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Computer program for prediction of aircraft lateral stability derivatives in sideslip at subsonic speeds

Associated software: ESDUpac A0025



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THE PREPARATION OF THIS DATA ITEM

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The computer program was prepared under contract to ESDU by Mr C. King.

COMPUTER PROGRAM FOR PREDICTION OF AIRCRAFT LATERAL STABILITY DERIVATIVES IN SIDESLIP AT SUBSONIC SPEEDS

CONTENTS

	Page
1. NOTATION AND UNITS	2
2. INTRODUCTION	9
3. INPUT DATA	11
3.1 General	11
3.2 Body Geometry (see Sketches 3.1 and 3.2)	11
3.3 Wing Geometry	12
3.3.1 Planform geometry (see Sketch 3.3)	12
3.3.2 Wing dihedral, section data, height and setting angle (see Sketch 3.4)	14
3.4 Fin Geometry (see Sketch 3.5)	15
3.5 Nacelle Geometry	15
3.5.1 Jet-engine nacelles (see Sketch 3.6)	16
3.5.2 Nacelles on propeller-driven aircraft (see Sketch 3.7)	16
3.6 Flap Geometry	19
3.6.1 Planform geometry (see Sketch 3.8)	19
3.6.2 Flap lift and profile drag coefficient increments (see Sketch 3.9)	20
4. OUTPUT DATA	26
4.1 Lift Coefficient	26
5. DERIVATION AND REFERENCES	27
5.1 Derivation	27
5.2 References	28
6. EXAMPLES	30
6.1 Example 1	30
6.1.1 Case (i)	30
6.1.2 Case (ii)	46
6.1.3 Case (ii) with user-defined values of ΔC_{Lff} and ΔC_{Dff}	59
6.2 Example 2	60
6.3 Example 3	70
APPENDIX A USE OF ESDUpac A9931	72
A1. Additional Notation	72
A2. Estimation of α_0	73

A3.	Estimation of ΔC_{Lff}	75
A4.	Listing of Output Files R9931, Rf 9931.	76
A4.1	File R9931	76
A4.2	File Rf 9931	78

COMPUTER PROGRAM FOR PREDICTION OF AIRCRAFT LATERAL STABILITY DERIVATIVES IN SIDESLIP AT SUBSONIC SPEEDS

SUMMARY

The construction of the input file for a program, ESDUpac A0025, which predicts aircraft lateral stability derivatives in sideslip at subsonic speeds, is described. Sketches, based on those in the Data Items from which the prediction methods have been taken, are given to aid the user in identifying geometric parameters but the underlying methods are not repeated.

The program is run online and is available via the ESDU 00025 page of the ESDU website, together with the example input files.

General information on the availability of Data Items on lateral stability derivatives can be found in ESDU Aero A.06.01.00 (Reference 9) and Aero A.07.01.00 (Reference 3).

1. NOTATION AND UNITS

The derivative notation is that proposed in ARC R&M 3562 (Hopkin, 1970) and described in ESDU 86021 (Reference 30). Coefficients and aeronormalised derivatives are evaluated in aerodynamic-body axes with origin at the aircraft centre of gravity and with the wing span as the characteristic length. The derivatives L_v , N_v and Y_v are often written as $\partial C_l / \partial \beta$, $\partial C_n / \partial \beta$ and $\partial C_Y / \partial \beta$ or $C_{l\beta}$, $C_{n\beta}$ and $C_{Y\beta}$ in other systems of notation but attention must be paid to the reference dimensions used and it is to be noted that a constant datum value of V is employed in the Hopkin system.

		SI	British
A	wing aspect ratio, b^2 / S		
a_1	wing lift-curve slope	rad^{-1}	rad^{-1}
b	wing span, see Sketch 3.3	m	ft
b_T	tailplane span, see Sketch 3.5	m	ft
C_L	lift coefficient, $(\text{lift}) / \frac{1}{2} \rho V^2 S$		
$\Delta C_{L_{fj}}$	total lift coefficient increment due to deployment of j 'th pair of flap panels, see Section 3.6		
$\Delta C_{D_{fj}}$	total profile drag coefficient increment due to deployment of j 'th pair of flap panels, see Section 3.6		
c	local wing chord	m	ft
c_{fj}	flap chord at mid-span of j 'th flap panel on each half wing, see Sketch 3.9	m	ft
c_j	wing chord at mid-span of j 'th flap panel on each half wing, see Sketch 3.9	m	ft
c_j'	extended chord due to single-slotted flap deflection at mid-span of j 'th flap panel on each half wing, see Sketch 3.9c	m	ft
c_r	root chord of exposed half wing, see Sketch 3.3	m	ft
c_{rF}	fin root chord, see Sketch 3.5	m	ft
c_t	wing tip chord, see Sketch 3.3	m	ft
c_{tF}	fin tip chord, see Sketch 3.5	m	ft
c_0	gross wing centre-line chord, see Sketch 3.2	m	ft
$d_{0,i}$	sawtooth depth at leading edge of inboard chord of i 'th trapezoidal panel on each half wing, positive for rearward shift of leading edge, see Sketch 3.3	m	ft

$d_{1,i}$	sawtooth depth at trailing edge of inboard chord of i 'th trapezoidal panel on each half wing, positive for rearward shift of trailing edge, see Sketch 3.3	m	ft
H	height of body cross-section containing quarter-chord point of wing centre-line chord, see Sketch 3.2	m	ft
h_{BF}	body cross-section height at fin root quarter-chord station, see Sketch 3.5	m	ft
h_{b1}	body cross-section height at $0.25l_b$, see Sketch 3.1	m	ft
h_{b2}	body cross-section height at $0.75l_b$, see Sketch 3.1	m	ft
h_F	height of fin, measured from fin root chord in direction normal to longitudinal axis, see Sketch 3.5	m	ft
h_{nek}	exit diameter for k 'th pair of nacelles, see Sketch 3.6	m	ft
h_{nk}	maximum cross-section height for k 'th pair of nacelles, see Sketches 3.6 and 3.7	m	ft
h_T	height of fin-mounted tailplane above fin root chord, see Sketch 3.5	m	ft
i_W	angle between wing centre-line chord and body longitudinal axis, positive leading-edge up	deg	deg
$\mathcal{L}$	rolling moment, see Sketch 1.1	N m	lbf ft
L_v	rolling moment derivative due to sideslip, $(\partial \mathcal{L} / \partial v) / \frac{1}{2} \rho V S b$		
l	longitudinal distance from body nose tip to moment reference point, see Sketch 3.1	m	ft
l_b	overall body length, see Sketch 3.1	m	ft
l_{hj}	longitudinal distance aft from moment reference point to intersection of extended flap hinge line with plane of symmetry for j 'th flap panel on each half wing, see Sketch 3.8; negative if intersection is forward of moment reference point	m	ft
l_{nk}	for k 'th pair of nacelles, overall length of jet-engine nacelle, see Sketch 3.6 or nacelle length forward of wing leading edge for propeller-driven aircraft, see Sketch 3.7	m	ft
M	Mach number		
m_{0k}	longitudinal distance of cowl leading edge forward of moment reference point for k 'th pair of nacelles, see Sketches 3.6 and 3.7	m	ft

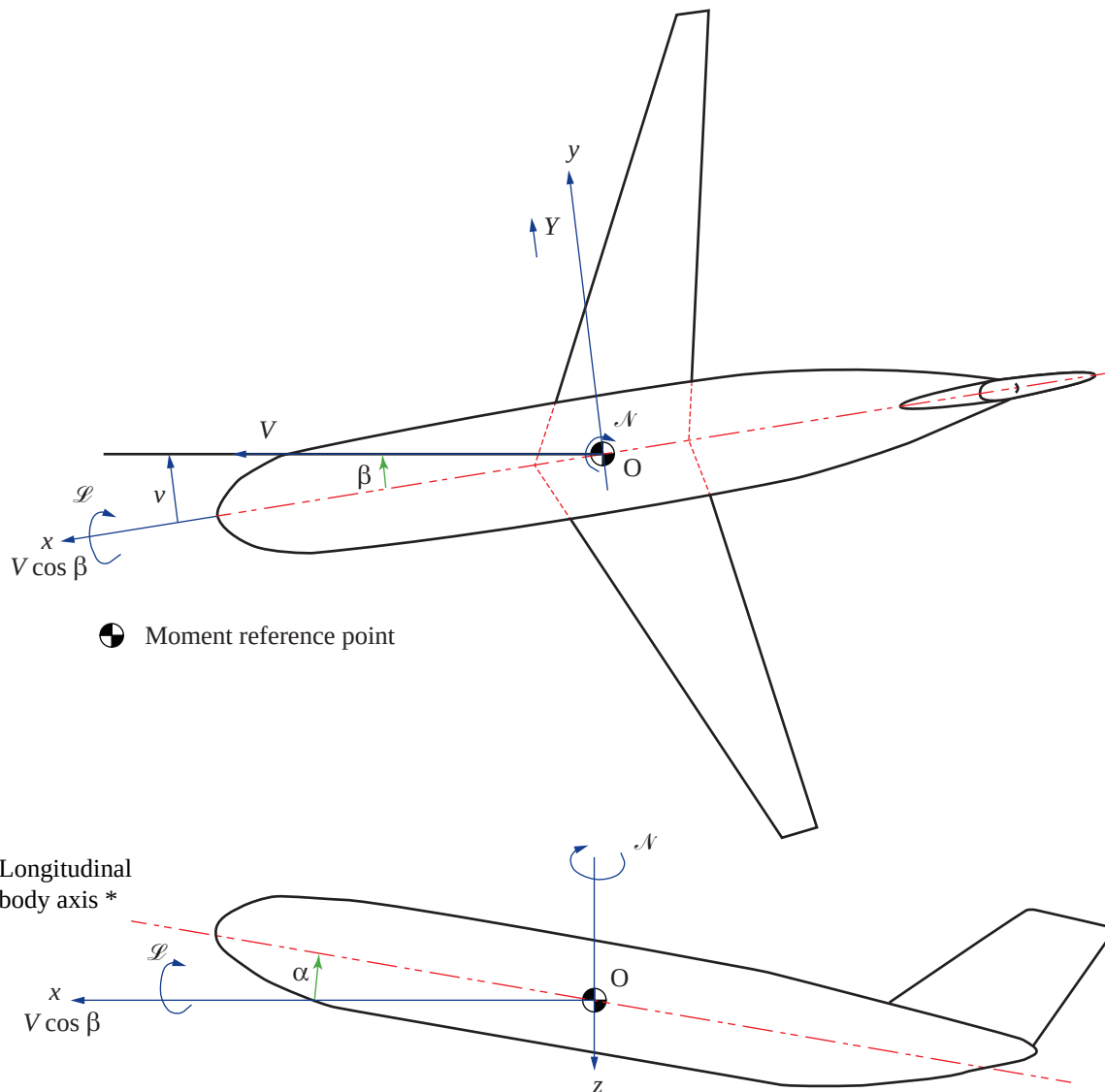
m_F	distance of fin root quarter-chord position aft of centre of gravity position, measured parallel to aircraft longitudinal body axis, see Sketch 3.5	m	ft
$\mathcal{N}$	yawing moment, see Sketch 1.1	N m	lbf ft
N_F	integer defining tailplane position, 0 = no tailplane, 1 = body mounted, 2 = fin mounted		
N_f	integer denoting number of flap panels on each half wing, $0 \leq N_f \leq 4$		
N_{fj}	integer defining flap type for j 'th flap panel, 1 = plain, 2 = split, 3 = single-slotted, 0 = user defined values of $\Delta C_{L_{fj}}$, $\Delta C_{D_{fj}}$, see Section 3.6.2		
N_n	integer denoting number of nacelles on each half wing		
N_{pan}	integer denoting number of panels representing each exposed half wing ($1 \leq N_{pan} \leq 20$), see Section 3.3.1		
N_T	integer denoting presence and type of engine nacelle; $N_T = 0$ denotes no nacelles, $N_T = 1$ denotes under-wing pylon-mounted jet-engine nacelles, $N_T = 2$ denotes wing-mounted engine nacelles for propeller-driven aircraft		
N_v	yawing moment derivative due to sideslip, $(\partial \mathcal{N} / \partial v) / \frac{1}{2} \rho V S b$		
N_T	integer denoting number of spanwise dihedral segments on each half wing, $1 \leq N_T \leq 20$, see Section 3.3.2		
NOP	integer determining output format, 1 generates full output and 0 a restricted output		
$Oxyz$	aerodynamic-body axis system, see Sketch 1.1		
R	Reynolds number based on aerodynamic mean chord of wing		
S	gross wing planform area	m ²	ft ²
S_b	area of side elevation of body	m ²	ft ²
S_x	area of body cross-section containing quarter-chord point of wing centre-line chord	m ²	ft ²
s	wing semi-span	m	ft
s_{fij} , s_{foj}	inboard, outboard spanwise limits, from plane of symmetry, of j 'th flap panel on each half wing, see Sketch 3.8	m	ft
s_i	spanwise distance from plane of symmetry of inboard chord of i 'th trapezoidal panel on each half wing, see Sketch 3.3	m	ft

s_{nk}	spanwise distance from plane of symmetry of centre line for k 'th pair of nacelles, see Sketches 3.6 and 3.7	m	ft
$s_{\Gamma m}$	spanwise distance from plane of symmetry to outboard limit of m 'th dihedral segment on each half wing, see Sketch 3.4	m	ft
t	local wing maximum section thickness	m	ft
$UNITS$	integer specifying unit system, 1 = <i>SI</i> , 2 = <i>British</i>		
V	velocity of aircraft relative to air, see Sketch 1.1	m/s	ft/s
v	sideslip velocity, see Sketch 1.1	m/s	ft/s
W	width of body cross-section containing quarter-chord point of wing centre-line chord, see Sketch 3.2	m	ft
w_{BF}	body cross-section width at fin root quarter-chord station, see Sketch 3.5	m	ft
x_r	longitudinal distance from body nose tip to leading edge of root chord of exposed half wing, see Sketch 3.3	m	ft
x_{tr}	distance of boundary layer transition point aft of wing leading edge (assumed same on wing upper and lower surfaces)	m	ft
Y	sideforce, see Sketch 1.1	N	lbf
Y_v	sideforce derivative due to sideslip, $Y_v = (\partial Y / \partial v) / \frac{1}{2} \rho V S$		
z_{crF}	height of fin root chord, measured from longitudinal body axis in direction normal to longitudinal body axis, see Sketch 3.5	m	ft
z_{nk}	for k 'th pair of nacelles, vertical distance of nacelle centre line below wing-pylon junction for jet-engine nacelle, see Sketch 3.6, or below local wing chord for nacelle on propeller-driven aircraft, see Sketch 3.7	m	ft
z_{0k}	for k 'th pair of nacelles, vertical distance of nacelle centre line below moment reference point (positive in Sketch 3.6 and negative in Sketch 3.7)	m	ft
z_W	distance of quarter-chord point of wing centre-line chord below centre of area of body cross-section, positive for low wing, see Sketch 3.4	m	ft
α	incidence of body longitudinal axis, see Sketch 1.1	deg	deg
α_0	value of α for zero wing-body lift	deg	deg
α_{0B}	value of α for zero lift of isolated body, see Section 3.2	deg	deg

α_{0rWB}	incidence of wing centre-line chord for zero lift, see ESDU 87031 (Reference 31) and 89042 (Reference 32) and Section 4.1	deg	deg
α_1, α_2	lower, upper limits of α for calculation	deg	deg
β	sideslip angle, $\sin^{-1}(v / V)$, see Sketch 1.1	rad	rad
Γ_m	dihedral angle of m 'th spanwise segment on each half wing, see Sketch 3.4	deg	deg
γ_j	angle between section chord line and line joining trailing edge of undeflected flap to mid-thickness point at hinge location for j 'th flap panel, see Sketch 3.9a	deg	deg
δ_{ff}	flap deflection angle for j 'th flap panel, see Sketch 3.9, measured in streamwise plane	deg	deg
δ_{tip}	upward wing tip deflection due to load (gives equivalent full span dihedral angle of $67.6 \delta_{tip} / s$ degrees, see ESDU Aero A.06.01.03 (Reference 7))	m	ft
θ	mean upsweep angle of rear body centre-line, see Sketch 3.1	deg	deg
Λ_{hj}	hinge-line sweep of j 'th flap panel on each half wing, see Sketch 3.8	deg	deg
$\Lambda_{0,i}$	leading-edge sweep of i 'th trapezoidal panel on each half wing, see Sketch 3.3	deg	deg
$\Lambda_{1,i}$	trailing-edge sweep of i 'th trapezoidal panel on each half wing, see Sketch 3.3	deg	deg
$\Lambda_{1/4}$	wing quarter-chord sweep, see Sketch 3.2	deg	deg
$\Lambda_{1/4F}$	sweep of fin quarter-chord line, see Sketch 3.5	deg	deg
λ	wing taper ratio (tip chord / centre-line chord)		
ρ	density of air	kg/m ³	slug/ft ³
τ	wing section trailing-edge angle	deg	deg

Subscripts

$()_b$	denotes contribution due to isolated body
$()_F$	denotes contribution due to fin/tailplane
$()_f$	denotes contribution due to flaps
$()_h$	denotes contribution due to wing-body interference arising from vertical position of wing on body
$()_n$	denotes contribution due to all nacelles
$()_{nk}$	denotes contribution due to k 'th pair of nacelles
$()_W$	denotes contribution due to wing planform
$()_{WB}$	denotes contribution due to wing-body
$()_\Gamma$	denotes contribution due to wing dihedral
$()_{90}$	denotes value at 90% chord
$()_{99}$	denotes value at 99% chord
i	integer identifying wing planform panel
j	integer identifying flap panel
k	integer identifying nacelle
m	integer identifying wing dihedral segment



Sketch 1.1 Aerodynamic-body axis system

* The longitudinal body axis is a reference axis, fixed in the body in the plane of symmetry and passing through the moment reference point. The exact direction of the axis in the plane of symmetry is conventionally determined by considerations of mid-body geometry, the axis being taken parallel to some convenient datum, see Section 1.2 of ESDU 81032 (Reference 21) and Section 1.2 of ESDU 82011 (Reference 23).

2. INTRODUCTION

This Data Item describes the construction of the input file for the program ESDUpac A0025, which evaluates the lateral stability derivatives in an aerodynamic-body axis system for an aircraft in sideslip. Values are calculated over a specified incidence range. The methods in the program are summarised in ESDU 81032 (Reference 21) and 82011 (Reference 23), which illustrate by worked examples the way in which the various contributions of aircraft components are evaluated and summed to provide total values. Tables 2.1 and 2.2 identify the individual Items from which methods have been taken for the program. The third column of each Table lists the primary Items that directly address the prediction of each component of the stability derivatives. The methods of those Items sometimes require the additional use of Items of a more general nature; for example the estimation of a wing lift-curve slope by the method of ESDU 70011 (Reference 10) is often essential. The fourth column of the Tables lists all such necessary supporting Items, the methods of which have all been incorporated in the program.

With flaps deployed, values of the lift coefficient and profile drag coefficient increments due to the deployment are required and are calculated using the Items that were available when the methods for predicting the effect of flap deployment on the lateral stability derivatives were developed*. The program allows plain, split or single-slotted flaps, divided into as many as four panels on each half wing. Lift coefficient increments are based on the full-span methods of ESDU 74011 (Reference 14), 74009 (Reference 12) and Aero F.01.01.08 (Reference 4), with ESDU 74012 (Reference 15) providing part-span correction. Profile drag coefficient increments are similarly provided for full-span flaps by ESDU 87024 (Reference 24), 74010 (Reference 13) and Aero F.02.01.06 (Reference 2), with Aero F.02.01.07 (Reference 1) providing part-span correction.

Users have the option to input their own values of flap lift and profile drag coefficient increments instead of accepting the values that would be calculated by the program. In particular, for the lift coefficient increments, empirical values for plain, split, single-slotted, double-slotted or triple-slotted flaps can be calculated by the Items described in ESDU 97002 (Reference 33), and programmed in ESDU 99031 (Reference 34), and input as user-defined values. (Example 14 in Section 7 of ESDU 99031 gives an example of the input file needed to calculate flap lift coefficients at zero incidence, and Appendix A of this Item gives the modified file appropriate to one of the worked examples of this Item.)

Section 3 details the input file construction and Section 4 describes the output format. Section 5 lists the Derivation and References, and Section 6 provides examples.

* With the exception of ESDU 87024 which was issued subsequently, and is available as an ESDUpac.

TABLE 2.1

Component	Due to	Primary method from Item No.	Supporting methods from Item Nos
$(Y_v)_{WB} (N_v)_{WB}$	wing-body combination	79006 ¹⁷	76003 ¹⁶
$(Y_v)_n (N_v)_n$	nacelles	79006 ¹⁷ and Section 3.5.2	—
$(Y_v)_F (N_v)_F$	fin (in presence of body, wing and tailplane), rudder undeflected	82010 ²²	70011 ¹⁰
$(Y_v)_f (N_v)_f$	trailing-edge flap deployment	81013 ²⁰	Aero F.01.01.08 ⁴ Aero F.02.01.06 ² and Aero F.02.01.07 ¹ Aero W.01.01.01 ⁵ Aero W.01.01.05 ⁶ 70011 ¹⁰ 74009 ¹² , 74010 ¹³ , 74011 ¹⁴ and 74012 ¹⁵ 76003 ¹⁶ 82010 ²² 87024 ²⁴ 97003 ²⁵

TABLE 2.2

Component	Due to	Primary method from Item No.	Supporting methods from Item Nos
$(L_v)_\Gamma$	wing dihedral	Aero A.06.01.03 ⁷ Aero A.06.01.09 ⁸	Aero W.01.01.01 ⁵ Aero W.01.01.05 ⁶ 76003 ¹⁶
$(L_v)_W$	wing planform	80033 ¹⁸	70011 ¹⁰ 76003 ¹⁶
$(L_v)_b$	body	73006 ¹¹	—
$(L_v)_h$	wing-body interference	73006 ¹¹	76003 ¹⁶
$(L_v)_n$	nacelles	73006 ¹¹ and Section 3.5.2	—
$(L_v)_F$	fin (in presence of body, wing and tailplane), rudder undeflected	82010 ²²	70011 ¹⁰
$(L_v)_f$	trailing-edge flap deployment	80034 ¹⁹	Aero F.01.01.08 ⁴ Aero W.01.01.01 ⁵ Aero W.01.01.05 ⁶ 70011 ¹⁰ 74009 ¹² , 74011 ¹⁴ and 74012 ¹⁵ 76003 ¹⁶ 97003 ²⁵

This page Amendment A

3. INPUT DATA

Table 3.1 lists, in order, the parameters required in the input file for the ESDUpac. They are described in Sections 3.1 to 3.6.

3.1 General

At the top of the input file there is provision for three lines of text 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 line left blank. These lines are repeated in the output file.

An integer *UNITS* is then set to 1 or 2 to define the unit system; 1 = *SI* units, 2 = *British* units. Within the program all calculations are dimensionless and the unit system is irrelevant. In the output only the wing planform area, *S*, and span are dimensional.

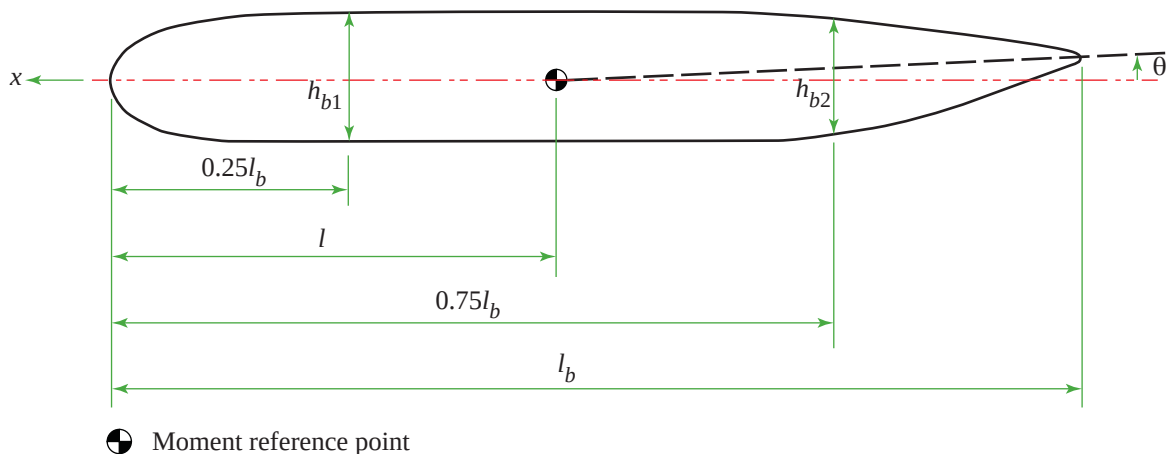
These are followed by the Mach number, *M*, the Reynolds number based on the aerodynamic mean chord of the wing, *R*, and the lower and upper limits of the incidence range for the calculation, α_1 and α_2 , in degrees, entered as a pair of values on one line separated by one or more spaces. The program calculates values at half degree intervals between α_1 and α_2 . The final entry in this Section is the integer *NOP* for which value of unity produces a full output and a value of zero a restricted output, see Section 4.

3.2 Body Geometry (see Sketches 3.1 and 3.2)

For the body, the necessary parameters are the overall body length, l_b , the distance from the nose tip to the moment reference point, *l*, the body cross-section heights, h_{b1} and h_{b2} , at $0.25l_b$ and $0.75l_b$ respectively, see Sketch 3.1, the area of the side elevation of the body, S_b , and the width, *W*, and height, *H*, of the body cross-section containing the quarter-chord point of the wing centre-line chord, see Sketch 3.2.

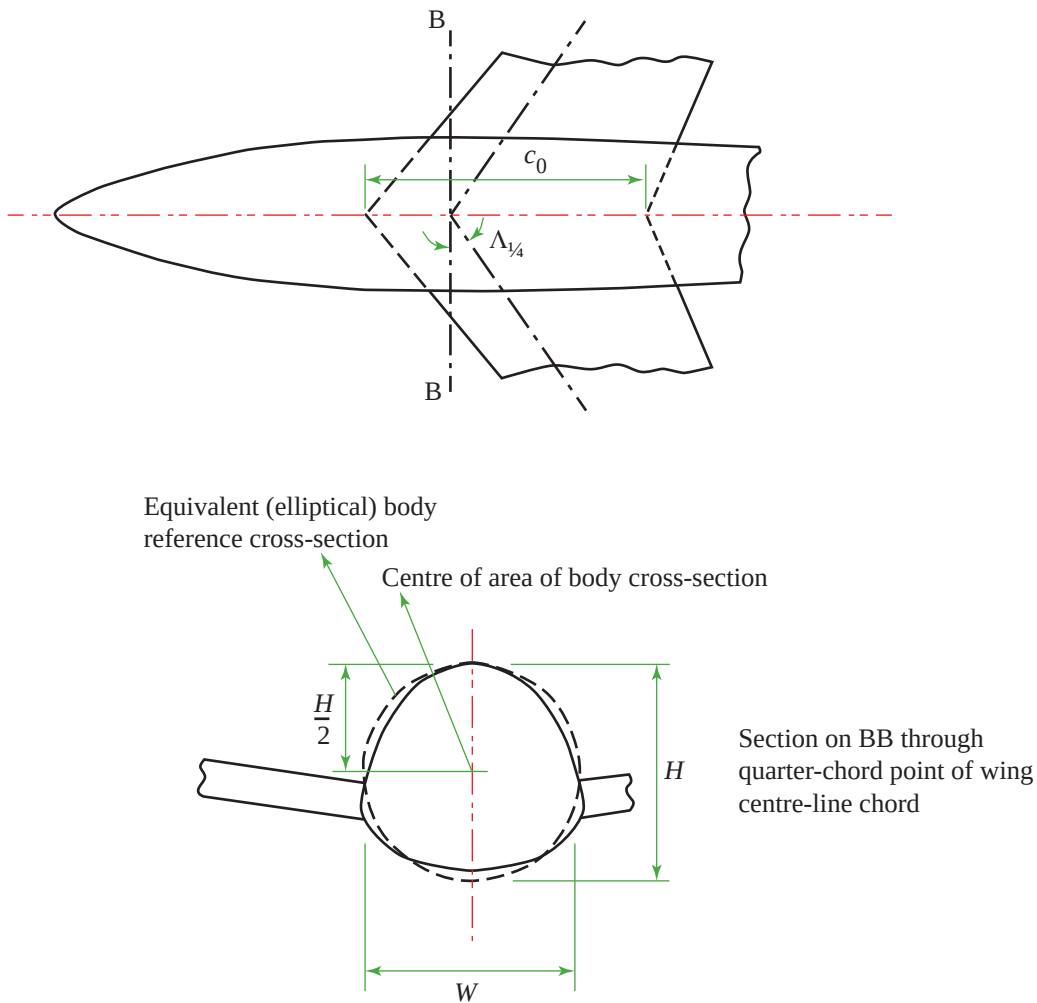
For bodies having a cross-section that is not circular or elliptical, *H* is taken as $(4 / \pi W)S_x$, where S_x is the area of the body cross-section containing the quarter-chord point of the wing centre-line chord. The cross section is thereby represented as an ellipse (or circle) of height *H* and width *W* which is assumed to be positioned with its centre of area coincident with that of the true cross section.

Finally, the angle of incidence of the body longitudinal axis for the isolated body to be at zero lift, α_{0B} , is entered in degrees. An experimental value may be used if available; otherwise the recommendation of ESDU 81032 is to employ a value of $\theta/2$ where θ is the mean upsweep angle of the rear body centre-line.



Sketch 3.1 Body geometry

This page Amendment A



Sketch 3.2 Body cross-section geometry

3.3 Wing Geometry

3.3.1 Planform geometry (see Sketch 3.3)

The next group of entries specifies the wing planform.

A gross wing planform that is straight-tapered is required for the calculation methods in the program^{*}. To obtain this each exposed half wing is treated as one or more trapezoidal panels.

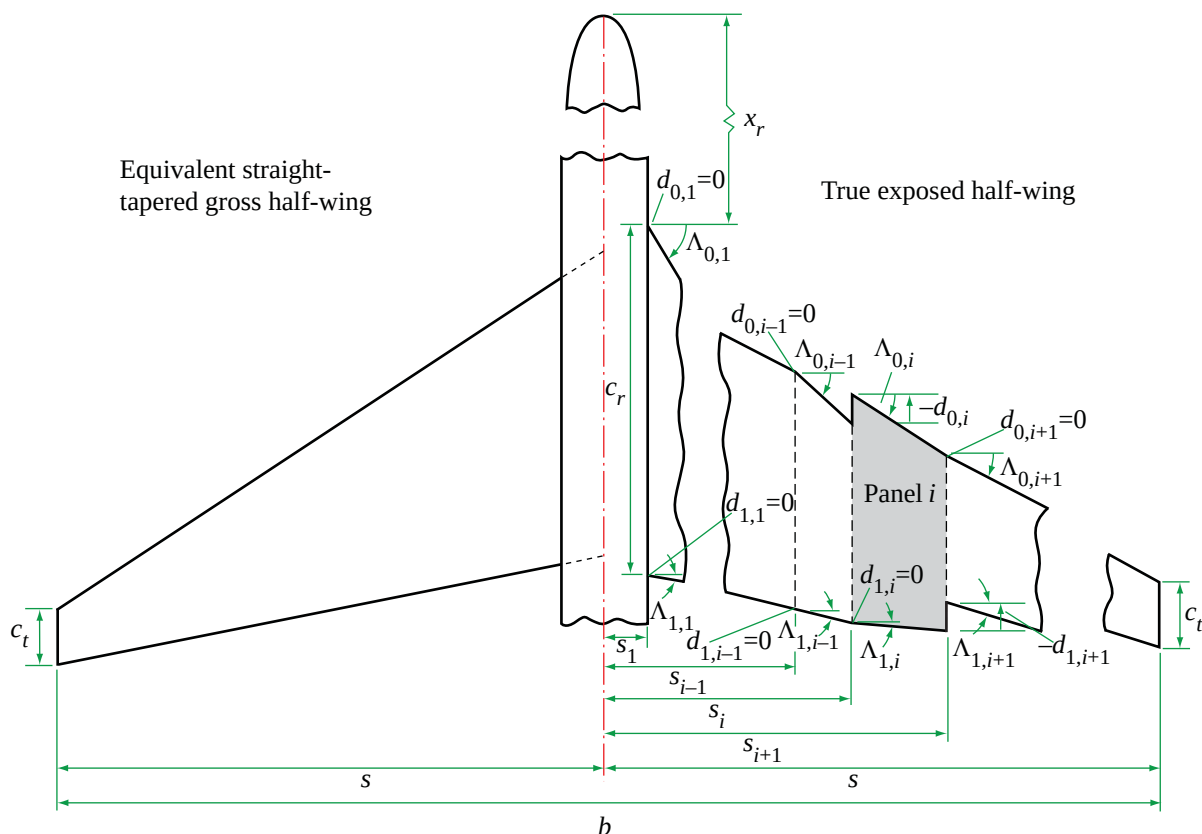
First, to dimension and locate the wing, the span, b , the longitudinal distance of the leading edge of the root chord of the exposed half wing aft of the body nose tip, x_r , and the length of that root chord, c_r , are entered.

^{*} Except for the methods for the flap contribution, $(Y_v)_f$ and $(N_v)_f$, which allow for the true planform geometry, see Section 3.6.

The number of panels forming each exposed half wing is identified by inspection and entered as the integer N_{pan} . If N_{pan} is unity the wing is straight-tapered. If there are discontinuities in sweep in the leading or trailing edges due to cranks or sawteeth, the spanwise positions of the discontinuities divide the exposed wing naturally into a number of panels. A curved wing might be represented as one with multiple cranks, if care is taken to ensure that the exposed areas of the two wings are the same.

Each panel is defined by specifying the spanwise distance from the plane of symmetry of its inboard chord, s_i , and then its leading-edge sweep angle, $\Lambda_{0,i}$, and the sawtooth depth at the leading edge, $d_{0,i}$, entered as a pair of values on one line separated by one or more spaces, followed by the trailing-edge angle, $\Lambda_{1,i}$, and the sawtooth depth at the trailing edge, $d_{1,i}$, entered as a pair of values on a second line, see Table 3.1. Positive sawtooth depths denote a rearward movement of the leading or trailing edge; they will be zero at a simple crank, and for the innermost panel it is only meaningful to have $d_{0,1} = d_{1,1} = 0$.

From this information the program uses the method of ESDU 76003 (Reference 16) to construct a straight-tapered gross wing that has the same exposed area, span and tip chord as the true wing. If $N_{pan} = 1$, this equivalent wing is identical to that which would be produced by extending the leading and trailing edges of each exposed half wing to the plane of symmetry.



Sketch 3.3 Wing planform geometry

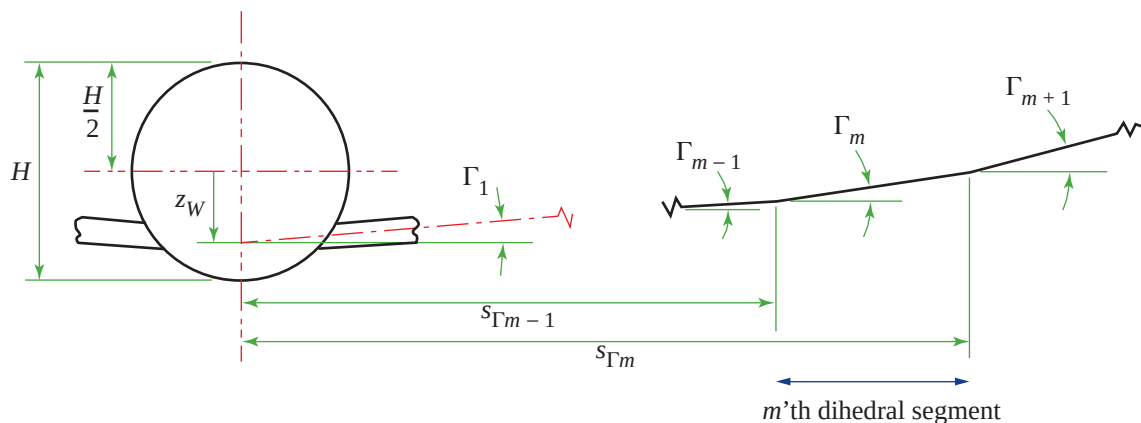
3.3.2 Wing dihedral, section data, height and setting angle (see Sketch 3.4)

Eight further parameters associated with the wing are now needed.

The contribution of geometric (rigid wing) dihedral to rolling moment, in the most general case, requires the representation of the wing as a number of spanwise segments, over each of which the dihedral angle is constant, see Sketch 3.4. (The definition of these segments is independent of the trapezoidal panels representing the wing planform.) The integer N_Γ is entered to specify the number of segments on each half wing followed by N_Γ entries of $s_{\Gamma m}$, to denote the spanwise distance from the plane of symmetry to the outboard limit of the m 'th segment, and Γ_m , the dihedral angle of the m 'th segment in degrees. For the simple case of constant full-span dihedral $N_\Gamma = 1$, $s_{\Gamma m} = s$ and $\Gamma_m = \Gamma_1$, the value of the constant dihedral angle. In addition, the program allows for the effect of flexible wing deflection and the next entry is the upward wing tip deflection under load, δ_{tip} , which the program translates into an effective increase in full-span dihedral of $67.6 \delta_{tip} / s$ degrees. The evaluation of the dihedral contribution requires an estimate of a wing sectional lift-curve slope and so there follow entries for the section parameters, taken at mid-semispan, say, for the wing maximum thickness to chord ratio, t/c , the wing thickness to chord ratios at 90% and 99% chord, $(t/c)_{90}$ and $(t/c)_{99}$, the section trailing-edge angle, τ , in degrees and the distance of the boundary-layer transition point aft of the wing leading edge as a fraction of local chord, x_{tr}/c , assumed the same on wing upper and lower surfaces.

For the calculation of wing-body interference, the distance of the quarter-chord point of the wing centre-line chord below the centre of area of the body cross-section, z_W , is needed (see Sketch 3.4 and the comments on the definition of body cross-section shape in Section 3.2).

Finally, α_0 , in degrees, the incidence of the body longitudinal axis at zero lift is required for the calculation of the lift coefficient, see Section 4.1.



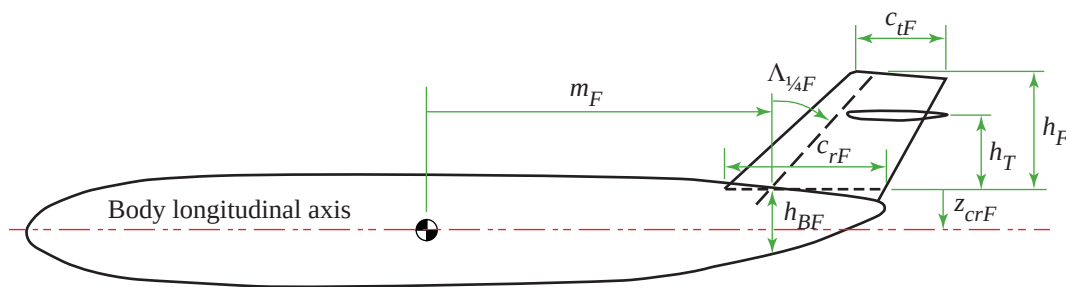
z_W is defined in the cross-section through the quarter-chord point of the wing centre-line chord, and is positive for a low wing.

Sketch 3.4 Wing height and dihedral geometry

3.4 Fin Geometry (see Sketch 3.5)

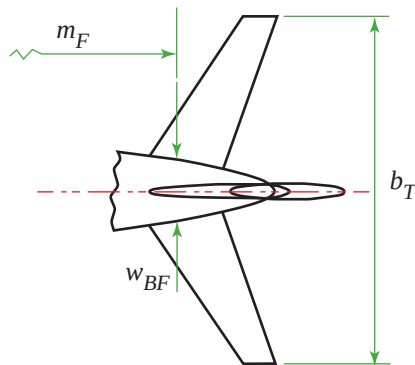
The entries which concern the fin/tailplane assembly vary slightly depending on the tailplane position. The first entry is the integer N_F , where $N_F = 0$ means there is no tailplane, $N_F = 1$ that it is body mounted and $N_F = 2$ that it is fin mounted. In all cases the following parameters are necessary: the longitudinal distance from the moment reference point to the fin root quarter-chord point, m_F , the body cross-section height and width at the fin root quarter-chord station, h_{BF} and w_{BF} , the distance of the moment reference point below the fin root chord, z_{crF} (positive when the moment reference point is below the fin root chord), the height of the fin from its root chord, h_F , the fin root chord, c_{rF} , tip chord, c_{tF} , and quarter-chord sweep, $\Lambda_{1/4F}$.

If a tailplane is present its span, b_T , is entered and, if it is mounted on the fin also its height above the fin root chord, h_T .

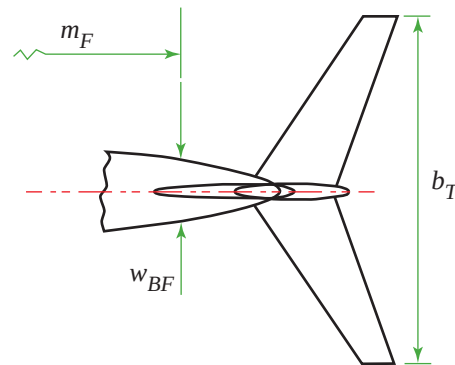


⊗ Moment reference point

For a body-mounted tailplane no value is entered for h_T



Body-mounted tailplane



Fin-mounted tailplane

m_F is measured to the quarter-chord point of the fin root chord

Sketch 3.5 Fin and tailplane geometry

3.5 Nacelle Geometry

The integer N_T is entered to denote the presence and type of nacelles. If N_T is zero there are no nacelles and no further data are needed. If $N_T = 1$ under-wing pylon-mounted jet-engine nacelles are present and if $N_T = 2$ wing-mounted engine nacelles for a propeller-driven aircraft are assumed. The integer N_n follows to indicate the number of nacelles on each half wing.

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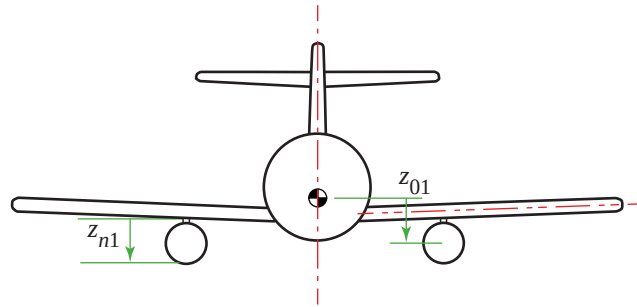
3.5.1 Jet-engine nacelles (see Sketch 3.6)

The program includes the methods from ESDU 73006 (Reference 11) and 79006 (Reference 17) that allow the influence of under-wing pylon-mounted nacelles to be calculated.

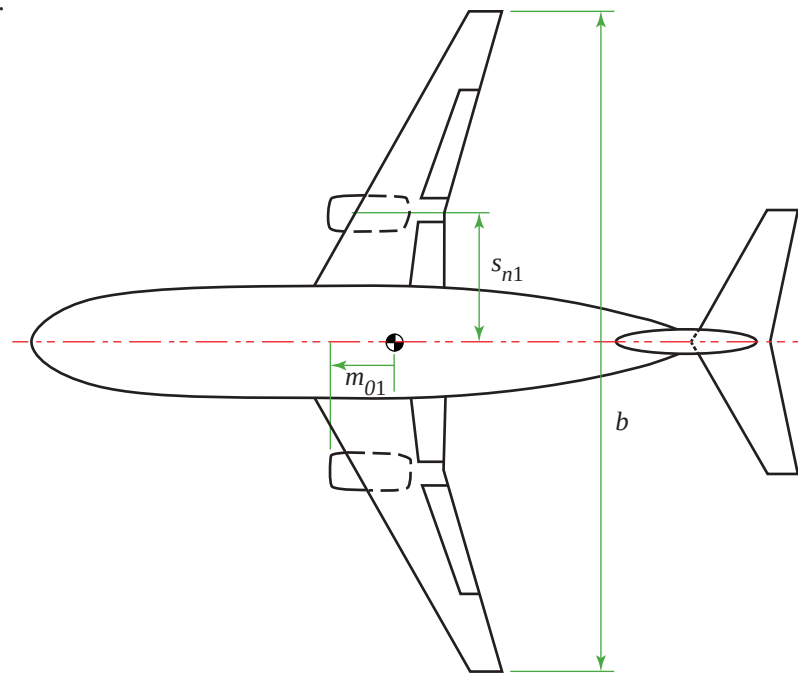
For each pair of nacelles (assumed symmetrically placed about the body centre-line, one to port and one to starboard), $1 \leq k \leq N_n$, the necessary data are the nacelle length, l_{nk} , the longitudinal distance of the nacelle leading edge ahead of the moment reference point, m_{0k} , the spanwise distance from the plane of symmetry to the nacelle centre-line, s_{nk} , the maximum nacelle cross-section height, h_{nk} , and exit diameter, h_{nek} . (With regard to h_{nek} it is to be noted that the prediction methods were based on an analysis of wind-tunnel test models fitted with free-flow nacelles.) Finally the vertical distances of the nacelle centre-line below the wing-pylon junction, z_{nk} , and below the moment reference point, z_{0k} , are required.

3.5.2 Nacelles on propeller-driven aircraft (see Sketch 3.7)

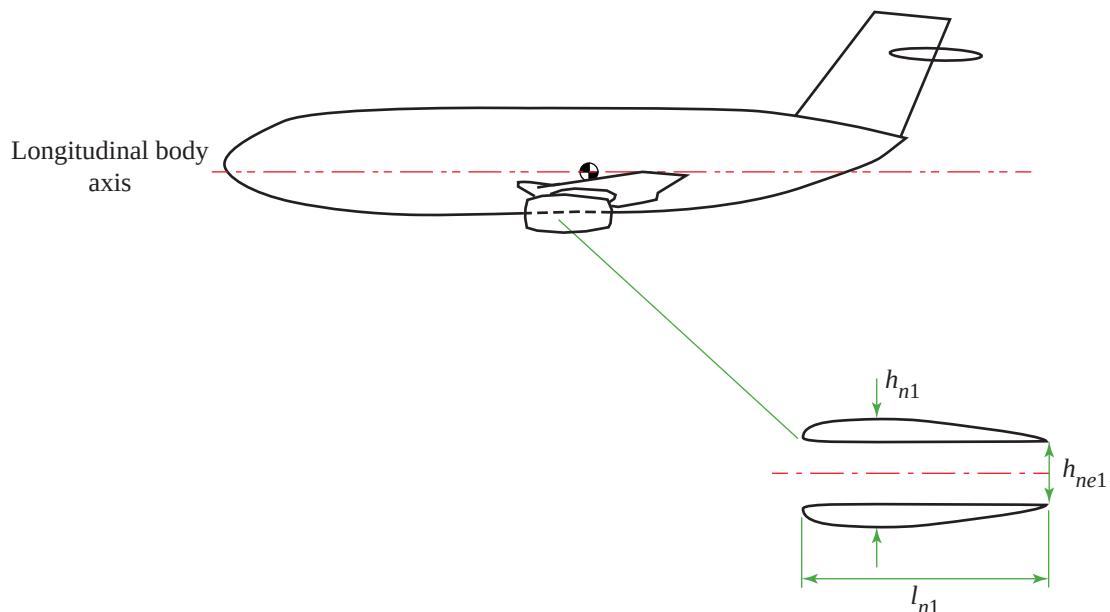
ESDU TM 178 (Reference 26) gives a tentative method for calculating the effect of wing-mounted engine nacelles of propeller-driven aircraft on lateral stability derivatives in sideslip. (Note that ESDU 06012 (Reference 35) gives the influence of propeller slipstream on aircraft rolling moment due to sideslip.) As explained in TM 178, Reference 29 suggests a method that, with slight modification, can be employed with the same input parameters as the jet-engine nacelles but with h_{nek} omitted, l_{nk} taken as the longitudinal distance of the cowl forward of the wing leading edge, and z_{nk} as the vertical distance of the nacelle centre-line below the local wing chord.



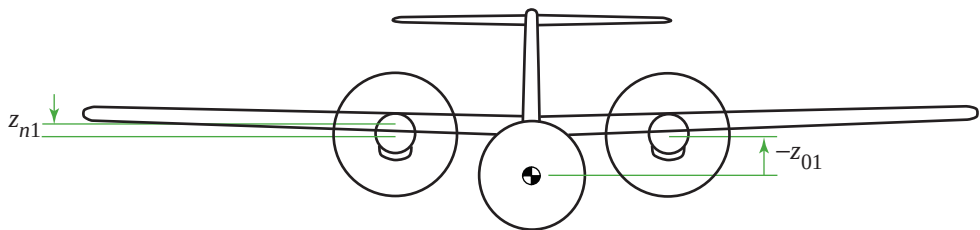
Shown for $N_n = 1$.
 $z_{nk} = z_{n1}$ etc.



⊙ Moment reference point

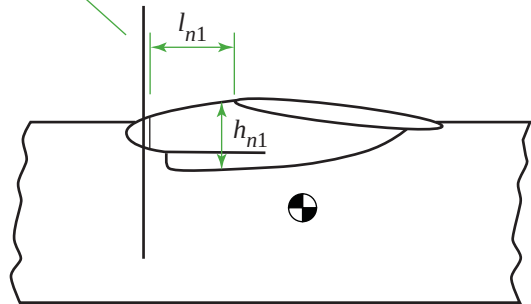
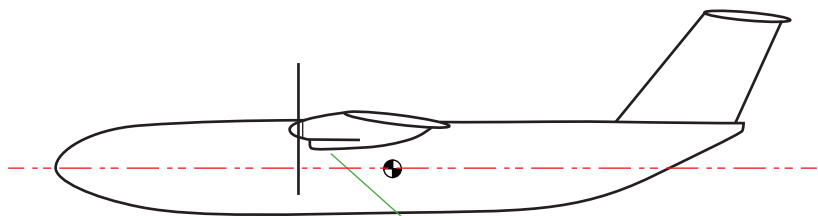
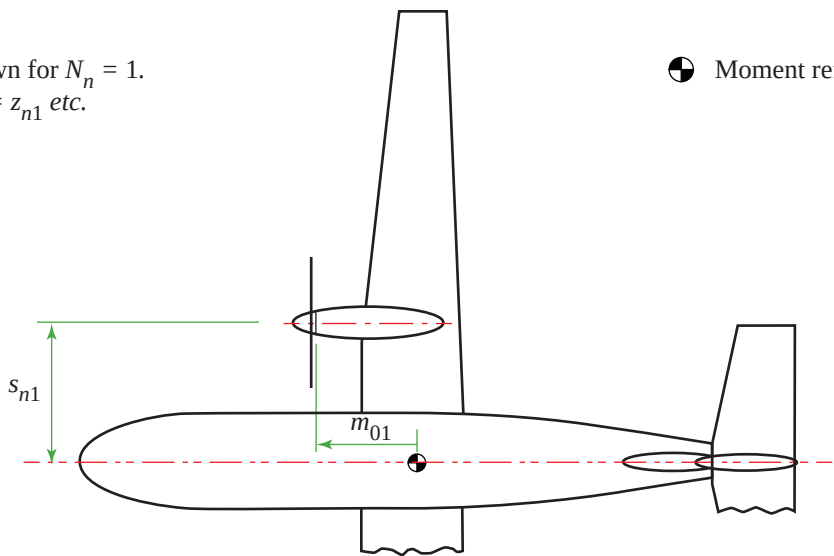


Sketch 3.6 Geometry for jet engine nacelles



Shown for $N_n = 1$.
 $z_{nk} = z_{n1}$ etc.

● Moment reference point



Sketch 3.7 Geometry for nacelles on propeller-driven aircraft

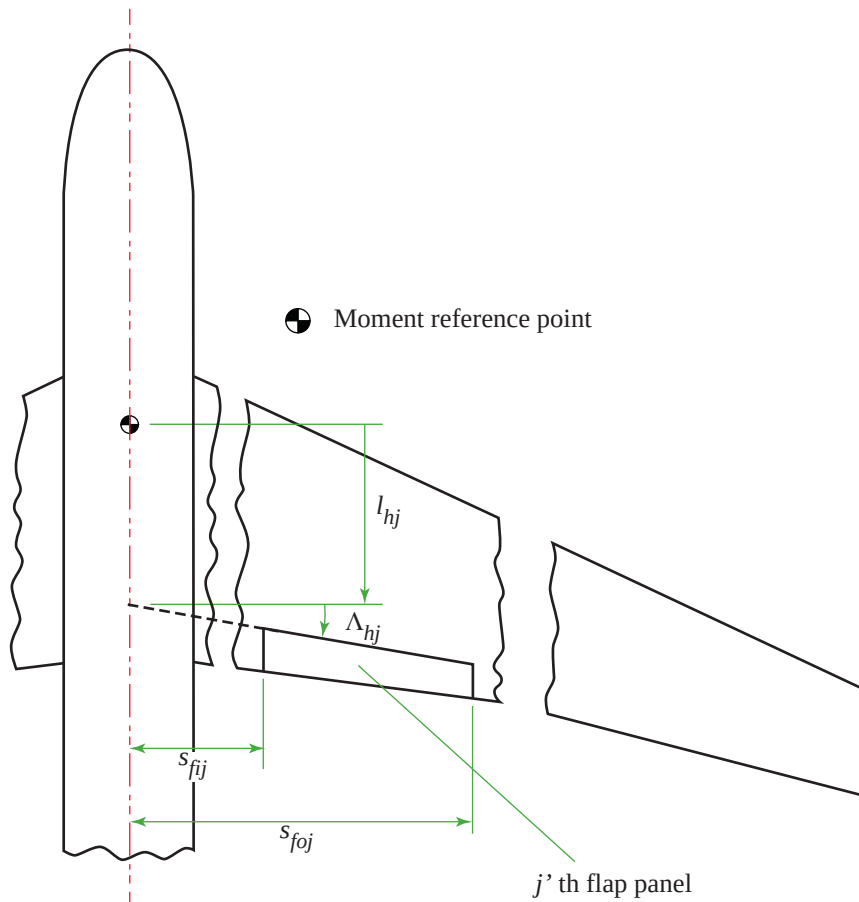
3.6 Flap Geometry

If flaps are deployed, the integer N_f is entered to denote the number of flap panels on each half wing. The maximum value of N_f is 4. Otherwise N_f must be set to zero to indicate that there are no flaps, in which case no other data are needed.

3.6.1 Planform geometry (see Sketch 3.8)

With flaps present, for each pair of panels (assumed symmetrically placed about the wing-body centre-line, one to port and one to starboard), $1 \leq j \leq N_f$, an integer N_{fj} is entered to denote the flap type. A value of zero indicates that the flap type is unspecified and that user-defined values will be entered for the flap lift and profile drag coefficient increments, $\Delta C_{L_{fj}}$ and $\Delta C_{D_{fj}}$, see Section 3.6.2. If the program is to calculate the increments, $N_{fj} = 1$ specifies a plain flap, $N_{fj} = 2$ a split flap and $N_{fj} = 3$ a single-slotted flap. In all four cases, after each N_{fj} , the inboard and outboard spanwise limits, from the plane of symmetry, of each panel, s_{fij} and s_{foj} , are entered as a pair of values on one line. These are followed by the hinge-line sweep, Λ_{hj} , and the longitudinal distance aft from the moment reference point to the intersection of the extended flap hinge line with the plane of symmetry, l_{hj} , see Table 3.1. Flaps of different type may be used in a single calculation.

If the flap coefficient increments are calculated by the program, the body interference effect that can occur is calculated by the method of ESDU 97003 (Reference 25), which requires that the inboard flaps fit closely to the body side and defines the fit as being such that $[(W/2) - s_{fi1}] / s < 0.05$.

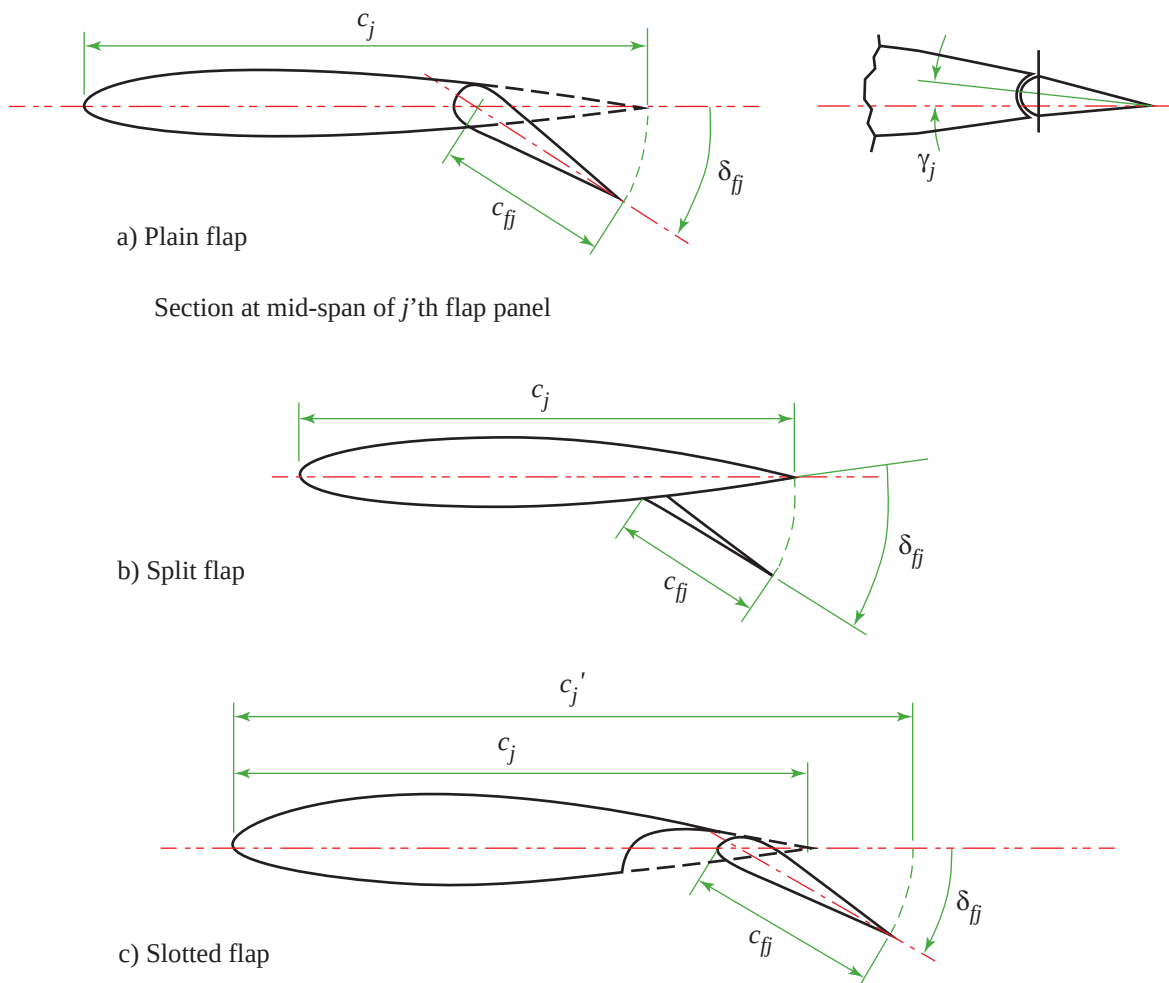


Sketch 3.8 Flap panel geometry

3.6.2 Flap lift and profile drag coefficient increments (see Sketch 3.9)

If $N_{ff} = 0$, values of ΔC_{Lff} and ΔC_{Dff} for each panel follow the flap planform parameters detailed in Section 3.6.1. Otherwise flap section geometry at the mid-span of each panel is needed. In all cases the flap chord to wing chord ratio, c_{ff}/c_j , and the flap deflection, δ_{ff} , measured streamwise in degrees, are needed. For split flaps no more data are required. However, for the other two types, additional data are necessary. For plain flaps, the angle between the section chord line and the line joining the trailing edge of the undeflected flap to the mid-thickness point at the hinge location (see Sketch 3.9a), γ_j , in degrees is required. For uncambered sections $\gamma_j = 0$. For single-slotted flaps, the extended chord (see Sketch 3.9c) to wing chord ratio is needed.

The program does not treat contributions from leading-edge high-lift devices. However, leading-edge slats that extend over most of the wing leading-edge contribute, typically, between 0.005 and 0.010 to L_v (see ESDU 80034 (Reference 19)). When a case with flaps deployed is treated, the program includes in the output a message to that effect as a reminder to the user. The small effect of slats on Y_v and N_v can be treated by employing user-defined values of ΔC_{Lff} and ΔC_{Dff} and including the slat contribution to lift coefficient in ΔC_{Lff} and the slat contribution to profile drag coefficient in ΔC_{Dff} (see ESDU 81013 (Reference 20)).



Sketch 3.9 Flap section geometry

TABLE 3.1

General, see Section 3.1.	
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 a 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.
UNITS	Integer determining units; 1 = <i>SI</i> units, 2 = <i>British</i> units.
M	Mach number.
R	Reynolds number based on aerodynamic mean chord of wing
$\alpha_1 \ \alpha_2$	Lower and upper limits of incidence of body longitudinal axis (deg), as pair of values on one line separated by one or more spaces. The program calculates values at half degree intervals between α_1 and α_2 .
NOP	Integer determining output format; 1 generates full output, 0 a restricted output.

continuing

Body geometry, see Section 3.2.	
l_b	Overall body length.
l	Longitudinal distance from nose tip to moment reference point.
h_{b1}	Body cross-section height at $0.25l_b$.
h_{b2}	Body cross-section height at $0.75l_b$.
S_b	Area of side elevation of body.
W	Width of body cross-section containing quarter-chord point of wing centre-line chord.
H	Height of body cross-section containing quarter-chord point of wing centre-line chord.
α_{0B}	Value of α for isolated body to be at zero lift.

continuing

Wing planform geometry, see Section 3.3.1.	
b	Wing span.
x_r	Longitudinal distance from body nose tip to leading edge of root chord of exposed half wing.
c_r	Root chord of exposed half wing.
N_{pan}	Integer. Number of trapezoidal panels representing each half wing. ($1 \leq N_{pan} \leq 20$). Note that the panels are numbered from 1 to N_{pan} successively moving outboard from the body.
s_i $\Lambda_{0,i} \quad d_{0,i}$ $\Lambda_{1,i} \quad d_{1,i}$	Spanwise distance from plane of symmetry of inboard chord of i'th trapezoidal panel. Leading edge sweep (deg) and sawtooth depth at leading edge of inboard chord for i'th trapezoidal panel, as pair of values on one line separated by one or more spaces. Positive $d_{0,i}$ denotes rearward movement of leading edge. Trailing edge sweep (deg) and sawtooth depth at trailing edge of inboard chord for i'th trapezoidal panel, as pair of values on one line separated by one or more spaces. Positive $d_{1,i}$ denotes rearward movement of trailing edge.

continuing

Additional wing geometry, see Section 3.3.2.	
N_Γ	Integer. Number of wing segments defining geometric dihedral on each half wing, ($1 \leq N_\Gamma \leq 20$). Note that the segments are numbered from 1 to N_Γ successively moving outboard from wing centre-line.
$s_{\Gamma m} \quad \Gamma_m$ For $1 \leq m \leq N_\Gamma$	Spanwise distance from plane of symmetry of outboard limit of m 'th segment on each half-wing ($= b/2$ for $m = N_\Gamma$). Dihedral angle (deg) for m 'th segment.
δ_{tip}	Wing tip deflection under load.
t / c	Wing maximum thickness/chord ratio.
$(t / c)_{90}$	Wing thickness/chord ratio at 90% chord.
$(t / c)_{99}$	Wing thickness/chord ratio at 99% chord.
τ	Wing section trailing-edge angle (deg).
x_{tr} / c	Distance of boundary layer transition point aft of leading edge as fraction of local chord, assumed the same on both upper and lower surfaces.
z_W	Distance of quarter-chord point of wing centre-line chord below body centre-line, positive for low wing.
α_0	Incidence of body longitudinal axis when the wing-body combination is at zero lift (deg), ($\alpha_{0rWB} - i_W$).

continuing