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Estimation of sideforce, yawing moment and rolling moment derivatives due to rate of roll for complete aircraft at subsonic speeds

Associated software: ESDUpac A8510



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ESTIMATION OF SIDEFORCE, YAWING MOMENT AND ROLLING MOMENT DERIVATIVES DUE TO RATE OF ROLL FOR COMPLETE AIRCRAFT AT SUBSONIC SPEEDS

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ESTIMATION OF SIDEFORCE, YAWING MOMENT AND ROLLING MOMENT DERIVATIVES DUE TO RATE OF ROLL FOR COMPLETE AIRCRAFT AT SUBSONIC SPEEDS

1. NOTATION AND UNITS (see Sketch 1.1)

The derivative notation used is that proposed in ARC R&M 3562 (Hopkin, 1970) and described in ESDU 86021. Coefficients and aero-normalised 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 Y_p , N_p and L_p are often written as C_{Yp} , C_{np} and C_{lp} in other systems of notation, but attention must be paid to the reference dimensions used. In particular, in forming C_{Yp} , C_{np} and C_{lp} differentiation of C_Y , C_n and C_l may be carried out with respect to $pb/2V$ not pb/V as implied in the Hopkin system. It is also to be noted that a constant datum value of V is employed by Hopkin.

This Item makes use of several other Items which have been produced at different times over a period of many years. Although the nomenclature in these Items is consistent for the important parameters such as stability derivatives, it involves some variation for the less significant parameters. Because of this, and to avoid repetition, the Notation given here is limited to the major quantities appearing in the main text of this Item and to quantities not appearing in other Items. When referred to the method in another Item the user should consult the Notation at the front of that particular Item before carrying out any calculations.

		SI	British
A	aspect ratio		
a_1	lift-curve slope	rad^{-1}	rad^{-1}
$(a_1)_{C_L = 0}$	value of a_1 at $C_L = 0$	rad^{-1}	rad^{-1}
b	wing span	m	ft
C_D	drag coefficient, $D/\frac{1}{2}\rho V^2 S$		
C'_D	viscous drag coefficient, $C_D - C_L^2/\pi A$		
C_L	lift coefficient, $L/\frac{1}{2}\rho V^2 S$		
ΔC_{Lf}	lift coefficient increment due to flap deployment		
C_l	rolling moment coefficient, $\mathcal{L}/\frac{1}{2}\rho V^2 S b$		
C_n	yawing moment coefficient, $\mathcal{N}/\frac{1}{2}\rho V^2 S b$		
C_Y	sideforce coefficient, $Y/\frac{1}{2}\rho V^2 S$		
D	drag	N	lbf
L	lift	N	lbf

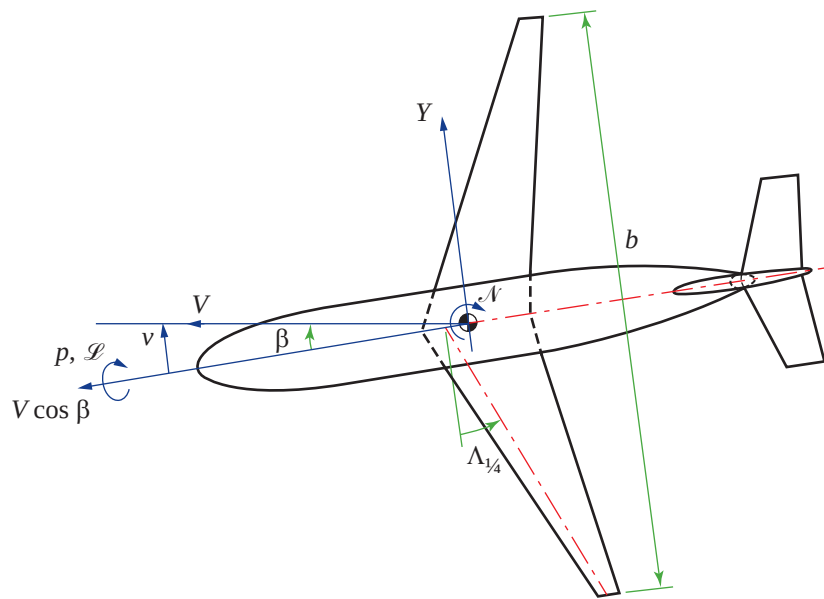
$\mathcal{L}$	rolling moment	N m	lbf ft
L_p	rolling moment derivative due to rate of roll, $L_p = (\partial \mathcal{L} / \partial p) / \frac{1}{2} \rho V S b^2$		
M	Mach number		
$\mathcal{N}$	yawing moment	N m	lbf ft
N_p	yawing moment derivative due to rate of roll, $N_p = (\partial \mathcal{N} / \partial p) / \frac{1}{2} \rho V S b^2$		
p	rate of roll	rad/s	rad/s
S	wing planform (reference) area	m ²	ft ²
s	wing semi-span	m	ft
V	velocity of aircraft relative to air	m/s	ft/s
v	sideslip velocity	m/s	ft/s
Y	sideforce	N	lbf
Y_p	sideforce derivative due to rate of roll, $Y_p = (\partial Y / \partial p) / \frac{1}{2} \rho V S b$		
α	angle of attack	degree	degree
β	angle of sideslip	degree	degree
γ	wing dihedral angle	degree	degree
κ	parameter, $\beta(a_{10})_M / 2\pi$		
Λ_0	wing leading-edge sweep angle	degree	degree
$\Lambda_{1/4}$	wing quarter-chord sweep angle	degree	degree
$\Lambda_{1/2}$	wing half-chord sweep angle	degree	degree
λ	ratio of wing tip chord to wing centre-line chord		
ρ	density of air	kg/m ³	slug/ft ³

Additional symbols

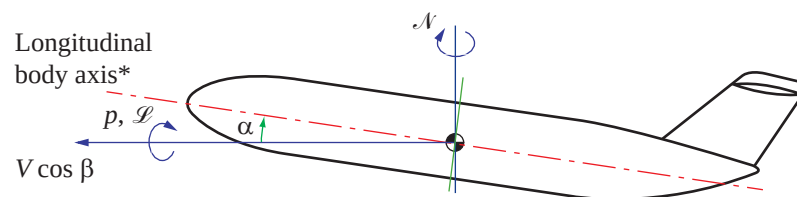
$()_F$	denotes component due to fin
$()_f$	denotes component due to trailing-edge flap deployment at constant C_L
$()_M$	denotes parameter at Mach number M
$()_T$	denotes component due to tailplane

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- $()_w$ denotes component due to wing planform
- $()_\Gamma$ denotes component due to wing dihedral
- $()_\varepsilon$ denotes component due to wing twist



⊙ Moment reference centre



Sketch 1.1 Sign conventions

* The longitudinal body axis is a reference axis fixed in the body in the plane of symmetry. The exact direction of the axis in that plane is conventionally determined by considerations of mid-body geometry.

2. INTRODUCTION

An aircraft's sideforce, yawing moment and rolling moment derivatives due to rate of roll, Y_p , N_p and L_p are normally estimated by calculating the individual contributions of the major components of the aircraft and adding together the part derivatives so obtained. For aircraft at subsonic speeds, separate Items on the various part derivatives have been issued over a number of years. This Item demonstrates how the methods in those separate Items may be combined and illustrates the overall accuracy of prediction by comparing estimated values with wind-tunnel and flight-test data for complete configurations.

Table 2.1 lists the major components of Y_p , N_p and L_p and the Items from which they may be estimated. The total values of the derivatives are obtained by evaluating each component at the same angle of attack and summing the results.

For Y_p the dihedral contribution is important at low α and C_L , because the wing planform and fin contributions are then small and the magnitude of $(Y_p)_\Gamma$ is large and sufficient to determine the sign of Y_p . As α and C_L increase the dihedral contribution remains constant but the planform and fin contributions become increasingly important.

The derivative N_p is determined mainly by the wing planform and fin contributions. The dihedral contribution is very small, and is insignificant except at low α and C_L when the planform and fin contributions are also small.

In general L_p is completely dominated by the wing planform contribution and for a rapid estimate it is sufficient only to calculate $(L_p)_w$.

Comparisons between experimental and predicted values of the rate of roll derivatives are discussed in Section 3. The Derivation is given in Section 4. Section 5 contains a detailed worked example that demonstrates the calculation of the component parts of the derivatives for a particular aircraft and their subsequent combination. The chosen aircraft is that used to demonstrate the calculation of lateral stability derivatives due to sideslip in ESDU 81032 (Derivation 37) and 82011 (Derivation 38) and those due to rate of yaw in ESDU 84002 (Derivation 40).

In Section 5 a separate subsection is devoted to the estimation of each major component. Because each subsection contains information and guidance that is additional to that contained in the basic Items listed in Table 2.1, it is useful to refer to the appropriate one when using those Items. In particular a correction for $(L_p)_w$ at high C_L is given in Section 5.3.2.

Details of a computer program for the present method, ESDUpac A8510, are given in Appendix A.

TABLE 2.1

Component	Due to	Calculated from Item No.
$(Y_p)_w, (N_p)_w$	Wing planform	81014 ³⁶
$(L_p)_w$	Wing planform	Aero A.06.01.01 ³¹
$(Y_p)_\Gamma, (N_p)_\Gamma, (L_p)_\Gamma$	Wing dihedral	85006 ⁴¹
$(Y_p)_F, (N_p)_F, (L_p)_F, (L_p)_T$	Fin and tailplane	83006 ³⁹
ESDU Aero A.06.01.00 ³² and Aero A.07.01.00 ²⁹ give brief introductions to the various components of the yawing and rolling moment stability derivatives. (Derivation numbers are given as indices.)		

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3. COMPARISONS WITH EXPERIMENT

The wind-tunnel and flight-test data in Derivations 1 to 25 have been used to assess the accuracy of prediction of the derivatives Y_p , N_p and L_p . Data have been studied for a wide variety of aircraft types representing civil transport aircraft, high performance combat aircraft and light and general aviation aircraft, together with results for a number of simpler wind-tunnel models of the type employed to study the effect of systematic variations of geometric parameters. Those data have been taken mostly from low-speed tests of clean-wing configurations with no high-lift devices deployed. Analysis of experimental data for other conditions includes a limited number of results from tests at high subsonic Mach numbers ($M \approx 0.8$) and a few data from low-speed tests on configurations with leading-edge slats or trailing-edge flaps deployed.

The wind-tunnel data come from two different types of tests. In one, the model is either rolled at a constant rate or is placed in a steady rolling flow, and in the other it is oscillated in roll. The oscillatory system provides a less direct and therefore less precise measurement of the roll rate derivatives. The flight-test data come from tests in which the dynamic response of an aircraft to various control inputs is measured, and the complete set of aerodynamic stability derivatives consistent with that response is deduced. The accuracy with which the various derivatives can be identified by this process depends on the sensitivity of the aircraft to each derivative. Satisfactory estimates are usually possible for the derivatives N_p and L_p , but the response of the aircraft is so insensitive to Y_p that it is usually omitted from the analysis.

3.1 Accuracy of Prediction at Low α and C_L

Comparisons with experimental data have revealed that at low α and C_L , as shown in Sketch 3.1, Y_p is estimated to within about ± 0.06 , N_p to within about ± 0.016 and L_p to within about ± 10 per cent. These figures are consistent with the accuracies associated with individual Items. As expected the best agreement is achieved with data from rolling model or rolling flow tests. The flight test data show rather more scatter, but the data from tests with oscillatory rigs are subject to the most scatter.

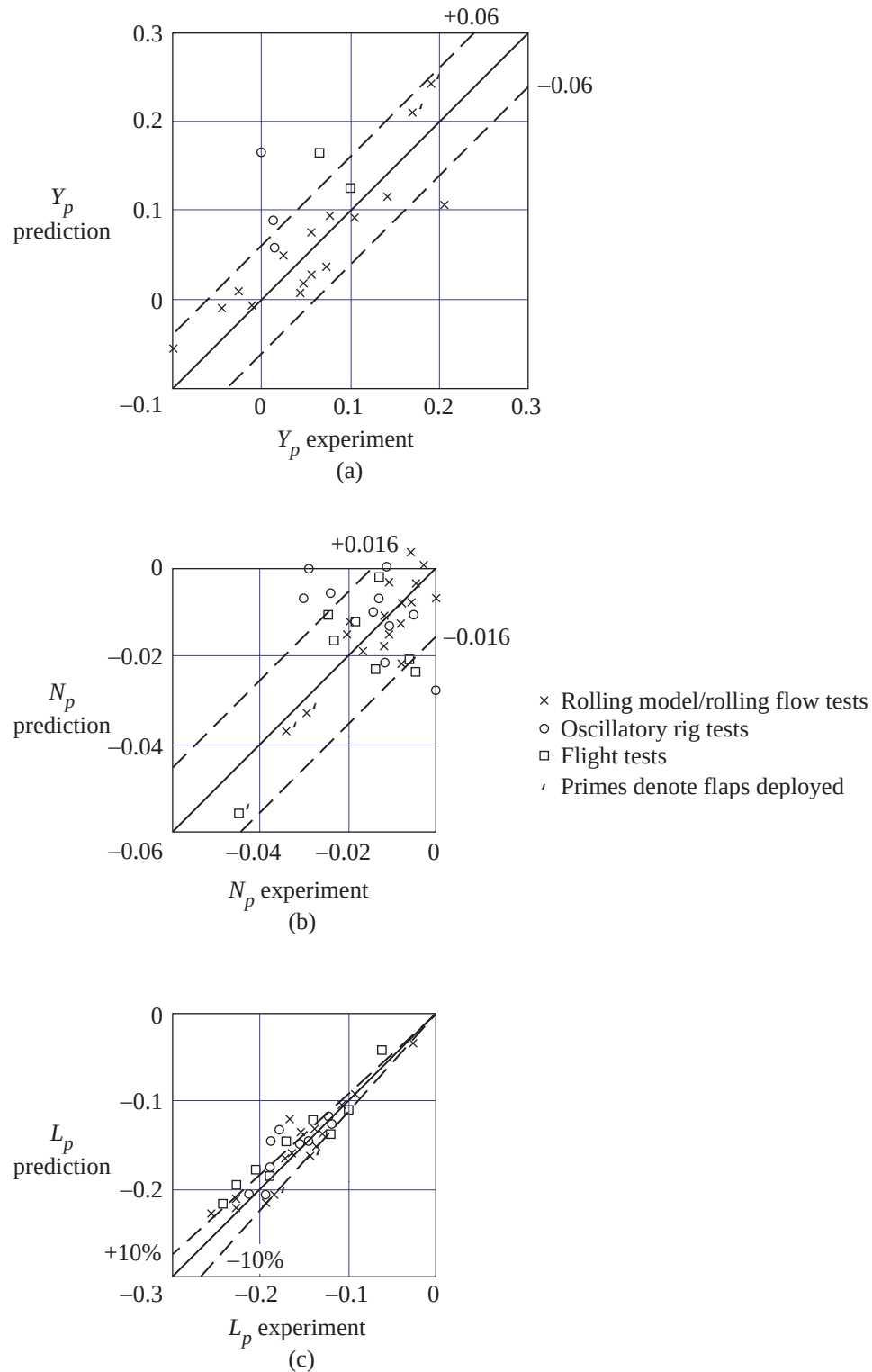
When considering the overall accuracy it is useful to consider Sketches 3.2a to 3.2d that show the predicted component contributions and total values for four different aircraft at cruise conditions.

For the sideforce derivative it can be seen that for the unswept-wing configurations in Sketches 3.2a and 3.2b the dihedral contributions dominate and give negative values of Y_p . The aircraft with swept wings in Sketches 3.2c and 3.2d have more important wing planform contributions that oppose any dihedral contributions and lead to positive values of Y_p . Because the planform and dihedral contributions act in opposite senses any errors in the prediction of either can represent a substantial fraction of the total value.

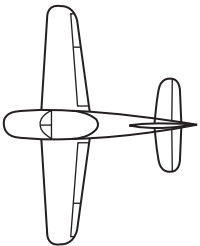
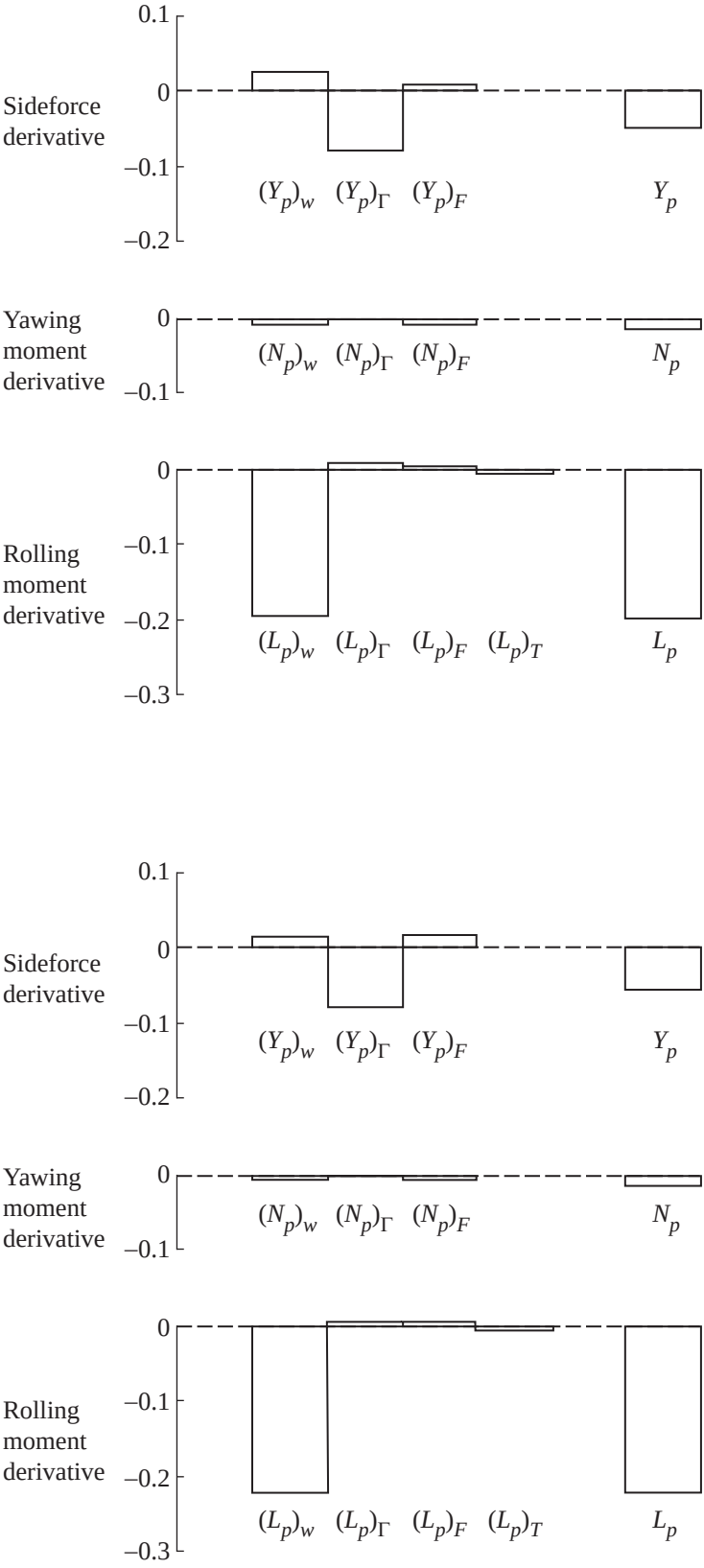
For the yawing moment derivative Sketches 3.2a to 3.2d show that N_p is comprised of small contributions from the wing planform and dihedral and from the fin. The fin contribution $(N_p)_F$ is particularly difficult to estimate reliably because it is strongly influenced by the wing and body sidewash (see ESDU 83006). Thus the uncertainty in N_p can often be of the same order as its overall magnitude (see Sketch 3.1b).

The situation is much simpler for the rolling moment derivative. In this case the wing planform contribution is completely dominant and the small percentage error in the prediction of L_p corresponds to the basic accuracy associated with the component $(L_p)_w$.

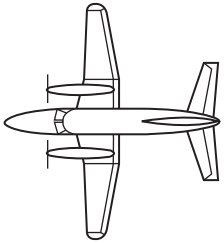
In wholly subsonic flow the effect of Mach number on the rate of roll derivatives is small and predictable, as illustrated in Sketch 3.3.



Sketch 3.1 Comparison of predicted values at low α and C_L

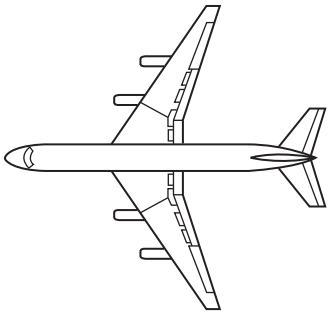
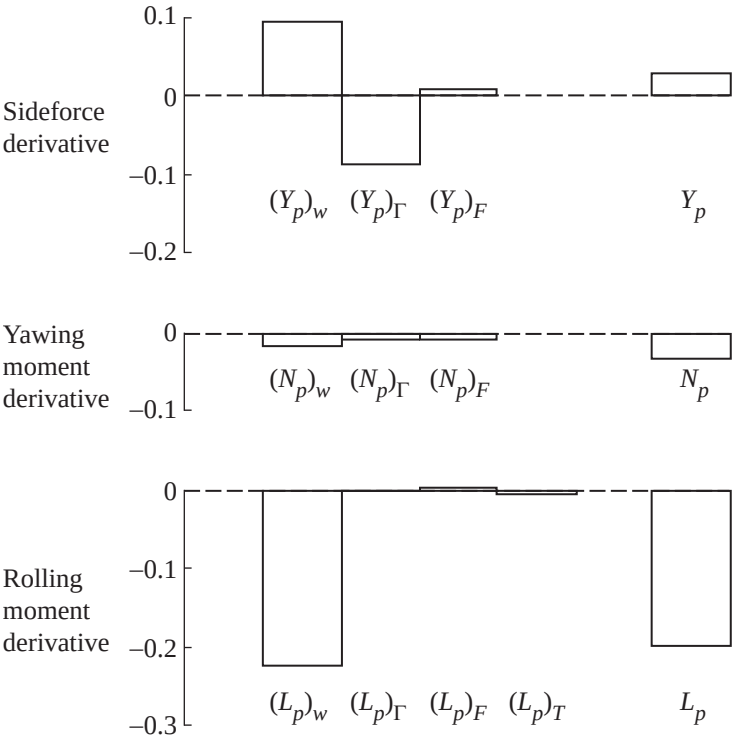


(a) Derivations 15 and 16

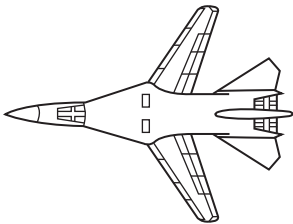
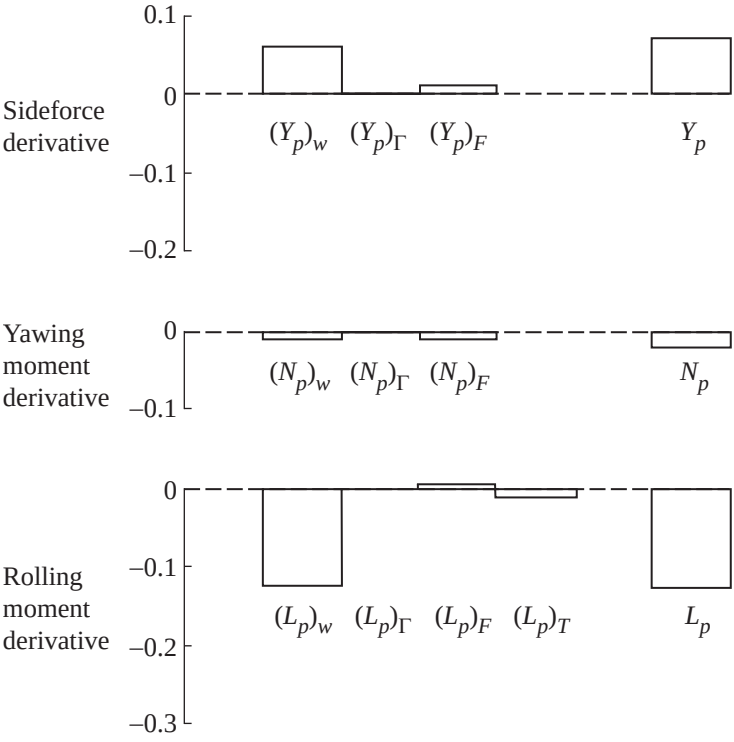


(b) Derivation 24

Sketch 3.2 Illustration of component break-down for cruise conditions

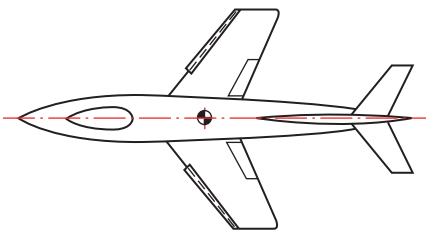
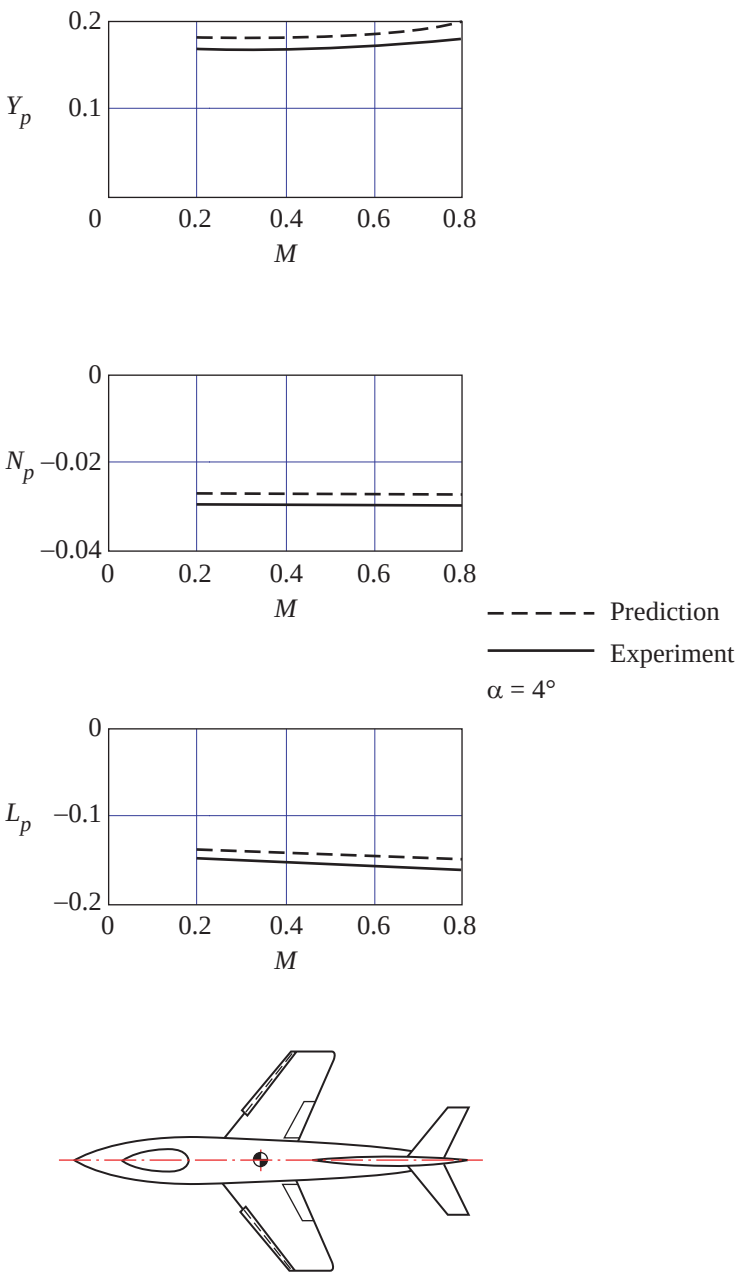


(c) Derivations 14 and 17



(d) Derivation 23

Sketch 3.2 Illustration of component break-down for cruise conditions (concluded)



Derivation 2

Sketch 3.3 Illustration of variation of derivatives with Mach number

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3.2 Variations with α and C_L

3.2.1 General

The accuracy of the prediction of the variation of the rate of roll derivatives with α and C_L cannot be readily quantified and is best illustrated through some typical examples.

It should be remembered that much of the characteristic behaviour of the experimental values of the wing planform contributions to Y_p and N_p is determined by the progressive breakdown of the leading-edge and tip suction forces following the onset of flow separation. Section 3 of ESDU 81014 describes the role of those edge forces with regard to the roll rate derivatives. In brief, the effect is that $(Y_p)_w$ and $(N_p)_w$ increase in magnitude with α until flow separation begins, at which point there is a rapid reduction in the edge forces and hence in the magnitudes of $(Y_p)_w$ and $(N_p)_w$. Eventually $(N_p)_w$, which is initially negative, changes sign and approaches the value $-(L_p)_w \tan \alpha$ that corresponds to a complete absence of edge forces. In ESDU 81014 a method is given for modelling the non-linear behaviour of $(N_p)_w$. This uses the experimental value of the rate of change with angle of attack of the viscous drag coefficient of the wing, i.e. $dC'_D/d\alpha$ where $C'_D = C_D - C_L^2/\pi A$, as a parameter for correlating the flow separation effects. No attempt is made to model the non-linear behaviour of $(Y_p)_w$, which can be expected to fall to zero as the edge forces disappear.

With regard to L_p , the experimental value is almost entirely determined by the wing contribution, and can therefore be expected to remain almost constant with α until flow separation effects influence $(L_p)_w$. Usually there is a large fall in the magnitude of $(L_p)_w$ at high α close to the stall. ESDU Aero A.06.01.01 only provides predictions for attached flow, but in Section 5.3.2 a method for modifying the prediction to allow for flow separation is discussed.

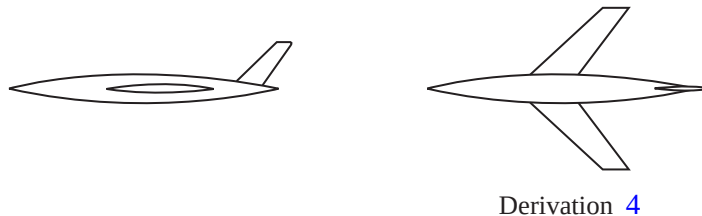
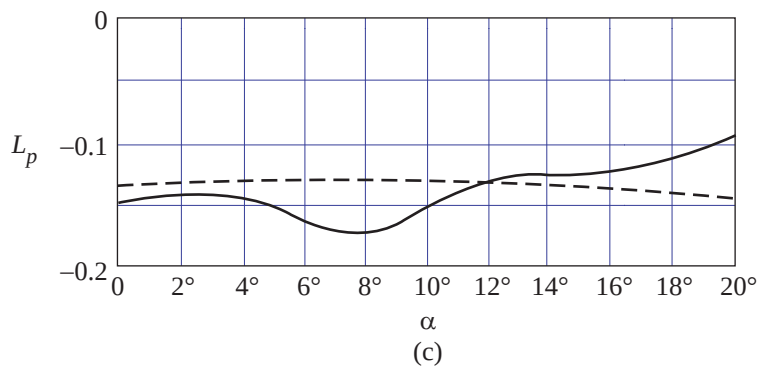
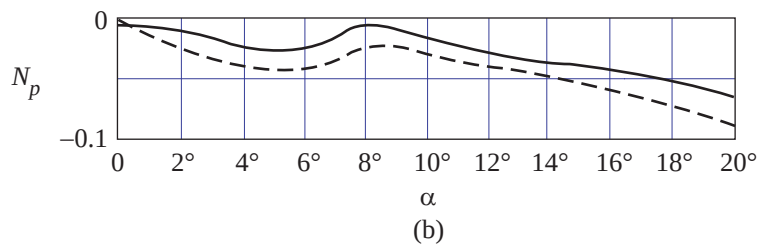
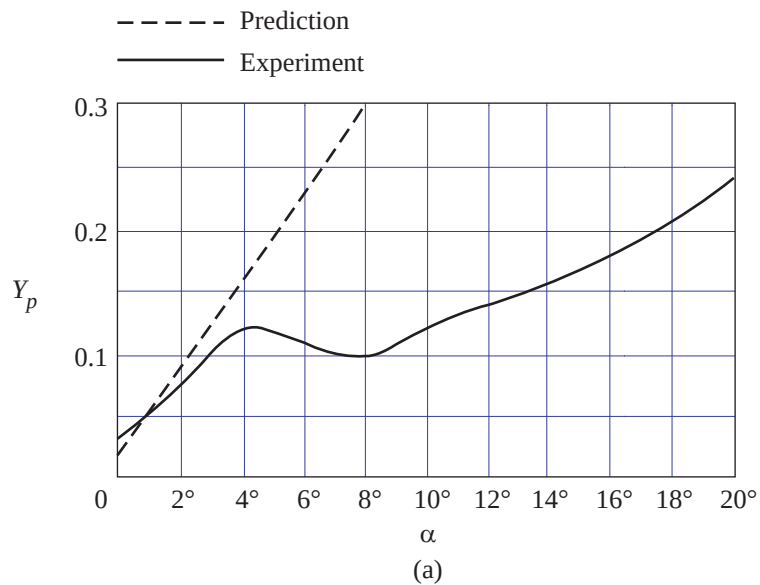
3.2.2 Example 1

Sketch 3.4 compares the total predicted and experimental values of Y_p , N_p and L_p for a model tested in rolling flow, and Sketch 3.5 demonstrates how the total predicted values break down into wing planform and fin contributions.

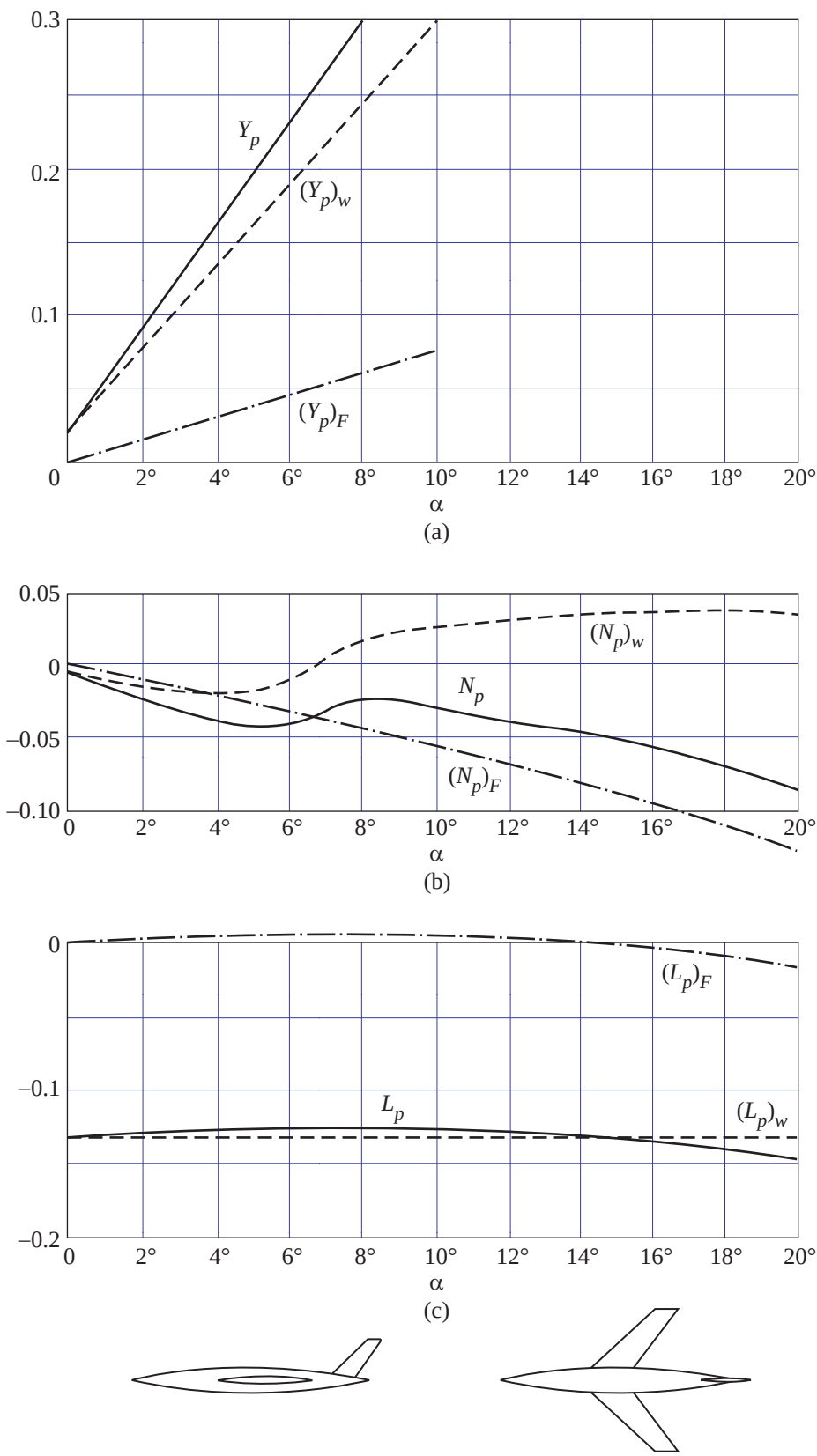
In Sketch 3.4a it can be seen that the experimental variation of Y_p with α is modelled quite well for $\alpha \leq 4^\circ$. Above that there is poor agreement because no attempt is made to predict the non-linear variation of the wing planform contribution. Sketch 3.5a shows that $(Y_p)_w$ provides the major contribution to Y_p with a smaller contribution of the same sign coming from $(Y_p)_F$.

For N_p , Sketch 3.4b shows that a fairly uniform discrepancy is maintained between the experimental and predicted values for $\alpha \geq 2^\circ$. The agreement gets only slightly worse as α increases. The values of N_p become increasingly negative until $\alpha \approx 4^\circ$, when the decay of edge forces begins. This causes a fall in the magnitude of the wing planform contributions which is mirrored by the decrease in the magnitude of N_p until $\alpha \approx 8^\circ$, after which N_p again assumes increasing negative values due to the growing influence of the fin contribution. Sketch 3.5b illustrates this interplay for the predicted values. Although $(N_p)_w$ changes sign and takes positive values from $\alpha \approx 7^\circ$, the steadily increasing negative contribution $(N_p)_F$ is sufficient to overcome this and determine the ultimate variation of N_p .

For L_p , Sketch 3.4c shows that there is an early fluctuation in the experimental value for $5^\circ \leq \alpha \leq 10^\circ$. This is presumably due to local flow separations. Apart from this the general accuracy of prediction is good up to $\alpha \approx 16^\circ$, after which the predicted and experimental values diverge rapidly as the stall is approached. Sketch 3.5c shows that the predicted value of L_p is almost entirely determined by $(L_p)_w$, but the tiny contribution $(L_p)_F$ adds shape to the curve.



Sketch 3.4 Comparison of predicted and experimental variations with α



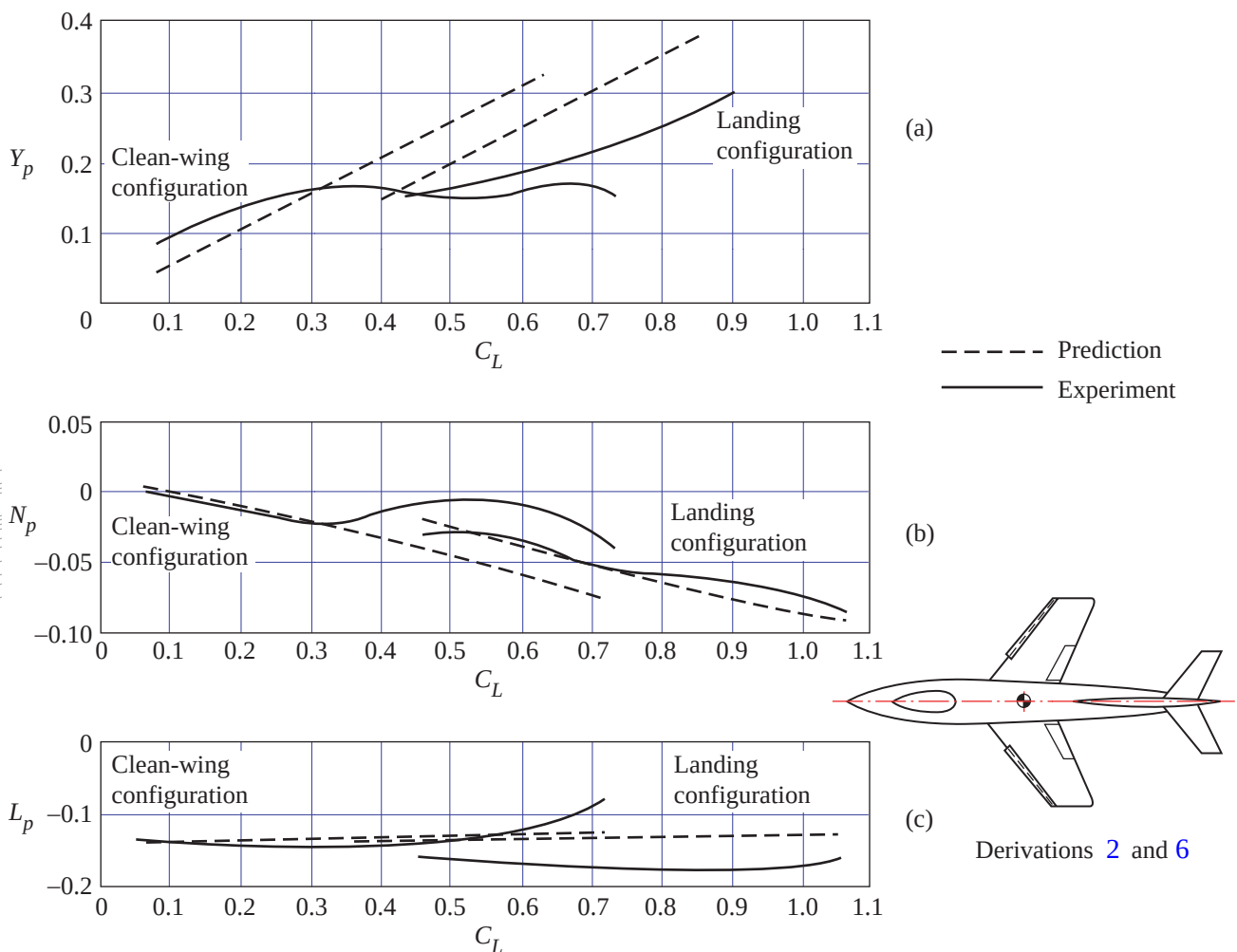
Sketch 3.5 Comparison of wing and fin contributions to total predicted values

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3.2.3 Example 2

In Sketch 3.6 the stability derivatives are plotted against C_L for a configuration tested in rolling flow both with a clean wing and in the landing configuration with wing leading-edge slats, trailing-edge flaps and undercarriage deployed. The clean-wing data show many of the same trends as the data in Sketch 3.4. Both the Y_p and N_p experimental curves increase fairly linearly at low C_L and lie close to the predicted values. However, at $C_L \approx 0.3$ both curves break and fall in magnitude. After that no serious attempt is made to predict Y_p . A prediction of N_p has been attempted but, because there are no experimental data on the drag of the wing, only a rough estimate of $dC_D'/d\alpha$ could be made. Too low an estimate of $dC_D'/d\alpha$ has resulted in an insufficient reduction in the predicted value of N_p for $0.35 \leq C_L \leq 0.7$. This highlights the difficulty of predicting the non-linear part of $(N_p)_w$ in the absence of experimental data on lift and drag. For L_p there is again good agreement until the experimental value declines rapidly at high C_L .

The experimental effect of deploying high-lift devices is to delay to a higher C_L the onset of flow separation and therefore to extend the range over which the stability derivatives show nearly linear variations. In Sketch 3.6 it can be seen that predicted values of Y_p are of the correct trend, those for N_p are good and those for L_p remain reasonable.

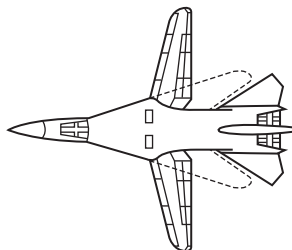
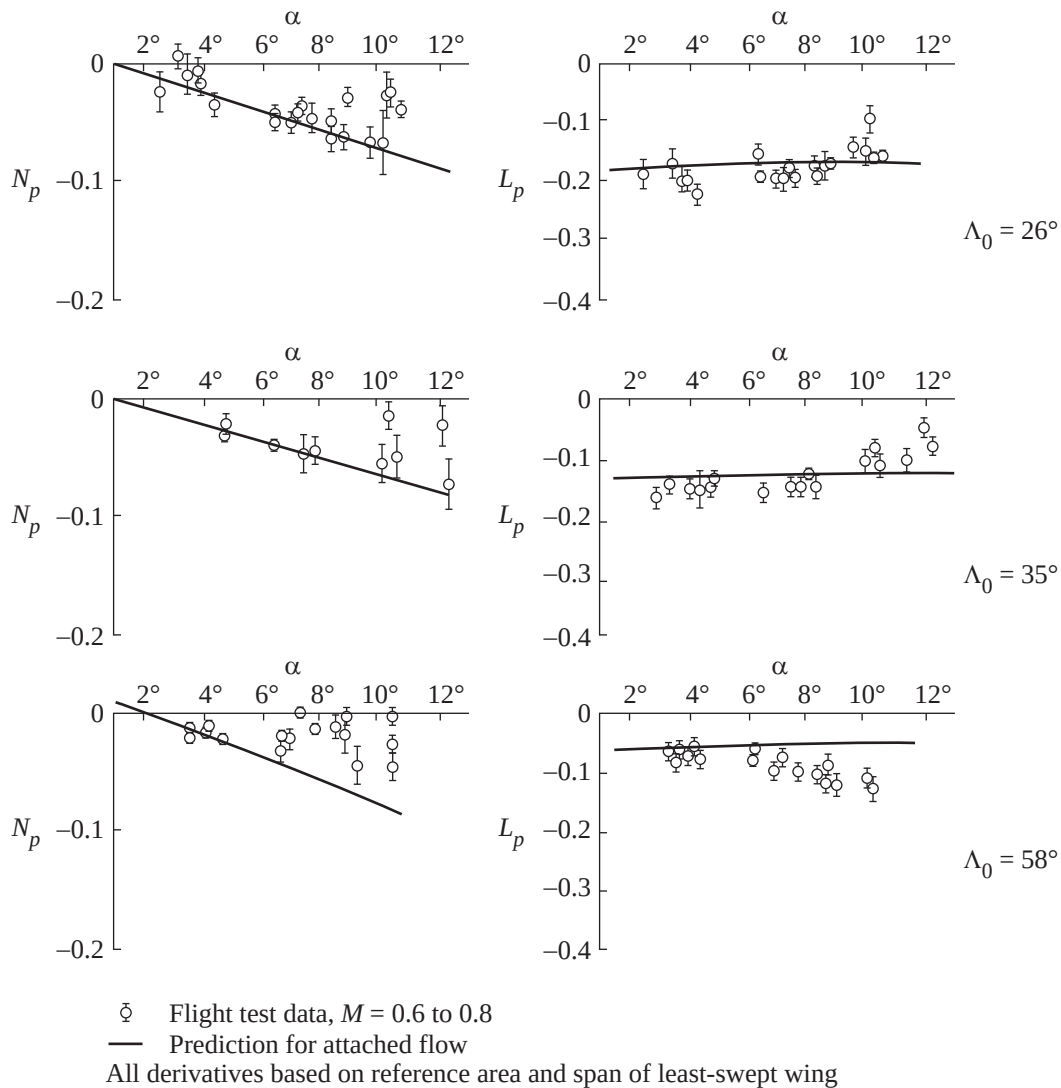


**Sketch 3.6 Comparison of predicted and experimental variations with C_L
(effect of high-lift devices)**

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3.2.4 Example 3

In Sketch 3.7 predicted values are compared with flight-test data. The results suggest that N_p and L_p are predicted quite well for wings with low and moderate sweep, but less well for highly swept wings. In general the agreement is very good at low α . The estimate of N_p at higher α would be improved if allowance were to be made for the decay of wing edge forces. An increase in wing sweep reduces the angle of attack at which that decay starts and N_p becomes non-linear. The increase in the magnitude of L_p apparent in the flight-test data for $\Lambda_0 = 58^\circ$ and $\alpha \geq 6^\circ$ is associated with the incremental non-linear lift due to the formation of leading-edge vortices.



Derivation 23

Sketch 3.7 Comparison of predicted and experimental variations with α

4. DERIVATION

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

Experimental Data

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LICHTENSTEIN, J.H.
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5. EXAMPLE

This section provides a worked example to show how Y_p , N_p , and L_p are calculated for the aircraft dimensioned as shown in Sketch 5.1 together with the additional geometric information in Table 5.1. Both inner and outer flaps are of the single-slotted type. The longitudinal body axis is taken parallel to the mid-body centre-line and passes through the aircraft moment reference centre. Angles of attack, α , are expressed in terms of this axis.

Calculations are performed for two flight conditions:

- (i) a cruise condition with $\alpha = 0$, $M = 0.78$ and Reynolds number/metre = 7.5×10^6 ,
- and (ii) a landing condition with $\alpha = 6^\circ$, $M = 0.20$ and Reynolds number/metre = 4.5×10^6 .

Where appropriate the components of Y_p , N_p , and L_p are also expressed as functions of α or C_L . The variations of the total values are illustrated by sketches in Section 5.7.

Note that for the aircraft used in this example ESDU 81032 and 82011 describe the estimation of lateral stability derivatives due to sideslip and ESDU 84002 those due to rate of yaw.

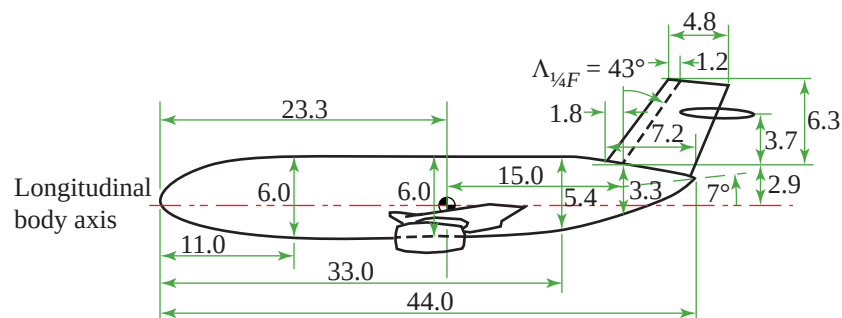
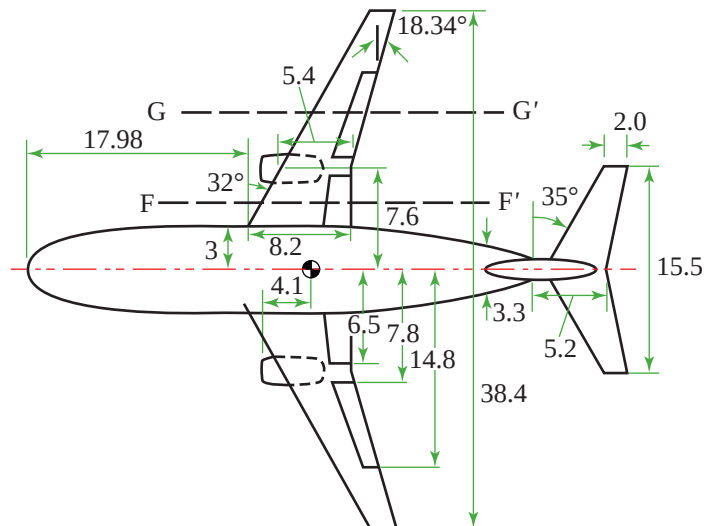
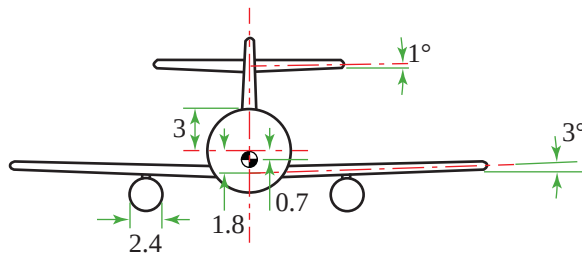
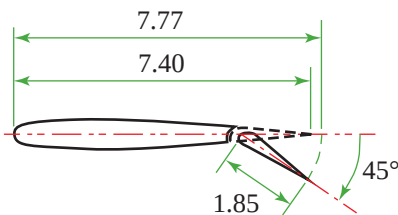
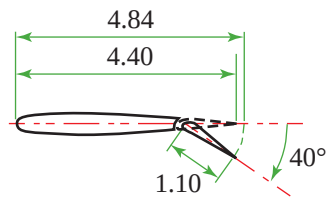
TABLE 5.1 Additional Geometric Parameters for Aircraft in Sketch 5.1

WING		
Angle between wing zero-lift line and longitudinal body axis	3°	
Average section trailing-edge angle	10°	
Average section thickness-to-chord ratio	0.10	
Trailing-edge inboard sweep angle	0.965°	
Trailing-edge outboard sweep angle	18.34°	
FLAPS (single slotted)	At section FF'	At section GG'
Flap-chord to wing-chord ratio	0.250	0.250
Flap-chord to extended-wing-chord ratio	0.238	0.227
Extended-wing-chord to wing-chord ratio	1.05	1.10
Flap deflection angle	45°	40°
BODY		
Maximum cross-sectional area	28.3 m ²	
Area of side elevation	224 m ²	
FIN		
Side area between tip and root chords	37.8 m ²	
Note (i) The wing and flap section parameters are taken in planes parallel to the aircraft plane of symmetry. (ii) Boundary-layer transition is assumed to occur at the leading edge of the wing.		

Wing and tailplane mid-semispan section properties

$t/c = 0.1$, $(t/c)_{90} = 0.0202$, $(t/c)_{99} = 0.0018$

$\tau = 10^\circ$, $x_{tr}/c = 0$



● Moment reference point
Dimensions in metres

$\alpha_0 = -3.0^\circ$

Sketch 5.1 Aircraft geometry

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5.1 Calculation of Wing Planform Parameters

See ESDU 76003 for Notation

Before commencing the estimation of Y_p , N_p and L_p it is usually necessary to calculate a number of geometric parameters for the wing planform that are not immediately available from Sketch 5.1 or Table 5.1. This is because the Items dealing with the wing are only directly applicable to straight-tapered wings. Therefore, unless the aircraft has this type of wing, for which the planform parameters can be readily obtained from a scale diagram, a straight-tapered wing equivalent to the true wing has to be constructed by the method in the Addendum to ESDU 76003 (Derivation 35). That Item represents a cranked wing by a straight-tapered wing that has the same span, the same tip chord, and the same exposed wing area outside the intersection of the wing and body planforms, as the true wing. The equivalent-wing planform parameters that result from applying the method in ESDU 76003 to the aircraft in Sketch 5.1 are summarised in Table 5.2.

The values and notation in Table 5.2 are used throughout the example for the wing geometry.

TABLE 5.2 Properties of Equivalent Straight-tapered Wing Planform

Parameter	Value	Parameter	Value
Wing planform area, S	194.3 m ²	Leading-edge sweep, Λ_0	32.0°
Aspect ratio, A	7.59	Quarter-chord sweep, $\Lambda_{1/4}$	28.6°
Aerodynamic mean chord, $\bar{c}$	5.68 m	Half-chord sweep, $\Lambda_{1/2}$	25.0°
Ratio of tip chord to root chord, λ	0.246		