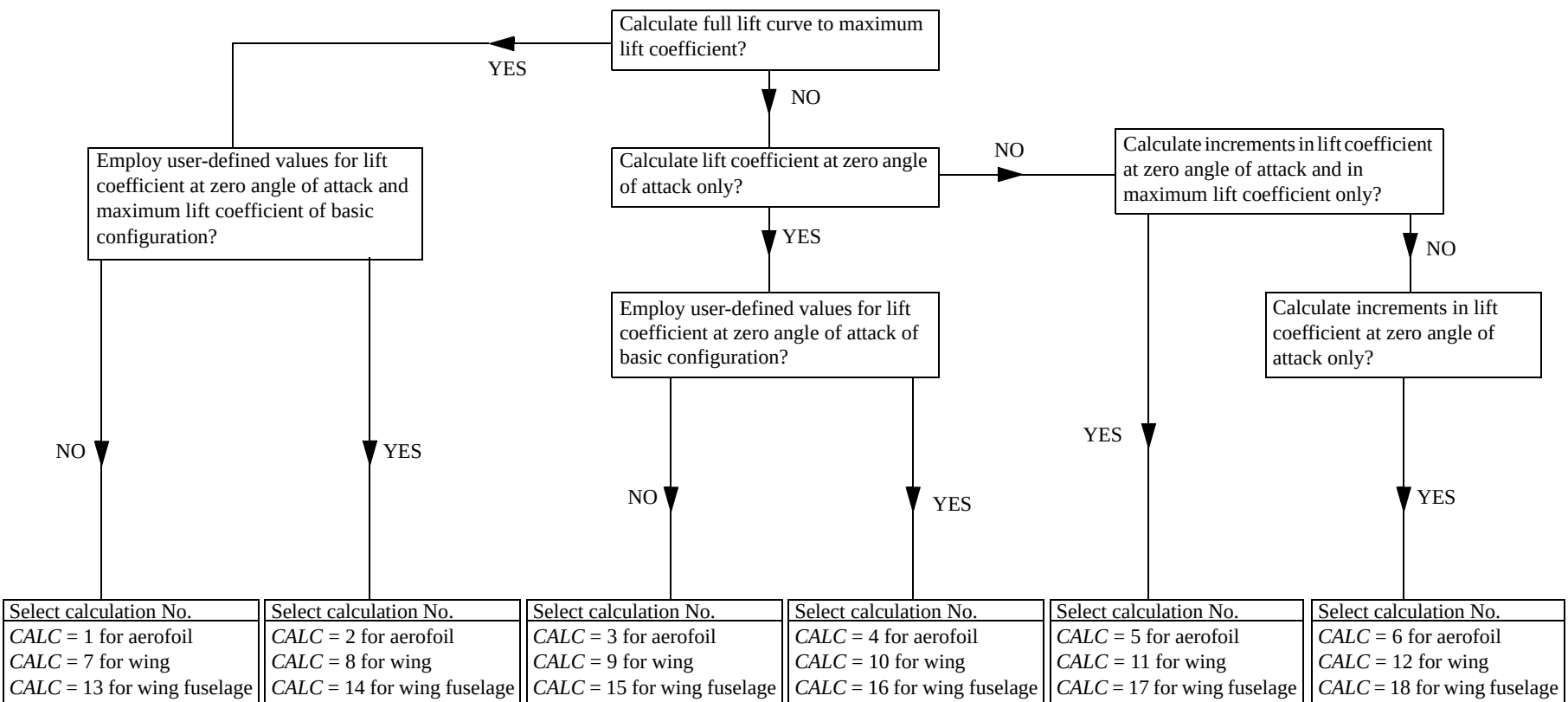
The user is allowed to select from a number of different types of run. Instead of calculating the whole lift curve, a run may be restricted to the calculation of the lift coefficient at zero angle of attack or to the calculation of the increment in the lift coefficient at zero angle of attack due to the deployment of high-lift devices, with or without the calculation of the increment in maximum lift coefficient. There is also a provision for user-defined lift coefficients to be used for the basic aerofoil, wing or wing-fuselage combination in place of the calculated values. Table 3.1 associates each type of run with an integer value of the parameter *CALC* which is used as a switch in the program. The flow chart in Sketch 3.1 guides the user through the run selection process.

For wing and wing-fuselage combinations, calculations of the increments due to device deployment require section data at the mid-span of each device panel, see Sketch 1.3. Where needed in the calculation of the increments in maximum lift coefficient, the program automatically converts section properties and leading-edge device deflection angles to those normal to the leading edge and converts trailing-edge flap deflection angles to those normal to the flap hinge-line. For calculations that do not include the estimation of increments in maximum lift coefficient fewer section data are required.

TABLE 3.1

CALC	Calculation for			Calculate					User-defined values for				
	Aerofoil	Wing	Wing-fuselage combination	C_{L0wf} , C_{L0w} or C_{L0} see NOTE	C_{Lmaxwf} , C_{Lmaxw} or C_{Lm} see NOTE	Lift-curve	ΔC_{L0w} or ΔC_{L0} only	$\Delta C_{L0w}, \Delta C_{Lmax}$ or $\Delta C_{L0}, \Delta C_{Lm}$ only	C_{L0B} or C_{L0Bw}	C_{LmaxB} or C_{LmB}	C_{L0Bf}	C_{LmaxBf}	ΔC_{Lmaxf}
1	✓			✓	✓	✓							
2	✓			✓	✓	✓			✓	✓			
3	✓			✓									
4	✓			✓					✓				
5	✓							✓					
6	✓						✓						
7		✓		✓	✓	✓							
8		✓		✓	✓	✓			✓	✓			
9		✓		✓									
10		✓		✓					✓				
11		✓						✓					
12		✓					✓						
13			✓	✓	✓	✓							✓
14			✓	✓	✓	✓					✓	✓	
15			✓	✓									
16			✓	✓							✓		
17			✓					✓					
18			✓				✓						

NOTE: It is possible to run the program with no high-lift devices deployed, in which case only the properties of the basic wing-fuselage combination, wing or aerofoil, C_{L0Bf} , C_{L0Bw} , C_{L0B} , C_{LmaxBf} , C_{LmaxB} or C_{LmB} will be output.



Sketch 3.1

The different types of high-lift devices that may be selected are given in Table 3.2, together with the integers N_l or $(N_l)_j$ and N_t or $(N_t)_j$ that will be used to identify them in the program input file.

TABLE 3.2

<i>Integers that identify the type of leading-edge and trailing-edge high-lift devices.</i>			
<i>Leading-edge device</i>		<i>Trailing-edge flap</i>	
N_l or $(N_l)_j$	Type	N_t or $(N_t)_j$	Type
1	plain flap	1	plain
2	droop	2	split
3	sealed slat	3	slotted
4	Krüger	4	double-slotted
5	slat	5	triple-slotted
6	vented Krüger		

The absence of leading-edge or trailing-edge devices is indicated by setting the number of deflection angles $N_{\delta l}$ or $N_{\delta t}$ to zero.

Calculations may be carried out with different types of high-lift devices over different spanwise segments of a wing. For example, there could be a slotted trailing-edge flap panel over the inboard part of a wing and a plain trailing-edge flap panel outboard. There is an important restriction in the case of leading-edge devices: if the calculation is to include the prediction of maximum lift coefficient, there can be only a single device panel which must extend to the wing tip.

Section 4 describes the input files for the programs A9931 and B9931 in detail.

4. INPUT FILES

4.1 Input File for A9931

Table 4.1 lists, in order, the parameters required in the input file for ESDUpac A9931. Unless specified as integers or characters all entries may be entered in free numerical format, separated by one or more spaces. A supplementary flow chart that gives a diagrammatic representation of the input is given in Appendix A but it includes fewer comments.

At the top of the input file there is provision for the use of three lines of text, each ending with a carriage return, to describe a particular run. These will be reproduced in the output file. A carriage return must still be entered for any blank line.

This is followed by the entry *CALC*, a calculation number chosen from the first column of Table 3.1 or via the flow chart in Sketch 3.1 to specify the type of run to be performed.

The number N_{MR} of Mach number and Reynolds number pairs at which calculations are to be performed is then entered, followed by a column listing of the values M_k, R_k for $1 \leq k \leq N_{MR}$. A maximum of ten pairs is permitted.

If a calculation for a wing or wing-fuselage combination has been selected, $CALC = 7$ to 18, the geometric parameters that define the wing planform, namely aspect ratio, A , taper ratio, λ , fractional chord line, n , and the sweep angle of that line, Λ_n , are entered.

For calculations involving the maximum lift coefficient of wings, *i.e.* $CALC = 7, 8, 11, 13, 14$ or 17, it is now necessary to enter the value of η_p , the dimensionless spanwise location of the peak loading due to angle of attack for the basic wing, which may be found by using ESDUpac B9931, see Section 4.2. For the same wing and wing-fuselage calculation options and for the aerofoil calculation options $CALC = 1, 2, 4, 5$ and 6 a value of 0 or 1 is now entered for the switch *TES* to indicate the absence or presence (respectively) of single or multiple slotted trailing-edge flaps for any one of the wing trailing-edge panels or the aerofoil. This affects both the required input data and some calculations made within the program, usually those for maximum lift coefficient, where the presence of slotted trailing-edge flaps affects the efficiency of leading-edge devices.

The next set of entries relates either to the geometry of the basic aerofoil or of the basic-wing section at η_p . A 'full' set of section data is required if the calculations involve the prediction of maximum lift for the basic aerofoil or wing ($CALC = 1, 7$ or 13) or the calculation of the lift coefficient at zero angle of attack for the basic aerofoil ($CALC = 3$). A much smaller set of section data or none is usually required in other calculations because user-defined values of lift coefficients are supplied or the calculations are restricted to increments due to device deployment or to zero angle of attack. However, there is the exception that if slotted trailing-edge flaps are present ($TES = 1$) for calculation options $CALC = 2, 5, 8, 11, 14$ or 17 the full set of basic section data must be entered at this point. This is because the increment in maximum lift coefficient due to such devices involves a small contribution from a datum value for the basic-section maximum lift coefficient.

The full set of section geometry data begins with the specification of the section category or type, see Section 3. Setting $TYPE = P$ allows the program to decide the category, while $TYPE = M$ specifies a modern section and $TYPE = C$ specifies a conventional section. Then the thickness to chord ratio, t/c , and the dimensionless chordwise location of boundary-layer transition, x_{tr}/c , are entered, followed by the dimensionless chordwise location and the value of the maximum upper-surface ordinate, x_{um}/c and z_{um}/c , and the trailing-edge angle τ , in degrees. Finally, there is a column of entries of the dimensionless upper-surface and lower-surface coordinates at each of 17 standard values of dimensionless chordwise

location, $(x/c)_i$, $(z_u/c)_i$ and $(z_l/c)_i$, for $(x/c)_i = 0, 0.01, 0.0125, 0.025, 0.05, 0.1, 0.2, 0.3, 0.4, 0.5, 0.6, 0.7, 0.8, 0.9, 0.95, 0.99$ and 1.0 . Fourteen of these are required to determine the section theoretical zero-lift angle of attack, with additional values at 0.01 and 0.0125 needed by the correlating parameters for C_{LMB} and the value at 0.99 required for the determination of the trailing-edge angles τ_a and τ_{au} used in the calculation of section lift-curve slope.

For the calculation option $CALC = 2$, provided that there are no slotted trailing-edge flaps, i.e. TES has been set to zero, the section geometry that is required is restricted to the thickness to chord ratio, t/c , the dimensionless chordwise position of boundary-layer transition, x_{tr}/c , the trailing-edge angles, τ , τ_a and τ_{au} , the dimensionless chordwise location and the value of the maximum upper-surface ordinate, x_{um}/c and z_{um}/c , and the upper-surface and lower-surface ordinates at $x/c = 0.05$, $(z_u/c)_5$ and $(z_l/c)_5$. These permit the method of Item Nos 72024² and 97020²¹ to be used to estimate the basic aerofoil lift-curve slope. Note that a basic lift-curve slope is also needed for the calculation of the effectiveness of slotted trailing-edge flaps on aerofoils but in that instance, to maintain consistency with the original correlation of the flap effectiveness, the method of Item No. Aero W.01.01.05¹ is used. If $TES = 1$ and $CALC = 4$ or 6 only the latter lift-curve slope is needed, in the calculation of ΔC_{L0t} , and the section data for the basic aerofoil are reduced to t/c , τ and τ_a .

Further data are necessary at this point if the calculation of C_{L0B} for the basic aerofoil or C_{L0Bw} for the basic wing is required. For the aerofoil ($CALC = 1$ or 3) the ratio of the camber line ordinates at $x = 0.92c$ and $0.5c$, z_{c92}/z_{c50} , which is used in the method of Item No. 98011²², is all that is necessary. (A value of unity is entered for z_{c92}/z_{c50} for uncambered sections.) For the wing or wing-fuselage ($CALC = 7, 9, 13$ or 15) the entries are more complex. Firstly, the twist angle at $\eta = 2/3$, $\delta_{2/3}$, is entered. Then come the root (centre-line) section dimensionless upper-surface and lower-surface coordinates at each of 15 standard values of dimensionless chordwise location, in a column of values $(x/c)_{ri}$, $(z_u/c)_{ri}$, $(z_l/c)_{ri}$ for $i = 1$ to 15 with $(x/c)_{ri} = 0, 0.025, 0.05, 0.1, 0.2, 0.3, 0.4, 0.5, 0.6, 0.7, 0.8, 0.9, 0.95, 0.99$ and 1.0 , followed by the camber ordinate ratio $(z_{c92}/z_{c50})_r$. Values of $(x/c)_{2/3i}$, $(z_u/c)_{2/3i}$, $(z_l/c)_{2/3i}$ and $(z_{c92}/z_{c50})_{2/3}$ follow for the section at $\eta = 2/3$. These data enable the zero-lift angle for the root section and the section at $\eta = 2/3$ to be calculated and thence, with $\delta_{2/3}$, the zero-lift angle of attack for the wing. It is important to note that the methods in the underlying Data Items, and therefore in the program, allow only for linear or non-linear but monotonic twist distributions and for non-uniform camber distributions that induce a linear or non-linear but monotonic variation of section zero-lift angle across the wing semi-span. This restriction allows the wing characteristics to be modelled satisfactorily in terms of geometric data entered only at the root section and the section at $\eta = 2/3$.

The next series of entries applies when a fuselage is present ($CALC = 13$ to 18). An option is provided through the entry INT to prohibit or allow the possible fuselage effect on trailing-edge flap performance that occurs with flaps fitted close to the fuselage side. With $INT = 0$ allowance is made only for the direct effect of the fuselage on the lift coefficient at zero angle of attack and on maximum lift coefficient. The required geometric data are then the ratio of fuselage width to wing semi-span, W/s , and the setting angle of the wing root (centre-line) chord with respect to the fuselage centre-line, i_w . If $INT = 1$, the effect of the fuselage interference on flap performance is permitted and the height of the wing root chord relative to the fuselage height, h/H , is a further necessary parameter. The fuselage height and width and the wing height are all measured in a spanwise section passing through the quarter-chord point of the wing root chord, see Sketch 1.10, which also illustrates i_w . These entries are omitted if there is no fuselage.

Within the program, with $INT = 1$, for flap panels with an inboard limit that is less than $0.005s$ outboard from the fuselage side the predicted interference effect, which is modelled through a fictitious inboard extension of the flap panel towards the fuselage centre-line, is included in full, see Item No. 97003¹⁸, but the effect is decreased linearly to zero as the gap size increases to $0.05s$, and remains zero for greater gaps.

For leading-edge devices, whatever the value of INT , the panel is assumed to extend inboard to the fuselage centre-line if the gap between the inboard panel limit and the fuselage side is less than or equal to 0.005s. It is assumed that there is no interference if the gap is greater than 0.005s, when the true inboard limit of the panel is used.

(In the above, the value of 0.005s has been chosen as a representative value indicating that some effort has been made to ensure that the leading-edge and trailing-edge devices fit closely to the fuselage side. It is not an exact demarcation.)

If the user has selected a calculation option that requires user-defined data for the basic aerofoil, wing or wing-fuselage combination or for the fuselage effect on maximum lift coefficient, the necessary data are input at this point according to $CALC = 2, 4, 8, 10, 13, 14$ or 16 as indicated in Table 4.1 and the input file flow chart in Appendix A. Values must be entered N_{MR} times.

The final entries concern the geometry of the high-lift system. In all cases the user first enters the number of different settings for the leading-edge device and trailing-edge flap by entering integers for $N_{\delta l}$ and $N_{\delta t}$. If both are zero no further data are required. Setting either to zero indicates the absence of the corresponding device. A maximum of ten settings is allowed for each device.

For the aerofoil, $CALC = 1$ to 6, it is necessary only to indicate the type of leading-edge and trailing-edge high-lift devices by specifying the two integers N_l and N_t obtained by a cross-reference to Table 3.2. If $N_{\delta l} = 0$, N_l is omitted and if $N_{\delta t} = 0$, N_t is omitted. For a wing or wing-fuselage combination, the device data entry is more complex because it is necessary to allow for a number of panels of different device type, although there is the restriction that for calculations of maximum lift there can be only a single leading-edge device panel which must extend to the wing tip. So that, if $CALC = 7$ to 18, N_{nl} is entered to denote the number of leading-edge device panels, followed by a series of entries $(N_l)_j, (\eta_{il})_j, (\eta_{ol})_j$ for $1 \leq j \leq N_{nl}$ to identify the type and inboard and outboard limits for each panel, starting with the inboard panel and proceeding outboard but, for $CALC = 7, 8, 11, 13, 14$ and 17, N_{nl} must be unity and $(\eta_{ol})_1$ must be unity and the program will not permit other entries. For the remaining calculation types, $CALC = 9, 10, 12, 15, 16$ and 18, there is an upper limit of 5 on N_{nl} . As for the aerofoil case, if $N_{\delta l} = 0$, no leading-edge device data are needed. For trailing-edge flaps, for all calculations, $CALC = 7$ to 18, N_{nt} is entered to denote the number of trailing-edge device panels, followed by a series of entries $(N_t)_j, (\eta_{it})_j, (\eta_{ot})_j$ for $1 \leq j \leq N_{nt}$ to denote the type and inboard and outboard limits of each panel, starting with the inboard panel and proceeding outboard. There is an upper limit of 5 on N_{nt} . If $N_{\delta t} = 0$ no trailing-edge device data are needed.

Values of deflection angle and dimensionless geometry are then entered for the aerofoil devices or each wing device panel as detailed in Table 4.1 and the input file flow chart in Appendix A. The data required vary with device type and may include some properties of the basic section. Data are entered first for the leading-edge devices and then for the trailing-edge flaps. Some parameters (e.g. gap size) are assumed to vary with each setting and must be entered $N_{\delta l}$ or $N_{\delta t}$ times, as indicated, while other parameters (e.g. flap chord) remain constant and are entered only once.

Calculations that involve maximum lift, $CALC = 1, 2, 5, 7, 8, 11, 13, 14$ and 17, may require more data than those concerned only with the lift coefficient at zero angle of attack, $CALC = 3, 4, 6, 9, 10, 12, 15, 16$ and 18. In particular, for wing and wing-fuselage calculations it is necessary to include the hinge-line sweep for trailing-edge flaps.

Note that leading-edge slats and vented-Krüger flaps (N_l or $(N_l)_j = 5$ or 6) require the input of the device chord to section chord ratio c_l / c but that the other types of leading-edge device require the input of the extended device chord to section chord ratio c_l' / c .

For Krüger flaps the underlying methods for maximum lift assume that the leading-edge radius of the flap is the same as that of the basic section. (Experimental data for the more general case were too sparse for the development of a more comprehensive method, see Section 4.3 of Item No. 94027¹⁰. In the absence of other information, for a Krüger flap with a leading-edge radius that is different from that of the basic section, it is suggested that the additional effect on maximum lift may be approximated as the difference between the maximum lift of the basic section and that of the basic section modified to have the same leading-edge radius as the Krüger flap.)

It is of particular interest to note that when a full input file has been created for a calculation to maximum lift the user can always generate the lift curve for the basic aerofoil, wing or wing-fuselage combination by setting $N_{\delta l}$ and $N_{\delta t}$ to zero. There is no need to delete the device geometry data because with $N_{\delta l} = N_{\delta t} = 0$ they are simply not read. However, as stated in Section 2, the prediction of the non-linear shape of the lift curve should be treated with caution because it employs a technique developed for use when high-lift devices are deployed. Setting $N_{\delta l} = N_{\delta t} = 0$ for calculation types where few program options are selected, *e.g.* $CALC = 6$, will not cause the program to fail but little or only trivial data will be output.

TABLE 4.1

Input	Comments
TEXT TEXT TEXT	The first three lines of the input file are read as text and are written to the output file to enable the user to describe the run. Each line may contain up to 72 characters and must end with a carriage return. A carriage return is still required for any blank line.
CALC	Integer. Indicator for calculation type, selected from Table 3.1 or Sketch 3.1.
N_{MR}	Integer. Number of Mach number and Reynolds number pairs at which calculations are made, $1 \leq N_{MR} \leq 10$.
$\vdots$ M_k $\vdots$ $\vdots$ R_k $\vdots$ $\vdots$	Pairs of Mach number and Reynolds number, $1 \leq k \leq N_{MR}$, $0 \leq M_k \leq 0.25$; $0.7 \leq R_k \times 10^{-6} \leq 9$ (aerofoils) or ≤ 12 (wings). [†]
IF CALC = 7 to 18 A λ n Λ_n	For a wing calculation, values of aspect ratio, taper ratio, fractional chord line and sweep of n^{th} -chord line (deg), $2 \leq A \leq 10$, $0.1 \leq \lambda \leq 1$, $0 \leq n \leq 1$, $0 \leq \Lambda_0 \leq 50^\circ$, $\tan^{-1} \left[-\frac{2(1-\lambda)}{A(1+\lambda)} \right] \leq \Lambda_{1/2} \leq 50^\circ$, $-20^\circ \leq \Lambda_1 \leq 50^\circ$.
IF CALC = 7, 8, 11, 13, 14 or 17 η_p	Dimensionless spanwise location of peak loading on basic wing. Program B9931, see Section 4.2, allows η_p to be calculated.
IF CALC = 1, 2, 4, 5, 6, 7, 8, 11, 13, 14 or 17 TES	Indicator for presence of slotted trailing-edge flaps. $TES = 1$ indicates presence of slotted trailing-edge flaps for aerofoil or wing. $TES = 0$ indicates absence of slotted trailing-edge flaps for aerofoil or wing.

continuing

[†] For wings, the range $0.7 \leq R_k \times 10^{-6} \leq 12$ reflects the limits of the experimental database studied in the development of the method for C_{LmaxB} , and the program halts with an error message if that range is exceeded for CALC = 7, 8, 13 or 14. The methods in the program additionally require that $0.7 \leq R_k(c_p/\bar{c}) \cos^2 \Lambda_0 \times 10^{-6} \leq 9$.

Section properties of basic aerofoil or of basic wing at η_p .	
IF $CALC = 1, 3, 7$ or 13 OR (IF $TES = 1$ AND $CALC = 2, 5, 8, 11, 14$ or 17)	
$TYPE^*$	Indicator for category of section (see Section 3), P indicates that the program will decide, M indicates a modern section, C indicates a conventional section.
t/c	Thickness to chord ratio, $0.06 \leq t/c \leq 0.24$.
x_{tr}/c	Dimensionless chordwise location of boundary-layer transition, $0 \leq x_{tr}/c \leq 0.5$.
x_{um}/c z_{um}/c	Dimensionless chordwise location and value of maximum upper-surface ordinate (see Sketch 1.1).
τ	Trailing-edge angle (deg) (see Sketch 1.1a), $0 \leq \tau \leq 25^\circ$.
$(x/c)_i$ $(z_u/c)_i$ $(z_l/c)_i$	Section dimensionless upper-surface and lower-surface ordinates at each of 17 standard values of dimensionless chordwise location, $(x/c)_i = 0, 0.01, 0.0125, 0.025, 0.05, 0.1, 0.2, 0.3, 0.4, 0.5, 0.6, 0.7, 0.8, 0.9, 0.95, 0.99$ and 1.0 , for $1 \leq i \leq 17$.
Restricted sets of section data for basic aerofoil.	
IF $TES = 0$ AND $CALC = 2$	
t/c	Thickness to chord ratio,
x_{tr}/c	Dimensionless chordwise position of boundary-layer transition, $0 \leq x_{tr}/c \leq 0.5$.
τ	Trailing-edge angle (deg) (see Sketch 1.1a), $0 \leq \tau \leq 25^\circ$.
τ_a	Trailing-edge angle (deg) (see Sketch 1.1b), $0 \leq \tau_a \leq 25^\circ$.
τ_{au}	Trailing-edge angle (deg) (see Sketch 1.1c), $0 \leq \tau_{au} \leq 25^\circ$.
x_{um}/c z_{um}/c	Dimensionless chordwise station and value of maximum upper-surface ordinate.
$(z_u/c)_5$ $(z_l/c)_5$	Upper-surface and lower-surface ordinates at $x/c = 0.05$.
IF $TES = 1$ AND $CALC = 4$ or 6	
t/c	Thickness to chord ratio.
τ	Trailing-edge angle (deg) (see Sketch 1.1a), $0 \leq \tau \leq 25^\circ$.
τ_a	Trailing-edge angle (deg) (see Sketch 1.1b), $0 \leq \tau_a \leq 25^\circ$.

continuing

* Comments on $TYPE$ selection, see Section 3, are output only for $CALC = 1, 7$ or 13 because it has no effect on $CALC = 3$ calculations, but is input for consistency of input file format, and has only a small effect, through a basic-section datum, on the increment in maximum lift coefficient due to deployment of slotted flaps.

Further data are required for the calculation of C_{L0B} for the basic aerofoil or C_{L0Bw} for the basic wing.	
IF $CALC = 1$ or 3 .	Ratio of camber ordinate at $x = 0.92c$ to camber ordinate at $x = 0.5c$. (Entered as unity for an uncambered section.)
z_{c92}/z_{c50}	
IF $CALC = 7, 9, 13$ or 15 .	Geometric twist angle at $\eta = 2/3$, in degrees, relative to the root (centre-line) chord, positive nose up. Section dimensionless upper-surface and lower-surface ordinates at each of 15 standard values of dimensionless chordwise location $(x/c)_i = 0, 0.025, 0.05, 0.1, 0.2, 0.3, 0.4, 0.5, 0.6, 0.7, 0.8, 0.9, 0.95, 0.99$ and 1.0 for $1 \leq i \leq 15$. Ratio of camber ordinate at $x = 0.92c$ to camber ordinate at $x = 0.5c$. (Entered as zero for an uncambered section.) Section dimensionless upper-surface and lower-surface ordinates at each of 15 standard values of dimensionless chordwise location $(x/c)_i = 0, 0.025, 0.05, 0.1, 0.2, 0.3, 0.4, 0.5, 0.6, 0.7, 0.8, 0.9, 0.95, 0.99$ and 1.0 for $1 \leq i \leq 15$. Ratio of camber ordinate at $x = 0.92c$ to camber ordinate at $x = 0.5c$. (Entered as unity for an uncambered section.)
$\delta_{2/3}$	
Data for root (centre-line) section:	
$\begin{matrix} \vdots & \vdots & \vdots \\ (x/c)_{ri} & (z_u/c)_{ri} & (z_l/c)_{ri} \\ \vdots & \vdots & \vdots \end{matrix}$	
$(z_{c92}/z_{c50})_r$	
Data for section at $\eta = 2/3$:	
$\begin{matrix} \vdots & \vdots & \vdots \\ (x/c)_{2/3i} & (z_u/c)_{2/3i} & (z_l/c)_{2/3i} \\ \vdots & \vdots & \vdots \end{matrix}$	
$(z_{c92}/z_{c50})_{2/3}$	

continuing

The following data are necessary if a fuselage is present, $CALC = 13$ to 18 , and are omitted otherwise.		
IF $CALC = 13$ to 18		
INT		Integer. If $INT = 0$ the fuselage effect on trailing-edge flap performance is neglected. If $INT = 1$ the possible effect is allowed, but decreases linearly to zero as the gap between the fuselage side and the inboard panel limit increases from $0.005s$ to $0.05s$. An interference effect is always included for a leading-edge device, whatever the value of INT , provided the inboard gap is less than $0.005s$ but is omitted otherwise.
IF $INT = 0$		Ratio of body width to wing semi-span, setting angle of wing root (centre-line) chord with respect to centre-line of fuselage (deg) (see Sketch 1.10).
W/s i_W		
IF $INT = 1$		Ratio of body width to wing semi-span, setting angle of wing root (centre-line) chord with respect to centre-line of fuselage (deg), height of wing relative to fuselage height (see Sketch 1.10).
W/s i_W h/H		

The following data are necessary if user-defined values are to be supplied. The quantities depend on the calculation procedure selected, see Table 3.1. Values are entered N_{MR} times.		
IF $CALC = 2$		User-defined values of lift coefficient at zero angle of attack and of maximum lift coefficient for basic aerofoil.
C_{L0B} C_{LmB}		
IF $CALC = 4$		User-defined value of lift coefficient at zero angle of attack for basic aerofoil.
C_{L0B}		
IF $CALC = 8$		User-defined values of lift coefficient at zero angle of attack and of maximum lift coefficient for basic wing.
C_{L0Bw} C_{LmaxB}		
IF $CALC = 10$		User-defined value of lift coefficient at zero angle of attack for basic wing.
C_{L0Bw}		
IF $CALC = 13$		User-defined value of effect of fuselage on maximum lift coefficient.
ΔC_{Lmaxf}		
IF $CALC = 14$		User-defined values of lift coefficient at zero angle of attack and of maximum lift coefficient for basic wing-fuselage combination.
C_{L0Bf} C_{LmaxBf}		
IF $CALC = 16$		User-defined value of lift coefficient at zero angle of attack for basic wing-fuselage combination.
C_{L0Bf}		

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Number of settings of high-lift system, $CALC = 1$ to 18		
$N_{\delta l}$	$N_{\delta t}$	Integers. Number of settings of leading-edge device and trailing-edge flap. If both are zero no further data are required. Two values must be entered; $N_{\delta l} = 0$ indicates no leading-edge device, $N_{\delta t} = 0$ indicates no trailing-edge flap. ($0 \leq N_{\delta l}, N_{\delta t} \leq 10$.)
High-lift system specification for aerofoil, $CALC = 1$ to 6.		
N_l		Integer. Identifies leading-edge device type. See Table 3.2. (Omitted if $N_{\delta l} = 0$.)
N_t		Integer. Identifies trailing-edge flap type. See Table 3.2. (Omitted if $N_{\delta t} = 0$.)
High-lift system specification for wing, $CALC = 7$ to 18		
N_{nl}		Integer. Identifies number of leading-edge device panels, $1 \leq N_{nl} \leq 5$. (Omitted if $N_{\delta l} = 0$.) For $CALC = 7, 8, 11, 13, 14$ or 17 , N_{nl} must be unity as only a single panel is allowed.
$(N_l)_j$	$(\eta_{il})_j$	$(\eta_{ol})_j$
N_{nt}		Integer. Identifies number of trailing-edge flap panels, $1 \leq N_{nt} \leq 5$. (Omitted if $N_{\delta t} = 0$.)
$(N_t)_j$	$(\eta_{it})_j$	$(\eta_{ot})_j$
		Integer that identifies leading-edge device type for each of N_{nl} panels, see Table 3.2, with the positions of the inboard and outboard limits of each panel as a fraction of wing semi-span, $1 \leq j \leq N_{nl}$. (Omitted if $N_{\delta l} = 0$.) Innermost panel entered first and proceeding outboard. For $CALC = 7, 8, 11, 13, 14$ or 17 , $(\eta_{ol})_1$ must be unity as the single permitted panel must extend to the wing tip.
		Integer that identifies trailing-edge device type for each of N_{nt} panels, see Table 3.2, with the positions of the inboard and outboard limits of each panel as a fraction of wing semi-span, $1 \leq j \leq N_{nt}$. (Omitted if $N_{\delta t} = 0$.) Innermost panel entered first and proceeding outboard.

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