PROGRAM FOR CALCULATION OF MAXIMUM LIFT COEFFICIENT OF PLAIN AEROFOILS AND WINGS AT SUBSONIC SPEEDS

The program has been written in “STRICT” Microsoft FORTRAN 77 for use on machines using PC/MS DOS. A diskette containing files for the source code, worked examples and an information file (README) is provided in the Aerodynamics Software Volume. Guidance on copying, compilation and running the program is given in the “Introduction to ESDUpacs” in that volume. Additional guidance for this particular program is given in the README file on the diskette. However, if any difficulty is experienced in using the program please contact ESDU International and we will do all we can to assist in overcoming the problem. The software is also available on CD Rom.

1. INTRODUCTION

This Item provides details of a program, called ESDUpac A9315, for the calculation of the maximum lift coefficient of plain aerofoils and wings at subsonic speeds. The methods employed are those given in Item No. 84026 (Derivation 4) for aerofoils and Item No. 89034 (Derivation 5) for wings. Item No. 84026 requires aerofoil lift-curve slope from Item No. Aero W.01.01.05 (Derivation 1) and Item No. 89034 requires data for the spanwise location of wing centre of pressure from Item No. 83040 (Derivation 3).

Section 2 summarises the program and Section 3 describes the input and output data. The input files appropriate to an aerofoil or wing calculation are described in tabular form. Section 4 list the Derivations. Section 5 gives some examples of input and output using the program, including those given in Item Nos 84026 and 89034. The original Items should be referred to for notation and applicability and limitations of the methods.

2. SUMMARY OF THE PROGRAM

The program combines the methods of Item Nos 84026 and 89034 to allow the user to carry out calculations for either aerofoils (2D) or wings (3D) determined by the input of the characters “2D” or “3D” in the input file.

If the calculations are required for an aerofoil the program reads in all the required geometric and flow data from a single input file (Section 3.1.1). For a given aerofoil, which can have either a “smooth” or “rough” leading-edge region as defined in Item No. 84026, calculations can be carried out for a range of Mach numbers and Reynolds numbers within the limits specified in the Item.

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The method of Item No. 84026 requires a knowledge of whether the aerofoil is a “conventional” or a “modern” one. With the program the user is given the complete flexibility of a choice between deciding for himself into which category his aerofoil falls or allowing the computer to decide. The computer decision involves a test based on the geometry over the rear 10% of the aerofoil chord. If $[z_u(0.9) - z_l(0.9)]/z_u(0.9) > 0.64$ and $t_b/c > 0.004$ then the assumption is made that the aerofoil is a modern one; it is otherwise assumed to be conventional. If the user elects to decide the category, a warning message is output should the choice be contrary to that which the computer would have made. If the computer is allowed to decide, a warning message is output concerning the choice.

For the case of a wing, which must have a smooth leading edge, it is necessary to divide the run into two parts. The first part, in which the planform geometry and required Mach numbers are input (Section 3.1.2(a)), is necessary to determine η_p, the spanwise location of the peak loading at each Mach number. This information enables the user to establish the section geometry at each value of η_p, if not already known, and is required by the program to carry out the calculation of the maximum lift coefficient for the section at η_p. The input file for the second part, concerning the calculation of C_{Lmax} (Section 3.1.2 (b)), consists of that for the first part plus the now known section geometry at each η_p, together with the geometry required to define the wing camber and twist variation across the span. Non-linear spanwise variations of geometric twist and camber are allowable provided that the twist variation and the resulting spanwise variation of section zero-lift angle due to camber are monotonic and give a modest effective tip twist angle $(|\delta_{et}| \leq 10°)$. For such cases geometrical data are required only for the root and 2/3 span sections. Finally, the Reynolds number values are input.

The Figures from the Items are, with one exception, stored within the program in digitised form and the data are extracted using interpolations via low order polynomial fits. The one exception concerns the Figures of Item No. Aero W.01.01.05; in that case the original empirical equations from Derivation 2 were used.

The output is written to a user-specified output file, see Section 3.2. The data given in the input files are checked for validity and appropriate error or warning messages are output if any invalidity is detected. Similarly, allowance is made for error or warning messages to be triggered during the calculation should any parameter be either within an area of uncertainty on the figures or outside specified ranges.

3. INPUT AND OUTPUT

3.1 Input

Tables 3.1 and 3.2 respectively list, in order, the parameters required in the input files for an aerofoil and wing calculation and, where appropriate, their notation in the relevant Item. Examples of typical input files which were constructed by reference to the tables are given in Section 5.

3.1.1 Aerofoils (2D)

TABLE 3.1 Input file for aerofoils

Variable name in program	*Notation in Item*	*Comments*
CHARR(1) CHARR(2) CHARR(3)	– – –	The first three lines of the input data file are read as text and are written to the output file to enable the user to describe the run. Each line may contain 72 characters and must end with a carriage return. In each case a blank line must be entered if no text is available.
FYLNAM	–	Name of file (up to 12 characters) to which output file is to be directed.
ID	–	Indicator for aerofoil or wing. For aerofoil ID = 2D or 2d.
MODERN	–	Variable used to describe aerofoil category. MODERN = Y or y; user has decided aerofoil is “modern”, see Section 6 of Item No. 84026. MODERN = N or n; user has decided aerofoil is “conventional”, see Section 6 of Item No. 84026. MODERN = ?; computer to decide aerofoil category, see Section 2 of this Item.
RORS	–	Indicator for rough (RORS = R or r) or smooth (RORS = S or s) leading-edge region, see Section 2.2 of Item No. 84026.
TBYC	t/c	Aerofoil thickness to chord ratio $(0.06 \le \mathrm{TBYC} \le 0.24)$.
XTRBYC	x_{tr}/c	Dimensionless location of boundary layer transition $(0 \le \mathrm{XTRBYC} \le 1)$.
XUMBYC, ZUMBYC	x_{um}/c, z_{um}/c	Dimensionless location and value of maximum upper-surface ordinate.
TAUDEG	τ	Trailing-edge angle, degrees.
XBYC(1), ZUBYC(1), ZLBYC(1) : : : XBYC(J), ZUBYC(J), ZLBYC(J) : : : XBYC(17), ZUBYC(17), ZLBYC(17)	x/c, z_u/c, z_l/c	Aerofoil dimensionless upper-surface and lower-surface coordinates at each of J = 1 to 17 standard values of XBYC(J) = 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, 1.0.
NMG	–	Number of Mach number values to be given $(1 \le \mathrm{NMG} \le 10)$.
EMS(1),.., EMS(J),.., EMS(NEMS)	M	Values of Mach number; on one line $(0 < \mathrm{EMS(J)} \le 0.4$, $1 \le \mathrm{NEMS} \equiv \mathrm{NMG} \le 10)$.
NRG	–	Number of Reynolds number values to be given $(1 \le \mathrm{NRG} \le 10)$.

TABLE 3.1 Input file for aerofoils

Variable name in program	*Notation in Item*	*Comments*
RCS(1),.., RCS(J),.., RCS(NRCS)	R_c	Values of Reynolds number; on one line $(0.7 \times 10^6 \leq \mathrm{RCS(J)} \leq 9 \times 10^6$ for RORS = S or s, $0.7 \times 10^6 \leq \mathrm{RCS(J)} \leq 6 \times 10^6$ for RORS = R or r, and $\mathrm{ZUBYC(3)} \geq 0.017$, RCS(J) = 6×10^6 for RORS = R or r, and $\mathrm{ZUBYC(3)} < 0.017$. $1 \leq \mathrm{NRCS} \equiv \mathrm{NRG} \leq 10)$.

3.1.2 Wings (3D)

For a given Mach number appropriate section geometry (at η_p) for a wing calculation can in general only be established when the value of η_p has been determined and so for a wing the run is split into two parts. Part I determines η_p and Part II is the full calculation for maximum lift coefficient. Obviously, if η_p, or the section geometry at η_p, is already known then only Part II is required.

(a) **Part I: Determination of η_p**

TABLE 3.2 (a) Input file for wings – Part I

Variable name in program	*Notation in Item*	*Comments*
CHARR(1) CHARR(2) CHARR(3)	– – –	The first three lines of the input data file are read as text and are written to the output file to enable the user to describe the run. Each line may contain 72 characters and must end with a carriage return. In each case a blank line must be entered if no text is available.
FYLNAM	–	Name of file (up to 12 characters) to which output file is to be directed.
ID	–	Indicator for aerofoil or wing. For aerofoil ID = 3D or 3d.
MODERN	–	Variable used to describe category of wing section at η_p. MODERN = Y or y; user has decided aerofoil is “modern”, see Section 6 of Item No. 84026. MODERN = N or n; user has decided aerofoil is “conventional”, see Section 6 of Item No. 84026. MODERN = ?; computer to decide aerofoil category, see Section 2 of this Item. NOTE: although this information is not actually required for a Part I run it is included to simplify the editing required to construct the input file for Part II, see Table 3.2(b).
ASP	A	Aspect ratio $(2 \leq \mathrm{ASP} \leq 12)$.
ELMDA	λ	Taper ratio $(0 \leq \mathrm{ELMDA} \leq 1)$.

TABLE 3.2 (a) Input file for wings – Part I

Variable name in program	*Notation in Item*	*Comments*
EN	n	Chord line parameter $(0 \le \mathrm{EN} \le 1)$.
ELMDAN	Λ_n	Sweepback of n'th chord line $(0 \le \mathrm{ELMDAN} \le 50°)$.
NMG	–	Number of Mach number values to be given $(1 \le \mathrm{NMG} \le 10)$.
EMS(1),.., EMS(J),.., EMS(NEMS)	M	Values of Mach number; on one line $(0 < \mathrm{EMS(J)} \le 0.8$, $1 \le \mathrm{NEMS} \equiv \mathrm{NMG} \le 10)$.

(b) **Part II: Determination of C_{Lmax}**

TABLE 3.2 (b) Input file for wings – Part II

Variable name in program	*Notation in Item*	*Comments*
CHARR(1) CHARR(2) CHARR(3)	– – –	The first three lines of the input data file are read as text and are written to the output file to enable the user to describe the run. Each line may contain 72 characters and must end with a carriage return. In each case a blank line must be entered if no text is available.
FYLNAM	–	Name of file (up to 12 characters) to which output file is to be directed.
ID	–	Indicator for aerofoil or wing. For aerofoil ID = 3D or 3d.
MODERN	–	Variable used to describe aerofoil category of wing section at η_p. MODERN = Y or y; user has decided aerofoil is “modern”, see Section 6 of Item No. 84026. MODERN = N or n; user has decided aerofoil is “conventional”, see Section 6 of Item No. 84026. MODERN = ?; computer to decide aerofoil category, see Section 2 of this Item.
ASP	A	Aspect ratio $(2 \le \mathrm{ASP} \le 12)$.
ELMDA	λ	Taper ratio $(0 \le \mathrm{ELMDA} \le 1)$.
EN	n	Chord line parameter $(0 \le \mathrm{EN} \le 1)$.
ELMDAN	Λ_n	Sweepback of n'th chord line $(0 \le \mathrm{ELMDAN} \le 50°)$.
NMG	–	Number of Mach number values to be given $(1 \le \mathrm{NMG} \le 10)$.

TABLE 3.2 (b) Input file for wings – Part II

Variable name in program	*Notation in Item*	*Comments*
EMS(1),.., EMS(J),.., EMS(NEMS)	M	Values of Mach number; on one line $(0 < \text{EMS(J)} \le 0.8$, $1 \le \text{NEMS} \equiv \text{NMG} \le 10)$.
For each value of EMS(J), *i.e*. for M = 1 to NEMS, the following data concerning the section geometry at η_p are required to determine $(C_{Lm})_p$ at low speeds via Item No. 84026.		
TBYC(M)	$(t/c)_p$	Section thickness to chord ratio $(0.06 \le \text{TBYC(M)} \le 0.24)$.
XTRBYC(M)	$(x_{tr}/c)_p$	Dimensionless location of boundary layer transition $(0 \le \text{XTRBYC(M)} \le 1)$.
XUMBYC(M), ZUMBYC(M)	$(x_{um}/c)_p$, $(z_{um}/c)_p$	Dimensionless location and value of maximum upper-surface ordinate.
TAUDEG(M)	$(\tau)_p$	Trailing-edge angle, degrees.
XBYC(1), ZUBYC(1,M), ZLBYC(1,M) : : : XBYC(J), ZUBYC(J,M), ZLBYC(J,M) : : : XBYC(17), ZUBYC(17,M), ZLBYC(17,M)	$(x/c)_p$, $(z_u/c)_p$, $(z_l/c)_p$	Section dimensionless upper-surface and lower-surface coordinates at each of J = 1 to 17 standard values of XBYC(J) = 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, 1.0.
Wing twist and camber data:		
TI, CI	–	Twist and camber indicators: TI = 0 for no geometrical twist TI = T for geometrical twist, linear or non-linear (but monotonic) CI = U for uniform (constant) camber across span CI = V for camber varying across span giving either linear or non-linear (but monotonic) variation of section zero-lift angle.
If CI = V:		
Two sets of aerofoil coordinates are required, for the root (K = 1) and 2/3 span (K = 2) sections to calculate α_{0r} and $\alpha_{02/3}$ from Equation (5.1) of Item No. 84026.		
XBYCT(1), ZUBYCT(1, K), ZLBYCT(1, K) . . . . XBYCT(J), ZUBYCT(J, K), ZLBYCT(J, K) . . . . XBYCT(14), ZUBYCT(14, K), ZLBYCT(14, K)	$(x/c)_r$, $(z_u/c)_r$, $(z_l/c)_r$, $(x/c)_{2/3}$, $(z_u/c)_{2/3}$, $(z_l/c)_{2/3}$	Section dimensionless upper-surface and lower-surface coordinates at each of J = 1 to 14 standard values of XBYCT(J) = 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 and 1.0.

TABLE 3.2 (b) Input file for wings – Part II

Variable name in program	*Notation in Item*	*Comments*
If TI = T:		
DELT23	$\delta_{2/3}$	Geometrical twist at 2/3 span (degrees).
NRG	–	Number of Reynolds number values to be given $(1 \le \mathrm{NRG} \le 10)$.
RCS(1),.., RCS(J),.., RCS(NRCS)	$R_{\bar{\bar{c}}}$	Values of Reynolds number; on one line $(0.7 \times 10^6 \le \mathrm{RCS(J)} \le 12 \times 10^6$ $1 \le \mathrm{NRCS} \equiv \mathrm{NRG} \le 10)$.

3.2 Output

The output begins with a standard header. Three lines of user-defined text, describing the run for example, then follow together with the chosen output file name. Should any errors be detected while reading in the input data, appropriate messages are output at this stage. Details of the input data then follow to act as a record and a check, preceded by a general heading to indicate whether the calculations are for an aerofoil or a wing.

The results of the calculations then follow, with general section or planform data preceding tabulated calculations for each requested Mach number.

Table 3.3 translates the output parameters into the notation of the relevant Items.

TABLE 3.3 Program output parameters

Output parameter	*Notation in Item*
A	A
(a1)0	$(a_1)_0$
Alpha0	α_0
Alpha0r	α_{0r}
Alpha02/3	$\alpha_{02/3}$
Beta	β
CLm	C_{Lm}
CLmax	C_{Lmax}
CL0	C_{L0}
c	c
cbb	$\bar{\bar{c}}$
cp	c_p
DeltaCL	ΔC_L
DeltaCLLambda	$\Delta C_{L\Lambda}$
DeltaCLM	ΔC_{LM}
DeltaCLR	ΔC_{LR}
DeltaCLT	ΔC_{LT}
Deltaet	δ_{et}
Delta2/3	$\delta_{2/3}$
Etab	η
Etap	η_p
FM	F_M
FS	F_S
Kappa	κ

Output parameter	*Notation in Item*
Lambda	λ
Lamdan	Λ_n
Lambda0	Λ_0
Lambda1/4	$\Lambda_{1/4}$
Lambda1/2	$\Lambda_{1/2}$
Lambda1	Λ_1
M	M
Mup	μ_p
n	n
Rc	R_c
Rcbb	$R_{\bar{\bar{c}}}$
Rcp	R_{cp}
Tau	τ
Taua	τ_a
Tauu	τ_u
t	t
tb	t_b
xtr	x_{tr}
xum	x_{um}
Zetap	ζ_p
zl	z_l
zu	z_u
zum	z_{um}

WARNING AND ERROR MESSAGES

Provision is made within the program for a number of warning and error messages to be output when triggered. Warnings are associated with successful runs and are cautionary. Error messages are associated with a failed or terminated run and are intended to help in pin-pointing any error or inconsistency in the input data.

4. DERIVATION

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

1.	ESDU	Slope of lift-curve for two-dimensional flow. ESDU International, Item No. Aero W.01.01.05, 1955.
2.	GARNER, H.C.	Charts for low speed characteristics of two-dimensional trailing-edge flaps. ARC R&M 3174, 1957.
3.	ESDU	Method for the rapid estimation of spanwise loading of wings with camber and twist in subsonic attached flow. ESDU International, Item No. 83040, 1983.
4.	ESDU	Aerofoil maximum lift coefficient for Mach numbers up to 0.4. ESDU International, Item No. 84026, 1984.
5.	ESDU	The maximum lift coefficient of plain wings at subsonic speeds. ESDU International, Item No. 89034, 1989.

5. EXAMPLES OF INPUT AND OUTPUT

Examples are given for both aerofoils (2D) and wings (3D). The input files were constructed according to Tables 3.1 or 3.2 (a) and 3.2 (b) as appropriate.

5.1 Aerofoils

5.1.1 Example 1

This is Example 1 of Item No. 84026.

Input

```
EXAMPLE 1 of Item No.84026.
Input file I01A9315 Output file R01A9315

R01A9315
2D
N
S
0.1
0.3
0.430  0.0610
0
0.0     0.0      0.0
0.01    0.012   -0.010
0.0125  0.013   -0.0105
0.025   0.018   -0.014
0.05    0.025   -0.019
0.10    0.036   -0.025
0.20    0.0495  -0.0334
0.30    0.0570  -0.0379
0.40    0.0607  -0.0392
0.50    0.0592  -0.0371
0.60    0.0522  -0.0308
0.70    0.0413  -0.0218
```

```
0.80    0.0278  -0.0119
0.90    0.0133  -0.0029
0.95    0.0062   0.0001
0.99    0.00124  0.00002
1.0     0.0      0.0
1
0.10
3
3.E6 6.E6 9.E6
```

Output

```
*****************************************************************************

 ESDU International plc

 Program               A9315

 ESDUpac Number:       A9315
 ESDUpac Title:        Maximum lift coefficient of plain aerofoils and wings
                       at subsonic speeds.
 Data Item Number:     93015
 Data Item Title:      Program for calculation of maximum lift coefficient
                       of plain aerofoils and wings at subsonic speeds.
 ESDUpac Version:      1.1    Issued January 1994

 See Data Item for full input/output specification and interpretation.

 *****************************************************************************

 INPUT DATA
 ==========

 EXAMPLE 1 of Item No.84026.
 Input file I01A9315 Output file R01A9315


 Output file name: R01A9315

 INPUT DATA ERRORS
 -----------------

       No error is detected


                   A E R O F O I L   C A L C U L A T I O N:  2D
                   --------------------------------------------

 AEROFOIL DATA
 -------------

 Aerofoil category is "conventional"

 Aerofoil leading-edge is: smooth

 Thickness-to-chord ratio,                                t/c =  .1000
 Location of boundary layer transition,                 xtr/c =  .3000
 Maximum dimensionless upper-surface ordinate,          zum/c =  .6100E-01
                        at chordwise location,          xum/c =  .4300
 Aerofoil trailing-edge angle,                  Tau (degrees) =  .0000
```

```
Aerofoil coordinates:

                j       xj/c       zuj/c       zlj/c

                1      .00000      .00000      .00000
                2      .01000      .01200     -.01000
                3      .01250      .01300     -.01050
                4      .02500      .01800     -.01400
                5      .05000      .02500     -.01900
                6      .10000      .03600     -.02500
                7      .20000      .04950     -.03340
                8      .30000      .05700     -.03790
                9      .40000      .06070     -.03920
               10      .50000      .05920     -.03710
               11      .60000      .05220     -.03080
               12      .70000      .04130     -.02180
               13      .80000      .02780     -.01190
               14      .90000      .01330     -.00290
               15      .95000      .00620      .00010
               16      .99000      .00124      .00002
               17     1.00000      .00000      .00000


FLOW CONDITIONS
---------------

Number of Mach numbers         =  1
Mach number values, M          =  .1000

Number of Reynolds numbers     =  3
Reynolds number values, Rc     =  .3000E+07  .6000E+07  .9000E+07


OUTPUT DATA
===========

Value of                                  zu1.25/c =   .1300E-01
Trailing-edge angle,                Taua (degrees) =   9.515

Aerofoil zero-lift angle,       Alpha0 (radians) =  -.2737E-01
                                       (degrees) =  -1.568


Mach number, M  =  .1000
------------------------

----------------------------------------------------------------------------
Rc(x 1.E-6)  (a1)0 (per radian)    CL0   DeltaCL     FS        FM        CLm
----------------------------------------------------------------------------
   3.000           5.889          .161    1.122     1.000     1.000     1.283
   6.000           6.007          .164    1.194     1.000     1.000     1.358
   9.000           6.061          .166    1.234     1.000     1.000     1.400
----------------------------------------------------------------------------

END OF OUTPUT  -------------------------------------------------------------
```

5.1.2 Example 2

Input

This is Example 2 of Item No. 84026.

```
EXAMPLE 2 of Item No.84026.
Input file I02A9315 Output file R02A9315

R02A9315
2D
N
R
0.1
0.0
0.430  0.0610
0
0.0     0.0      0.0
0.01    0.012   -0.010
0.0125  0.013   -0.0105
0.025   0.018   -0.014
0.05    0.025   -0.019
0.10    0.036   -0.025
0.20    0.0495  -0.0334
0.30    0.0570  -0.0379
0.40    0.0607  -0.0392
0.50    0.0592  -0.0371
0.60    0.0522  -0.0308
0.70    0.0413  -0.0218
0.80    0.0278  -0.0119
0.90    0.0133  -0.0029
0.95    0.0062   0.0001
0.99    0.00124  0.00002
1.0     0.0      0.0
1
0.10
1
6.E6
```

Output

```
******************************************************************************

 ESDU International plc

 Program              A9315

 ESDUpac Number:      A9315
 ESDUpac Title:       Maximum lift coefficient of plain aerofoils and wings
                      at subsonic speeds.
 Data Item Number:    93015
 Data Item Title:     Program for calculation of maximum lift coefficient
                      of plain aerofoils and wings at subsonic speeds.
 ESDUpac Version:     1.1    Issued January 1994

 See Data Item for full input/output specification and interpretation.

 ******************************************************************************
```

```
INPUT DATA
==========

EXAMPLE 2 of Item No.84026.
Input file I02A9315 Output file R02A9315


Output file name: R02A9315

INPUT DATA ERRORS
-----------------

      No error is detected


                   A E R O F O I L   C A L C U L A T I O N:  2D
                   --------------------------------------------

AEROFOIL DATA
-------------

Aerofoil category is "conventional"

Aerofoil leading-edge is: rough

Thickness-to-chord ratio,                                t/c =  .1000
Location of boundary layer transition,                 xtr/c =  .0000
Aerofoil trailing-edge angle,                  Tau (degrees) =  .0000

Aerofoil coordinates:

                 j       xj/c       zuj/c       zlj/c

                 1     .00000      .00000      .00000
                 2     .01000      .01200     -.01000
                 3     .01250      .01300     -.01050
                 4     .02500      .01800     -.01400
                 5     .05000      .02500     -.01900
                 6     .10000      .03600     -.02500
                 7     .20000      .04950     -.03340
                 8     .30000      .05700     -.03790
                 9     .40000      .06070     -.03920
                10     .50000      .05920     -.03710
                11     .60000      .05220     -.03080
                12     .70000      .04130     -.02180
                13     .80000      .02780     -.01190
                14     .90000      .01330     -.00290
                15     .95000      .00620      .00010
                16     .99000      .00124      .00002
                17    1.00000      .00000      .00000


FLOW CONDITIONS
---------------

Number of Mach numbers          =  1
Mach number values, M           =  .1000

Number of Reynolds numbers      =  1
Reynolds number values, Rc      =  .6000E+07
```

```
OUTPUT DATA
===========

WARNING!!  See Section 2.2 of Item No. 84026 to find the
           meaning of "rough" in the context of the method.

Value of                              zu1.25/c =   .1300E-01
Trailing-edge angle,            Taua (degrees) =   9.515

Aerofoil zero-lift angle,      Alpha0 (radians) =  -.2737E-01
                                      (degrees) =  -1.568


Mach number, M  =  .1000
------------------------

----------------------------------------------------------------------------
Rc(x 1.E-6)  (a1)0 (per radian)    CL0    DeltaCL     FS       FM       CLm
----------------------------------------------------------------------------
   6.000            5.954         .163      .886     1.000    1.000    1.049
----------------------------------------------------------------------------

END OF OUTPUT  -------------------------------------------------------------
```

5.1.3 Example 3

This is Example 3 of Item No. 84026.

Input

```
EXAMPLE 3 of Item No.84026.
Input file I03A9315 Output file R03A9315

R03A9315
2D
?
S
0.1
0.3
0.430  0.0610
0
0.0     0.0      0.0
0.01    0.012   -0.010
0.0125  0.013   -0.0105
0.025   0.018   -0.014
0.05    0.025   -0.019
0.10    0.036   -0.025
0.20    0.0495  -0.0334
0.30    0.0570  -0.0379
0.40    0.0607  -0.0392
0.50    0.0592  -0.0371
0.60    0.0522  -0.0308
0.70    0.0413  -0.0218
0.80    0.0278  -0.0119
0.90    0.0133  -0.0029
0.95    0.0062   0.0001
0.99    0.00124  0.00002
1.0     0.0      0.0
4
0.1 0.2 0.3 0.4
1
6.E6
```

Output

```
******************************************************************************

 ESDU International plc

 Program              A9315

 ESDUpac Number:      A9315
 ESDUpac Title:       Maximum lift coefficient of plain aerofoils and wings
                      at subsonic speeds.
 Data Item Number:    93015
 Data Item Title:     Program for calculation of maximum lift coefficient
                      of plain aerofoils and wings at subsonic speeds.
 ESDUpac Version:     1.1    Issued January 1994

 See Data Item for full input/output specification and interpretation.

 ******************************************************************************

 INPUT DATA
 ==========

 EXAMPLE 3 of Item No.84026.
 Input file I03A9315 Output file R03A9315


 Output file name: R03A9315

 INPUT DATA ERRORS
 -----------------

       No error is detected


                   A E R O F O I L   C A L C U L A T I O N:  2D
                   --------------------------------------------

 AEROFOIL DATA
 -------------

 Aerofoil category to be decided by computer

 Aerofoil leading-edge is: smooth

 Thickness-to-chord ratio,                              t/c =  .1000
 Location of boundary layer transition,               xtr/c =  .3000
 Maximum dimensionless upper-surface ordinate,        zum/c =  .6100E-01
                        at chordwise location,        xum/c =  .4300
 Aerofoil trailing-edge angle,                Tau (degrees) =  .0000

 Aerofoil coordinates:

                 j       xj/c       zuj/c       zlj/c

                 1     .00000      .00000      .00000
                 2     .01000      .01200     -.01000
                 3     .01250      .01300     -.01050
                 4     .02500      .01800     -.01400
                 5     .05000      .02500     -.01900
                 6     .10000      .03600     -.02500
                 7     .20000      .04950     -.03340
```

```
                   8      .30000       .05700      -.03790
                   9      .40000       .06070      -.03920
                  10      .50000       .05920      -.03710
                  11      .60000       .05220      -.03080
                  12      .70000       .04130      -.02180
                  13      .80000       .02780      -.01190
                  14      .90000       .01330      -.00290
                  15      .95000       .00620       .00010
                  16      .99000       .00124       .00002
                  17     1.00000       .00000       .00000


FLOW CONDITIONS
---------------

Number of Mach numbers          =  4
Mach number values, M           =  .1000  .2000  .3000  .4000

Number of Reynolds numbers      =  1
Reynolds number values, Rc      =  .6000E+07


OUTPUT DATA
===========

WARNING!!  The calculation will assume that the aerofoil
           is "conventional" with FS = 1, see Section 6 of
           Item  No. 84026.

Value of                                zu1.25/c =   .1300E-01
Trailing-edge angle,              Taua (degrees) =   9.515

Aerofoil zero-lift angle,      Alpha0 (radians) =  -.2737E-01
                                       (degrees) =  -1.568


Mach number, M  =  .1000
------------------------

----------------------------------------------------------------------------
Rc(x 1.E-6)  (a1)0 (per radian)    CL0   DeltaCL     FS       FM       CLm
----------------------------------------------------------------------------
   6.000            6.007         .164    1.194     1.000    1.000    1.358
----------------------------------------------------------------------------


Mach number, M  =  .2000
------------------------

----------------------------------------------------------------------------
Rc(x 1.E-6)  (a1)0 (per radian)    CL0   DeltaCL     FS       FM       CLm
----------------------------------------------------------------------------
   6.000            6.007         .164    1.194     1.000     .925    1.257
----------------------------------------------------------------------------


Mach number, M  =  .3000
------------------------

----------------------------------------------------------------------------
Rc(x 1.E-6)  (a1)0 (per radian)    CL0   DeltaCL     FS       FM       CLm
----------------------------------------------------------------------------
```

```
   6.000             6.007          .164     1.194      1.000      .793     1.077
 ----------------------------------------------------------------------------


 Mach number, M  =  .4000
 ------------------------

 ----------------------------------------------------------------------------
 Rc(x 1.E-6)  (a1)0 (per radian)    CL0    DeltaCL      FS        FM        CLm
 ----------------------------------------------------------------------------
   6.000             6.007          .164     1.194      1.000      .708      .962
 ----------------------------------------------------------------------------

 END OF OUTPUT  -------------------------------------------------------------
```

5.1.4 Example 4

This is an example of a calculation for a modern aerofoil, NASA LS (1) – 0147 (previously NASA GA(W) – 1). The resulting values of C_{Lm} compare very well with test data in Derivation 11 of Item No. 84026.

Input

```
NASA LS(1)-0147 (i.e. NASA GA(W)-1)
NASA TM-X-72843. xtr/c=0.075, M=0.15, Rc=2.1, 4.3, 6.3 million.
NASA minimum roughness. Input file I04A9315 Output file R04A9315
R04A9315
2D
?
S
0.17
0.075
0.41  0.10510
6.9
0.0      0.0        0.0
0.01     0.02800   -0.01930
0.0125   0.03069   -0.02051
0.025    0.04165   -0.02691
0.05     0.05600   -0.03569
0.10     0.07309   -0.04700
0.20     0.09209   -0.05926
0.30     0.10169   -0.06448
0.40     0.10500   -0.06483
0.50     0.10269   -0.06091
0.60     0.09374   -0.05061
0.70     0.07639   -0.03383
0.80     0.05291   -0.01587
0.90     0.02639   -0.00352
0.95     0.01287   -0.00257
0.99     0.00230   -0.00600
1.0     -0.00074   -0.00783
1
0.15
3
2.1E6 4.3E6 6.3E6
```

Output

```
****************************************************************************

 ESDU International plc

 Program               A9315

 ESDUpac Number:       A9315
 ESDUpac Title:        Maximum lift coefficient of plain aerofoils and wings
                       at subsonic speeds.
 Data Item Number:     93015
 Data Item Title:      Program for calculation of maximum lift coefficient
                       of plain aerofoils and wings at subsonic speeds.
 ESDUpac Version:      1.1    Issued January 1994

 See Data Item for full input/output specification and interpretation.

 ****************************************************************************

 INPUT DATA
 ==========

 NASA LS(1)-0147 (i.e. NASA GA(W)-1)
 NASA TM-X-72843. xtr/c=0.075, M=0.15, Rc=2.1, 4.3, 6.3 million.
 NASA minimum roughness. Input file I04A9315 Output file R04A9315

 Output file name: R04A9315

 INPUT DATA ERRORS
 -----------------

       No error is detected


                   A E R O F O I L   C A L C U L A T I O N:  2D
                   --------------------------------------------

 AEROFOIL DATA
 -------------

 Aerofoil category to be decided by computer

 Aerofoil leading-edge is: smooth

 Thickness-to-chord ratio,                             t/c =  .1700
 Location of boundary layer transition,              xtr/c =  .7500E-01
 Maximum dimensionless upper-surface ordinate,       zum/c =  .1051
                        at chordwise location,       xum/c =  .4100
 Aerofoil trailing-edge angle,               Tau (degrees) =  6.900

 Aerofoil coordinates:

                 j       xj/c       zuj/c       zlj/c

                 1      .00000      .00000      .00000
                 2      .01000      .02800     -.01930
                 3      .01250      .03069     -.02051
                 4      .02500      .04165     -.02691
                 5      .05000      .05600     -.03569
                 6      .10000      .07309     -.04700
                 7      .20000      .09209     -.05926
```

8	.30000	.10169	-.06448
9	.40000	.10500	-.06483
10	.50000	.10269	-.06091
11	.60000	.09374	-.05061
12	.70000	.07639	-.03383
13	.80000	.05291	-.01587
14	.90000	.02639	-.00352
15	.95000	.01287	-.00257
16	.99000	.00230	-.00600
17	1.00000	-.00074	-.00783

```
FLOW CONDITIONS
---------------

Number of Mach numbers          =  1
Mach number values, M           =  .1500

Number of Reynolds numbers      =  3
Reynolds number values, Rc      =  .2100E+07  .4300E+07  .6300E+07


OUTPUT DATA
===========

WARNING!!  The aerofoil has (zu-zl)/zu =  .1133E+01 at  x/c = 0.9
                and                tb/c =  .7090E-02
           The calculation will therefore assume that the aerofoil is
           "modern" with FS > 1, see Section 6 of Item No. 84026.

Value of                                zu1.25/c =   .3069E-01
Trailing-edge angle,              Taua (degrees) =   13.69
Trailing-edge angle,              Tauu (degrees) =   10.10

Aerofoil zero-lift angle,       Alpha0 (radians) =  -.7802E-01
                                       (degrees) =  -4.470


Mach number, M  =  .1500
------------------------
```

Rc(x 1.E-6)	(a1)0 (per radian)	CL0	DeltaCL	FS	FM	CLm
2.100	5.813	.453	1.015*	1.087	.997	1.592
4.300	5.991	.467	1.180	1.113	.997	1.829
6.300	6.064	.473	1.251	1.129	.997	1.942

```
*  WARNING!!  The value of Delta CL should be treated with caution.
              It lies in "data less certain" region of Figure 2 of
              Item No. 84026.

END OF OUTPUT  -------------------------------------------------------------
```

5.2 Wings

5.2.1 Example 5

This is the Example of Item No. 89034 with additional values of Reynolds number.

Part I: Determination of η_p

Input

```
Example 1 of Item No. 89034.
Input file for Part I.
Input file I05A9315.1 Output file R05A9315.1
R05A9315.1
3D
?
8
0.4
0.25
25
1
0.3
```

Output

```
****************************************************************************
    ESDU International plc

 Program               A9315

 ESDUpac Number:       A9315
 ESDUpac Title:        Maximum lift coefficient of plain aerofoils and wings
                       at subsonic speeds.
 Data Item Number:     93015
 Data Item Title:      Program for calculation of maximum lift coefficient
                       of plain aerofoils and wings at subsonic speeds.
 ESDUpac Version:      1.1    Issued January 1994

 See Data Item for full input/output specification and interpretation.

 ****************************************************************************

 INPUT DATA
 ==========

 Example 1 of Item No. 89034.
 Input file for Part I.
 Input file I05A9315.1 Output file R05A9315.1

 Output file name: R05A9315.1

 INPUT DATA ERRORS
 -----------------

       No error is detected
```

```
                W I N G   C A L C U L A T I O N:   3D  (PART I)
                -----------------------------------------------

  WING PLANFORM GEOMETRY
  ----------------------

  Wing aspect ratio,                                           A =  8.000
  Chord line parameter,                                        n =  .2500
  Wing sweepback,                           Lambdan (degrees) =  25.00
  Wing taper ratio,                                      Lambda =  .4000


  CALCULATION OF Etap
  -------------------

                           ---------------------------
                              M       Etab       Etap
                           ---------------------------
                            .3000     .4376      .6876
                           ---------------------------

  END OF OUTPUT  -------------------------------------------------------------
```

Part II: Determination of C_{Lmax}

Input

```
Example 1 of Item No. 89034 for a range of Reynolds numbers.
Input file for Part II.
Input file I05A9315.2 Output file R05A9315.2
R05A9315.2
3D
?
8.
0.4
0.25
25.
1
0.3
0.12
0.3
0.384 0.07420
0.
0.0     0.0      0.0
0.01    0.01580 -0.01340
0.0125  0.01747 -0.01453
0.025   0.02420 -0.01908
0.05    0.03420 -0.02550
0.10    0.04780 -0.03358
0.20    0.06470 -0.04280
0.30    0.07280 -0.04606
0.40    0.07380 -0.04434
0.50    0.06890 -0.03856
0.60    0.05890 -0.02944
0.70    0.04550 -0.01876
0.80    0.03000 -0.00810
0.90    0.01420  0.00000
0.95    0.00680  0.00190
0.99    0.00140  0.00100
1.0     0.0      0.0
```

```
T,V
0.0     0.0      0.0
0.025   0.02102 -0.02102
0.05    0.02925 -0.02925
0.10    0.04039 -0.04039
0.20    0.05342 -0.05342
0.30    0.05930 -0.05930
0.40    0.05920 -0.05920
0.50    0.05370 -0.05370
0.60    0.04420 -0.04420
0.70    0.03210 -0.03210
0.80    0.01902 -0.01902
0.90    0.00707 -0.00707
0.95    0.00250 -0.00250
1.00    0.0      0.0
0.0     0.0      0.0
0.025   0.02414 -0.01918
0.05    0.03408 -0.02565
0.10    0.04760 -0.03381
0.20    0.06434 -0.04311
0.30    0.07237 -0.04645
0.40    0.07333 -0.04477
0.50    0.06841 -0.03900
0.60    0.05847 -0.02991
0.70    0.04510 -0.01918
0.80    0.02967 -0.00845
0.90    0.01396 -0.00017
0.95    0.00670  0.00173
1.00    0.0      0.0
-2.9333
8
2.E6 3.E6 4.E6 5.E6 6.E6 7.E6 8.E6 9.E6
```

Output

```
*****************************************************************************

 ESDU International plc

 Program              A9315

 ESDUpac Number:      A9315
 ESDUpac Title:       Maximum lift coefficient of plain aerofoils and wings
                      at subsonic speeds.
 Data Item Number:    93015
 Data Item Title:     Program for calculation of maximum lift coefficient
                      of plain aerofoils and wings at subsonic speeds.
 ESDUpac Version:     1.1    Issued January 1994

 See Data Item for full input/output specification and interpretation.

 *****************************************************************************

 INPUT DATA
 ==========

 Example 1 of Item No. 89034 for a range of Reynolds numbers.
 Input file for Part II.
 Input file I05A9315.2 Output file R05A9315.2
```

```
Output file name: R05A9315.2

INPUT DATA ERRORS
-----------------

      No error is detected


              W I N G   C A L C U L A T I O N:   3D (PART II)
              -----------------------------------------------

WING PLANFORM GEOMETRY
----------------------

Wing aspect ratio,                                        A =  8.000
Chord line parameter,                                     n =  .2500
Wing sweepback,                         Lambdan (degrees) =  25.00
Wing taper ratio,                                    Lambda =  .4000


DATA FOR AEROFOIL AT Etap
-------------------------

Aerofoil category to be decided by computer

M =  .3000  (Etap =  .6876)
---------------------------

Thickness-to-chord ratio,                               t/c =  .1200
Location of boundary layer transition,                xtr/c =  .3000
Maximum dimensionless upper-surface ordinate,         zum/c =  .7420E-01
                       at chordwise location,         xum/c =  .3840
Aerofoil trailing-edge angle,                 Tau (degrees) =  .0000

Aerofoil coordinates:

                j       xj/c       zuj/c       zlj/c

                1     .00000      .00000      .00000
                2     .01000      .01580     -.01340
                3     .01250      .01747     -.01453
                4     .02500      .02420     -.01908
                5     .05000      .03420     -.02550
                6     .10000      .04780     -.03358
                7     .20000      .06470     -.04280
                8     .30000      .07280     -.04606
                9     .40000      .07380     -.04434
               10     .50000      .06890     -.03856
               11     .60000      .05890     -.02944
               12     .70000      .04550     -.01876
               13     .80000      .03000     -.00810
               14     .90000      .01420      .00000
               15     .95000      .00680      .00190
               16     .99000      .00140      .00100
               17    1.00000      .00000      .00000

WING TWIST AND CAMBER GEOMETRY
------------------------------

Spanwise geometrical twist type:        Linear or non-linear (monotonic)
Spanwise variation of section camber:   Linear or non-linear (monotonic)
```

Aerofoil data at root:

j	xj/c	zuj/c	zlj/c
1	.00000	.00000	.00000
2	.02500	.02102	-.02102
3	.05000	.02925	-.02925
4	.10000	.04039	-.04039
5	.20000	.05342	-.05342
6	.30000	.05930	-.05930
7	.40000	.05920	-.05920
8	.50000	.05370	-.05370
9	.60000	.04420	-.04420
10	.70000	.03210	-.03210
11	.80000	.01902	-.01902
12	.90000	.00707	-.00707
13	.95000	.00250	-.00250
14	1.00000	.00000	.00000

Aerofoil data at 2/3 span:

j	xj/c	zuj/c	zlj/c
1	.00000	.00000	.00000
2	.02500	.02414	-.01918
3	.05000	.03408	-.02565
4	.10000	.04760	-.03381
5	.20000	.06434	-.04311
6	.30000	.07237	-.04645
7	.40000	.07333	-.04477
8	.50000	.06841	-.03900
9	.60000	.05847	-.02991
10	.70000	.04510	-.01918
11	.80000	.02967	-.00845
12	.90000	.01396	-.00017
13	.95000	.00670	.00173
14	1.00000	.00000	.00000

```
       Geometrical twist,              Delta2/3 (degrees) =  -2.933

FLOW CONDITIONS
---------------

Number of Reynolds numbers     =  8
Reynolds number values,  Rcbb  =  .2000E+07  .3000E+07  .4000E+07  .5000E+07
                                  .6000E+07  .7000E+07  .8000E+07  .9000E+07



OUTPUT DATA
===========

Wing leading-edge sweep angle,        Lambda0   (degrees) = 27.469
Wing quarter-chord sweep angle,       Lambda1/4 (degrees) = 25.000
Wing mid-chord sweep angle,           Lambda1/2 (degrees) = 22.428
Wing trailing-edge sweep angle,       Lambda1   (degrees) = 16.993
Wing taper parameter,                              Kappa =  .4286


Mach number, M  =  .3000
```

```
------------------------

WARNING!!  The calculation will assume that the aerofoil
           is "conventional" with FS = 1, see Section 6 of
           Item  No. 84026.

Reduced aspect ratio,                                BetaA =  7.632
Spanwise centre of pressure position,                 Etab =  .4376
Spanwise location of peak loading due to incidence,  Etap =  .6876
Peak value of normalised local lift coefficient,      Mup =  1.152
Ratio of local chord to m.a.c. at Etap,            cp/cbb =  .7908
Value of zu1.25/c for section at Etap,        (zu1.25/c)p =  .1747E-01
Leading-edge shape parameter for section at Etap,   Zetap =  .1969E-01

Upper-surface angle parameter for section at Etap,
                                    tan(Tauu)sec(Lambda1) =  .1260

Zero-lift angle for section at:
           root,                         Alpha0r (degrees) =   .000
           2/3span,                    Alpha02/3 (degrees) =  -2.09

Effective tip twist angle,               Deltaet (degrees) =  -1.27


----------------------------------------------------------------------------
Rcbb    Rcp    Rcpcos^2  Delta   CLm   CLm/Mup Delta  Delta  Delta  Delta  CLmax
 x       x     (Lambda0)  CL                    CLM    CLR    CL     CLT
1.E-6  1.E-6   x 1.E-6                                       Lambda
----------------------------------------------------------------------------
2.000  1.582  1.245   1.129* 1.378  1.196  -.116   .000  -.029   .002  1.053
3.000  2.372  1.868   1.212* 1.464  1.272  -.116   .000  -.029   .002  1.129
4.000  3.163  2.490   1.276* 1.532  1.330  -.116   .000  -.029   .002  1.187
5.000  3.954  3.113   1.326  1.583  1.375  -.116   .000  -.029   .002  1.232
6.000  4.745  3.735   1.363  1.622  1.408  -.116   .000  -.029   .002  1.265
7.000  5.535  4.358   1.392  1.651  1.434  -.116   .000  -.029   .002  1.291
8.000  6.326  4.980   1.415  1.675  1.454  -.116   .000  -.029   .002  1.312
9.000  7.117  5.603   1.433  1.694  1.471  -.116   .000  -.029   .002  1.328
----------------------------------------------------------------------------

*  WARNING!!  The value of Delta CL should be treated with caution.
              It lies in "data less certain" region of Figure 2 of
              Item No. 84026.

END OF OUTPUT  -------------------------------------------------------------
```

5.2.2 Example 6

This is an example of a calculation for a medium aspect ratio, highly swept wing with linear - lofted geometric twist and varying spanwise camber.

$$A = 6,\ \Lambda_{1/4} = 45°,\ \lambda = 0.6\,.$$

$$\left.\begin{array}{l}\text{Root section NACA 4412}\\ \\ \text{Tip section NACA 2412}\end{array}\right\}\ \text{camber varying linearly across span.}$$

Linear-lofted twist with $\delta_t = -2°$, giving $\delta_{2/3} = -1.091°$.

$M = 0.6$, $R_c = 2$, 4 and 6×10^6.

Part I: Determination of η_p

Input

```
A = 6, Lambda1/4 = 45 deg., Lambda = 0.6. Root section: NACA 4412, tip
section: NACA 2412, camber varying linearly spanwise. Linear-lofted twist
with Deltat = -2 deg. M = 0.6, Rcbb = 2, 4, 6E6.
R06A9315.1
3D
N
4
0.6
0.25
45
1
0.6
```

Output

```
*****************************************************************************

 ESDU International plc

 Program              A9315

 ESDUpac Number:      A9315
 ESDUpac Title:       Maximum lift coefficient of plain aerofoils and wings
                      at subsonic speeds.
 Data Item Number:    93015
 Data Item Title:     Program for calculation of maximum lift coefficient
                      of plain aerofoils and wings at subsonic speeds.
 ESDUpac Version:     1.1    Issued January 1994

 See Data Item for full input/output specification and interpretation.

 *****************************************************************************

 INPUT DATA
 ==========

 A = 6, Lambda1/4 = 45 deg., Lambda = 0.6. Root section: NACA 4412, tip
 section: NACA 2412, camber varying linearly spanwise. Linear-lofted twist
 with Deltat = -2 deg. M = 0.6, Rcbb = 2, 4, 6E6.

 Output file name: R06A9315.1

 INPUT DATA ERRORS
 -----------------

       No error is detected
```

```
                W I N G   C A L C U L A T I O N:   3D  (PART I)
                -----------------------------------------------

 WING PLANFORM GEOMETRY
 ----------------------

 Wing aspect ratio,                                          A =  4.000
 Chord line parameter,                                       n =  .2500
 Wing sweepback,                            Lambdan (degrees) =  45.00
 Wing taper ratio,                                      Lambda =  .6000


 CALCULATION OF Etap
 -------------------

                       ---------------------------
                          M        Etab       Etap
                       ---------------------------
                        .6000      .4521      .6297
                       ---------------------------

 END OF OUTPUT  -------------------------------------------------------------
```

Part II: Determination of C_{Lmax}

Input

```
A = 6, Lambda1/4 = 45 deg., Lambda = 0.6. Root section: NACA 4412, tip
section: NACA 2412, camber varying linearly spanwise. Linear-lofted twist
with Deltat = -2 deg. M = 0.6, Rcbb = 2, 4, 6E6.
R06A9315.2
3D
N
4
0.6
0.25
45
1
0.6
0.12
0.2
0.340 0.0863
15.01
0.0     0.0      0.0
0.01    0.01846 -0.01576
0.0125  0.02063 -0.01725
0.025   0.02947 -0.02283
0.05    0.04197 -0.02913
0.10    0.05882 -0.03484
0.20    0.07793 -0.03683
0.30    0.08571 -0.03433
0.40    0.08544 -0.03062
0.50    0.07958 -0.02630
0.60    0.06999 -0.02127
0.70    0.05719 -0.01609
0.80    0.04145 -0.01101
0.90    0.02285 -0.00611
0.95    0.01245 -0.00369
0.99    0.00350 -0.00174
1.0     0.00126 -0.00126
```

```
T,V
0.0     0.0      0.0
0.025   0.03099 -0.02131
0.05    0.04492 -0.02618
0.10    0.06433 -0.02933
0.20    0.08738 -0.02738
0.30    0.09752 -0.02252
0.40    0.09803 -0.01803
0.50    0.09183 -0.01405
0.60    0.08118 -0.01008
0.70    0.06664 -0.00664
0.80    0.04845 -0.00401
0.90    0.02670 -0.00226
0.95    0.01446 -0.00168
1.0     0.00126 -0.00126
0.0     0.0      0.0
0.025   0.02938 -0.02292
0.05    0.04180 -0.02930
0.10    0.05850 -0.03516
0.20    0.07738 -0.03738
0.30    0.08502 -0.03502
0.40    0.08470 -0.03136
0.50    0.07886 -0.02702
0.60    0.06933 -0.02193
0.70    0.05664 -0.01664
0.80    0.04104 -0.01142
0.90    0.02263 -0.00633
0.95    0.01233 -0.00381
1.0     0.00126 -0.00126
-1.091
3
2.E6 4.E6 6.E6
```

Output

```
*****************************************************************************

 ESDU International plc

 Program              A9315

 ESDUpac Number:      A9315
 ESDUpac Title:       Maximum lift coefficient of plain aerofoils and wings
                      at subsonic speeds.
 Data Item Number:    93015
 Data Item Title:     Program for calculation of maximum lift coefficient
                      of plain aerofoils and wings at subsonic speeds.
 ESDUpac Version:     1.1    Issued January 1994

 See Data Item for full input/output specification and interpretation.

 *****************************************************************************

 INPUT DATA
 ==========

 A = 6, Lambda1/4 = 45 deg., Lambda = 0.6. Root section: NACA 4412, tip
 section: NACA 2412, camber varying linearly spanwise. Linear-lofted twist
 with Deltat = -2 deg. M = 0.6, Rcbb = 2, 4, 6E6.

 Output file name: R06A9315.2
```

```
INPUT DATA ERRORS
-----------------

      No error is detected


              W I N G   C A L C U L A T I O N:   3D (PART II)
              -----------------------------------------------

WING PLANFORM GEOMETRY
----------------------

Wing aspect ratio,                                         A =  4.000
Chord line parameter,                                      n =  .2500
Wing sweepback,                         Lambdan (degrees) =  45.00
Wing taper ratio,                                     Lambda =  .6000


DATA FOR AEROFOIL AT Etap
-------------------------

Aerofoil category is "conventional"

M =  .6000  (Etap =  .6297)
---------------------------

Thickness-to-chord ratio,                                t/c =  .1200
Location of boundary layer transition,                 xtr/c =  .2000
Maximum dimensionless upper-surface ordinate,          zum/c =  .8630E-01
                       at chordwise location,          xum/c =  .3400
Aerofoil trailing-edge angle,                  Tau (degrees) =  15.01

Aerofoil coordinates:

                j       xj/c       zuj/c       zlj/c

                1     .00000      .00000      .00000
                2     .01000      .01846     -.01576
                3     .01250      .02063     -.01725
                4     .02500      .02947     -.02283
                5     .05000      .04197     -.02913
                6     .10000      .05882     -.03484
                7     .20000      .07793     -.03683
                8     .30000      .08571     -.03433
                9     .40000      .08544     -.03062
               10     .50000      .07958     -.02630
               11     .60000      .06999     -.02127
               12     .70000      .05719     -.01609
               13     .80000      .04145     -.01101
               14     .90000      .02285     -.00611
               15     .95000      .01245     -.00369
               16     .99000      .00350     -.00174
               17    1.00000      .00126     -.00126

WING TWIST AND CAMBER GEOMETRY
------------------------------

Spanwise geometrical twist type:        Linear or non-linear (monotonic)
Spanwise variation of section camber:   Linear or non-linear (monotonic)

Aerofoil data at root:
```

j	xj/c	zuj/c	zlj/c
1	.00000	.00000	.00000
2	.02500	.03099	-.02131
3	.05000	.04492	-.02618
4	.10000	.06433	-.02933
5	.20000	.08738	-.02738
6	.30000	.09752	-.02252
7	.40000	.09803	-.01803
8	.50000	.09183	-.01405
9	.60000	.08118	-.01008
10	.70000	.06664	-.00664
11	.80000	.04845	-.00401
12	.90000	.02670	-.00226
13	.95000	.01446	-.00168
14	1.00000	.00126	-.00126

Aerofoil data at 2/3 span:

j	xj/c	zuj/c	zlj/c
1	.00000	.00000	.00000
2	.02500	.02938	-.02292
3	.05000	.04180	-.02930
4	.10000	.05850	-.03516
5	.20000	.07738	-.03738
6	.30000	.08502	-.03502
7	.40000	.08470	-.03136
8	.50000	.07886	-.02702
9	.60000	.06933	-.02193
10	.70000	.05664	-.01664
11	.80000	.04104	-.01142
12	.90000	.02263	-.00633
13	.95000	.01233	-.00381
14	1.00000	.00126	-.00126

```
       Geometrical twist,              Delta2/3 (degrees) =  -1.091

FLOW CONDITIONS
---------------

Number of Reynolds numbers     =  3
Reynolds number values,  Rcbb  =  .2000E+07  .4000E+07  .6000E+07


OUTPUT DATA
===========

Wing leading-edge sweep angle,        Lambda0   (degrees) = 46.736
Wing quarter-chord sweep angle,       Lambda1/4 (degrees) = 45.000
Wing mid-chord sweep angle,           Lambda1/2 (degrees) = 43.152
Wing trailing-edge sweep angle,       Lambda1   (degrees) = 39.094
Wing taper parameter,                               Kappa =  .4583


Mach number, M  =  .6000
------------------------

Reduced aspect ratio,                               BetaA =  3.200
```

```
Spanwise centre of pressure position,                 Etab =  .4521
Spanwise location of peak loading due to incidence,  Etap =  .6297
Peak value of normalised local lift coefficient,      Mup =  1.121
Ratio of local chord to m.a.c. at Etap,            cp/cbb =  .9161
Value of zu1.25/c for section at Etap,        (zu1.25/c)p =  .2063E-01
Leading-edge shape parameter for section at Etap,   Zetap =  .3010E-01

Upper-surface angle parameter for section at Etap,
                                    tan(Tauu)sec(Lambda1) =  .1685

Zero-lift angle for section at:
           root,                        Alpha0r (degrees) =  -4.13
           2/3span,                   Alpha02/3 (degrees) =  -2.75

Effective tip twist angle,              Deltaet (degrees) =  -3.70
```

Rcbb x 1.E-6	Rcp x 1.E-6	Rcpcos^2 (Lambda0) x 1.E-6	Delta CL	CLm	CLm/Mup	Delta CLM	Delta CLR	Delta CL Lambda	Delta CLT	CLmax
2.000	1.832	.861	.887*	1.288	1.149	-.096	.177	.039	-.022	1.246
4.000	3.664	1.721	1.013*	1.426	1.273	-.096	.112	.039	-.022	1.305
6.000	5.496	2.582	1.103*	1.521	1.357	-.096	.070	.039	-.022	1.347

```
*  WARNING!!  The value of Delta CL should be treated with caution.
              It lies in "data less certain" region of Figure 2 of
              Item No. 84026.

END OF OUTPUT  ------------------------------------------------------------
```

THE PREPARATION OF THIS DATA ITEM

The work on this particular Data Item was monitored and guided by the Aerodynamics Committee, which first met in 1942 and now has the following membership:

Chairman
Mr H.C. Garner – Independent

Vice-Chairman
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Miss J. Willaume	–	Aérospatiale, Toulouse, France.

* Corresponding Member

The technical work in the assessment of the available information and the construction and subsequent development of the Data Item was carried out by

Mr P.D. Chappell – Head of Aircraft Aerodynamics Group.

The program was prepared under contract to ESDU by

Prof. G.V. Groves.