



Excrescence drag magnification factors at the drag-rise condition for aerofoils with a specified form of upper-surface pressure distribution

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EXCRESCENCE DRAG MAGNIFICATION FACTORS AT THE DRAG-RISE CONDITION FOR AEROFOILS WITH A SPECIFIED FORM OF UPPER-SURFACE PRESSURE DISTRIBUTION

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EXCRESCENCE DRAG MAGNIFICATION FACTORS AT THE DRAG-RISE CONDITION FOR AEROFOILS WITH A SPECIFIED FORM OF UPPER-SURFACE PRESSURE DISTRIBUTION

1. NOTATION AND UNITS

		SI	British
C_L	lift coefficient, (lift) / $q_\infty c$		
C_p	pressure coefficient		
c	aerofoil chord	m	ft
M	Mach number		
m_d	excrecence drag magnification factor		
q	kinetic pressure, $\frac{1}{2}\rho u^2$	N/m ²	lbf/ft ²
R_c	Reynolds number based on aerofoil chord and free-stream flow properties at drag-rise Mach number		
T	static temperature	K	K
t	aerofoil maximum thickness	m	ft
u	velocity	m/s	ft/s
x	streamwise co-ordinate, measured along chord from leading edge	m	ft
x_R	chordwise location of end of upper-surface pressure plateau ("roof-top")	m	ft
x_{tr}	chordwise location of boundary-layer transition	m	ft
α	aerofoil angle of incidence	deg	deg
θ	boundary-layer momentum thickness	m	ft
ρ	density of air	kg/m ³	slug/ft ³
<i>Subscripts</i>			
D	denotes value at drag-rise (<i>i.e.</i> design) condition		
E	denotes value at excrecence location		
l	denotes lower-surface value		

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T	denotes value at trailing edge
u	denotes upper-surface value
1	denotes value at edge of boundary layer
∞	denotes free-stream value

2. INTRODUCTION

This Item presents the chordwise variation at the drag-rise condition of the excrescence drag magnification factor, m_d , over the upper and lower surfaces of aerofoils designed to combine a high drag-rise Mach number with a high lift coefficient for a given thickness/chord ratio. The data can be considered as typical of aerofoils designed for such a purpose. The aerofoils are members of the families described in Item No. 71020 (Derivation 1) which gives details of geometry and pressure distribution at the design condition. The corresponding profile drag data are given in Item No. 67011 (Reference 6).

The data for m_d were obtained using the computer program for the method for turbulent boundary layers given in Item No. 91028 (Derivation 3) which is based on a compressible flow form of the Nash and Bradshaw equation (Derivation 2) with a compatible method for the boundary-layer momentum thickness development. Section 3 gives a brief résumé of the aerofoil design philosophy, describes the aerofoil designation system used and lists the aerofoils. Section 4 outlines the basis of the data, which are presented in two forms. Detailed normalised local flow properties together with boundary-layer momentum thickness ratio and m_d data for the upper and lower surfaces are given in tabulated form while the chordwise variations of m_d are also presented graphically. Section 5 discusses a number of features of the chordwise variations of m_d as affected by the aerofoil design variables. Section 6 gives the Derivation and References and Section 7 illustrates, by way of an example, a typical use of the data.

3. THE AEROFOIL FAMILIES

3.1 Design Philosophy

The aerofoils for which data are provided in this Item are members of the families (Derivation 1) designed to combine a high drag-rise Mach number, M_D , with a high lift coefficient, C_{LD} , for a given thickness/chord ratio, t/c . The aerofoils are specified primarily by the form of the upper-surface pressure distribution at the drag-rise condition and the form of the thickness distribution. Of lesser importance are the values ascribed, firstly, to the Reynolds number, R_c , based on chord length and free-stream flow properties at the drag-rise Mach number and, secondly, to the location of boundary-layer transition, x_{tr}/c , on each surface. Therefore, for a specified combination of M_D and C_{LD} (and, strictly, for given values of R_c and x_{tr}/c) there is a unique camber line and incidence, α_D , associated with the chosen form of thickness distribution and t/c .

The particular form of upper-surface pressure distribution chosen to comply with the conditions for a high value of M_D in conjunction with a high C_{LD} is the flat "roof-top" form in which a pressure plateau extends from the immediate vicinity of the leading edge to a given chordwise station x_R/c , followed by a linear rise to the trailing edge. The associated local Mach number over that part of the upper surface encompassed by the roof top is taken as 1.02. The philosophy behind the design, and the design process itself, are described in detail in Item No. 71019 (Reference 8) in the Transonic Aerodynamics Sub-series.

3.2 Aerofoil Designation

Any aerofoil corresponding to the general specification described in Section 3.1 is, as noted, primarily described by the thickness form and the values of t/c , x_R/c and M_D ; the values of R_c and x_{tr}/c on each surface having a minor influence. The aerofoils are designated by means of identifiers relating to each of these seven parameters. The aerofoils of Item No. 71020 are based on the RAE 101 to 104 series of thickness distributions (Reference 4), which have a wedge-shaped trailing-edge and are somewhat similar in form to the NACA 6A-series (Reference 5). An example of the aerofoil designation system is

$$102 - 10 - 40 - 75 \quad (7.0, 05, 05)$$

where the seven identifiers taken in sequence have the following meaning

102 denotes RAE 102 series thickness distribution

$$10 = 100 \, t/c \quad (i.e. \, t/c = 0.10)$$

$$40 = 100 \, x_R/c \quad (i.e. \, x_R/c = 0.40)$$

$$75 = 100 \, M_D \quad (i.e. \, M_D = 0.75)$$

$$7 = \text{index of } R_c \text{ expressed to a base of 10 } (i.e. \, R_c = 10^7)$$

$$05 = 100 \, x_{tru}/c \quad (i.e. \, x_{tru}/c = 0.05)$$

$$05 = 100 \, x_{trl}/c \quad (i.e. \, x_{trl}/c = 0.05).$$

For convenience, the bracketed numbers in the designation are omitted because they are invariant throughout, *i.e.* the aerofoils were all designed for $R_c = 10^7$ and $x_{tr}/c = 0.05$ on both surfaces.

The aerofoils, taken from Item No. 71020, are listed in Table 3.1 together with their design values of incidence, α_D , and lift coefficient, C_{LD} , *i.e.* values at the drag-rise condition.

TABLE 3.1 Aerofoils taken from Item No. 71020

<i>Aerofoil Designation</i>	<i>Design Incidence α_D (degrees)</i>	<i>Lift Coefficient C_{LD}</i>
101 - 16 - 30 - 60	3.95	0.97
101 - 06 - 30 - 65	3.27	0.98
101 - 10 - 30 - 65	2.98	0.82
101 - 14 - 30 - 65	2.74	0.65
101 - 16 - 30 - 65	2.64	0.57
101 - 06 - 30 - 70	2.23	0.68
101 - 10 - 30 - 70	1.95	0.50
101 - 14 - 30 - 70	1.69	0.31
101 - 06 - 30 - 75	1.41	0.43
101 - 10 - 30 - 75	1.12	0.22
101 - 06 - 30 - 80	0.77	0.21
102 - 10 - 40 - 65	3.03	0.92
102 - 14 - 40 - 65	2.85	0.76
102 - 16 - 40 - 65	2.78	0.68
102 - 06 - 40 - 70	2.25	0.76
102 - 10 - 40 - 70	2.02	0.58
102 - 14 - 40 - 70	1.83	0.40
102 - 16 - 40 - 70	1.75	0.30
102 - 06 - 40 - 75	1.45	0.49
102 - 10 - 40 - 75	1.21	0.28
102 - 06 - 40 - 80	0.82	0.25
103 - 14 - 50 - 65	2.90	0.87
103 - 16 - 50 - 65	2.86	0.78
103 - 06 - 50 - 70	2.21	0.83
103 - 10 - 50 - 70	2.04	0.67
103 - 14 - 50 - 70	1.93	0.49
103 - 16 - 50 - 70	1.88	0.40
103 - 06 - 50 - 75	1.46	0.55
103 - 10 - 50 - 75	1.27	0.35
103 - 06 - 50 - 80	0.86	0.30
104 - 14 - 60 - 65	2.87	0.94
104 - 16 - 60 - 65	2.84	0.86
104 - 06 - 60 - 70	2.10	0.91
104 - 10 - 60 - 70	2.00	0.75
104 - 14 - 60 - 70	1.95	0.56
104 - 16 - 60 - 70	1.91	0.47
104 - 06 - 60 - 75	1.40	0.61
104 - 10 - 60 - 75	1.29	0.42
104 - 14 - 60 - 75	1.21	0.20
104 - 06 - 60 - 80	0.85	0.34
104 - 06 - 60 - 85	0.40	0.08

4. BASIS OF DATA

The data for the variation of m_d across the upper and lower surfaces of the 41 aerofoils listed in Table 3.1 at their drag-rise condition were obtained by means of the computer program for the method of Item No. 91028 (Derivation 3). The program evaluates the compressible flow form of the Nash and Bradshaw equation

$$m_d = \frac{\rho_{1T}}{\rho_\infty} \left(\frac{u_{1E}}{u_\infty} \right)^{(4.2 - 0.6M_\infty^2)} \left(\frac{u_\infty}{u_{1T}} \right)^{(1 - M_\infty^2)} \left(\frac{\theta_E}{\theta_T} \right)^{0.2} \quad (4.1)$$

and assumes that the boundary layer is turbulent from the aerofoil leading edge ($x_{tr}/c = 0$) with the momentum thickness ratio, θ_E/θ_T , being given by means of the Spence method adapted to compressible flow and with the required flow parameters being determined from the isentropic flow relationships (see Derivation 3) for a given pressure distribution. Therefore, the only input data required were the free-stream Mach number for the drag-rise condition, $M_\infty = M_D$, and the pressure distribution for the relevant surface taken from Item No. 71020. As noted in Item No. 91028, the values of m_d determined from Equation (4.1) are independent of Reynolds number. Further, the method ignores the minor effect of the small extent of laminar flow over the first 5% of the chord taken in the design of the aerofoils. The resulting values of m_d , together with the momentum thickness ratio and normalised local flow properties across the upper and lower surfaces of each aerofoil are presented in tabular form in Tables 8.1 to 8.41. The data for m_d are also presented graphically in Figures 1 to 18 in which, for convenience, the data for a given combination of M_D and x_R/c are collected together in each Figure. The values of m_d for $x/c > 0.8$ are given as broken lines, indicating a need for caution in their use, see Section 5.1 of Item No. 91028. As an aid in locating the relevant Table and Figure for a particular aerofoil Table 4.1 gives an index.

TABLE 4.1 Index to tabulated and graphical data

<i>Aerofoil Designation</i>	<i>Table No.</i>	<i>Figure No.</i>
101 - 16 - 30 - 60	8.1	1
101 - 06 - 30 - 65	8.2	2
101 - 10 - 30 - 65	8.3	2
101 - 14 - 30 - 65	8.4	2
101 - 16 - 30 - 65	8.5	2
101 - 06 - 30 - 70	8.6	6
101 - 10 - 30 - 70	8.7	6
101 - 14 - 30 - 70	8.8	6
101 - 06 - 30 - 75	8.9	10
101 - 10 - 30 - 75	8.10	10
101 - 06 - 30 - 80	8.11	14
102 - 10 - 40 - 65	8.12	3
102 - 14 - 40 - 65	8.13	3
102 - 16 - 40 - 65	8.14	3
102 - 06 - 40 - 70	8.15	7
102 - 10 - 40 - 70	8.16	7
102 - 14 - 40 - 70	8.17	7
102 - 16 - 40 - 70	8.18	7
102 - 06 - 40 - 75	8.19	11
102 - 10 - 40 - 75	8.20	11
102 - 06 - 40 - 80	8.21	15
103 - 14 - 50 - 65	8.22	4
103 - 16 - 50 - 65	8.23	4
103 - 06 - 50 - 70	8.24	8
103 - 10 - 50 - 70	8.25	8
103 - 14 - 50 - 70	8.26	8
103 - 16 - 50 - 70	8.27	8
103 - 06 - 50 - 75	8.28	12
103 - 10 - 50 - 75	8.29	12
103 - 06 - 50 - 80	8.30	16
104 - 14 - 60 - 65	8.31	5
104 - 16 - 60 - 65	8.32	5
104 - 06 - 60 - 70	8.33	9
104 - 10 - 60 - 70	8.34	9
104 - 14 - 60 - 70	8.35	9
104 - 16 - 60 - 70	8.36	9
104 - 06 - 60 - 75	8.37	13
104 - 10 - 60 - 75	8.38	13
104 - 14 - 60 - 75	8.39	13
104 - 06 - 60 - 80	8.40	17
104 - 06 - 60 - 85	8.41	18

5. INFLUENCE OF SYSTEMATIC CHANGES IN PRIMARY DESIGN VARIABLES

5.1 General

In Item No. 91028 it is noted that the parameter having by far the most significant effect on m_d is the local velocity at the excrescence location, mainly due to the magnitude of the index to the term u_{1E}/u_∞ in Equation (4.1) compared to that on the momentum thickness ratio term, θ_E/θ_T . That being so, it follows that those design variables having a major effect on local pressure, and hence velocity, will also have a major effect on m_d . It also follows from a study of Equation (4.1) that maximum values of m_d will occur in regions of high local velocity coupled with large values of momentum thickness. From Tables 8.1 to 8.41 and Figures 1 to 18 it is seen that, typically, such regions occur at a chordwise location corresponding to the end of an upper-surface roof-top, and often at the same chordwise location on the lower surface, depending upon the magnitude of t/c . An excrescence located in such regions would obviously suffer a higher drag penalty than when located elsewhere.

A study of Figures 1 to 18 reveals a number of features in the chordwise variation of m_d as affected by systematic changes in any one of the primary design variables, t/c , x_R/c and M_D as the other two are held constant.

5.2 Variation of t/c

The design specification for the upper-surface pressure distribution means that it depends mainly on the two parameters M_D and x_R/c . Consequently, Figures 1 to 18 show that the variation of m_d for the upper surface of the aerofoils with given values of M_D and x_R/c is very largely the same for wide variations of t/c . The minor variation with t/c is mainly dictated by the trailing-edge pressure variation with t/c .

In contrast, the Figures also show that the lower-surface chordwise variation of m_d for fixed values of M_D and x_R/c is very heavily dependent on t/c , with a pronounced increase in m_d , especially in the region around $x = x_R$, as t/c increases. This tendency is a direct reflection of the tendency of the loading to concentrate more over the forward and rearward parts of the aerofoil as t/c increases with, typically, an associated change in the camber line shape from convex (wholly positive camber) at low t/c to an S-shape (negative camber forward, positive rearward) at high t/c , as discussed in Item No. 71020. These changes also result in a reduction of C_{LD} as t/c increases, as can be seen from Table 3.1 for given values of M_D and x_R/c .

Sketch 5.1 illustrates the trends with variation of t/c taking $x_R/c = 0.4$ and $M_D = 0.70$ as typical.

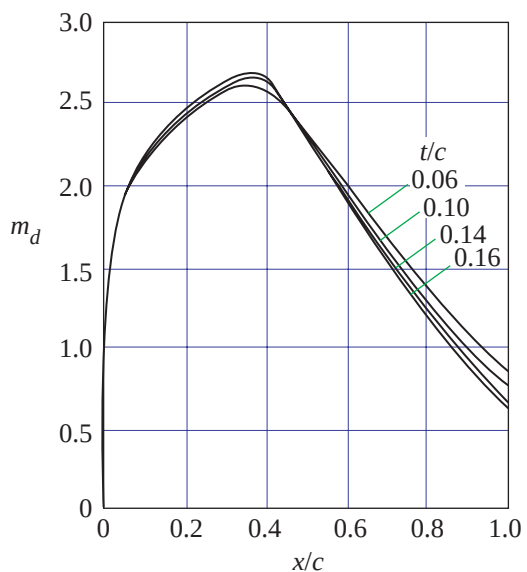
5.3 Variation of x_R/c

If x_R/c is varied, with M_D and t/c fixed, then the variation of m_d over the upper surface up to $x = x_R$ is virtually independent of x_R/c as can be seen from Figures 2 to 5, 6 to 9, 10 to 13 or 14 to 17. This is a direct reflection of the constant pressure over the roof-top being dependent only on M_D . Over the upper surface, for $x > x_R$, m_d decreases approximately linearly to the trailing edge, at which point it is substantially independent of x_R/c . For the lower surface with fixed values of M_D and t/c , as x_R/c is increased the same groups of Figures show that up to $x = x_R$ the values of m_d decrease, reflecting the corresponding trends in pressure distribution discussed in Item No. 71020. For $x > x_R$ the trends in m_d with x/c are very similar to those for the upper surface, but less linear, especially for the larger values of x_R/c and t/c .

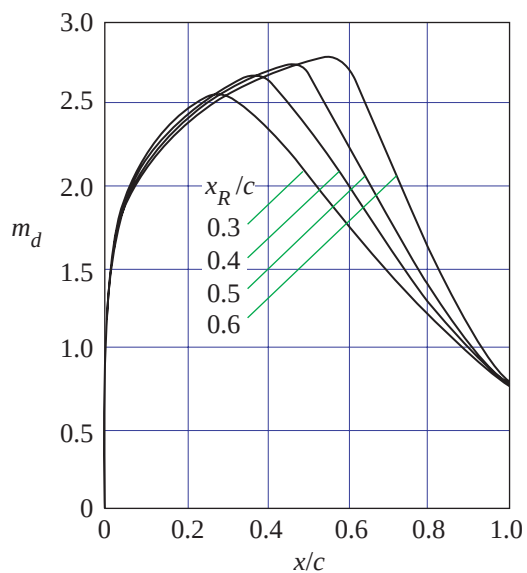
Sketch 5.2 illustrates the trends with variation of x_R/c taking $M_D = 0.70$ and $t/c = 0.10$ as typical.

5.4 Variation of M_D

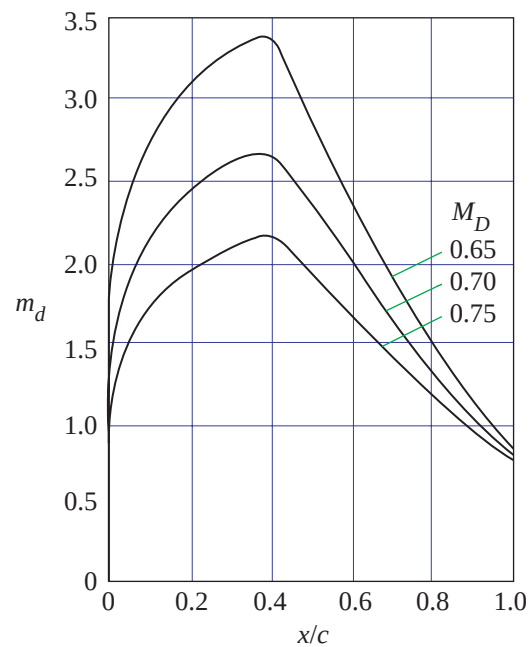
The effects of varying M_D for fixed values of x_R/c and t/c are very similar for any combination of those two parameters. A typical example is given for $x_R/c = 0.4$ and $t/c = 0.10$ in Figures 3, 7 and 11 for M_D varying from 0.65 to 0.75, the data for which are reproduced in Sketch 5.3. That sketch shows that, for a given value of x/c the variation of m_d on either surface is roughly linear with M_D , with the upper-surface values decreasing with increase in M_D and the lower-surface values increasing with increase in M_D . For a given combination of x_R/c and t/c the trailing-edge values of m_d are only weakly dependent on M_D .



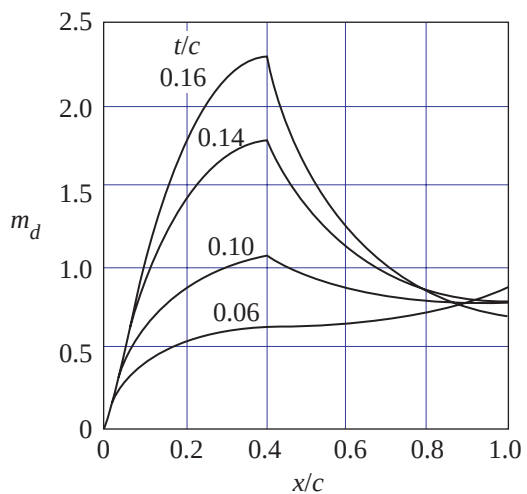
(a) Upper surface



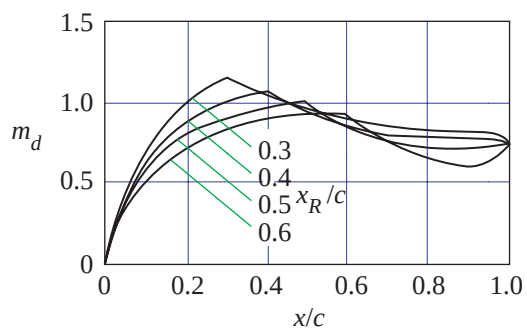
(a) Upper surface



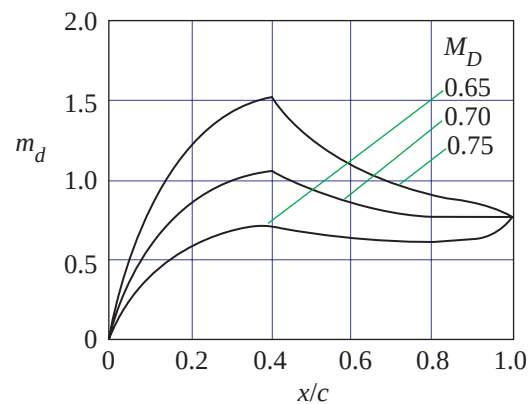
(a) Upper surface



(b) Lower surface



(b) Lower surface



(b) Lower surface

Sketch 5.1 Variation of t/c
($x_R/c = 0.4$, $M_D = 0.70$)

Sketch 5.2 Variation of x_R/c
($M_D = 0.70$, $t/c = 0.10$)

Sketch 5.3 Variation of M_D
($x_R/c = 0.4$, $t/c = 0.10$)

6. DERIVATION AND REFERENCES

6.1 Derivation

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

1. ESDU Aerofoils having a specified form of upper-surface pressure distribution: Details and comments on design.
ESDU International, Item No. 71020, 1971.
2. PATEL, B.
LOCK, R.C. Calculation of excrescence drag magnification using the Nash and Bradshaw formula adapted to compressible flow and a modified BVGK computer program (U).
ESDU International, Memor. No. 63, 1987 (with Amendment A, 1989).
3. ESDU Simplified method for the prediction of aerofoil excrescence drag magnification factor for turbulent boundary layers at subcritical Mach numbers.
ESDU International, Item No. 91028, 1991.

6.2 References

The References list sources of information supplementary to that given in this Item.

4. PANKHURST, R.C.
SQUIRE, H.B. Calculated pressure distributions for the RAE 100 – 104 aerofoil sections.
ARC CP 80, 1952.
5. ABBOTT, I.H.
VON DOENHOFF, A.E. *Theory of Wing Sections*.
Dover Publications, New York, 1959.
6. ESDU Profile drag at the drag-rise condition of aerofoils having a specified form of upper-surface pressure distribution at this condition.
ESDU International, Item No. 67011, 1967.
7. ESDU The compressible two-dimensional turbulent boundary layer, both with and without heat transfer on a smooth flat plate, with application to wedges, cylinders and cones.
ESDU International, Item No. 68020, 1968.
8. ESDU Drag-rise Mach number of aerofoils having a specified form of upper-surface pressure distribution: Charts and comments on design.
ESDU International, Item No. 71019, 1971.
9. ESDU Drag due to grooves in a flat plate with a turbulent boundary layer, at subsonic and supersonic speeds.
ESDU International, Item No. 75028, 1975.
10. ESDU An introduction to aircraft excrescence drag.
ESDU International, Item No. 90029, 1990.

7. EXAMPLE

In an assessment of excrescence drag in the cruise condition of an aircraft with a high aspect ratio wing of modest sweep, the relative penalties due to the narrow lateral grooves formed by the gaps between adjacent skin panels on the wing upper surface is required. The wing mid-semi-span region may be assumed to be subject to quasi-two-dimensional chordwise flow, to which the present Item is applicable. The wing skin joints of interest in that region are at $0.15c$, $0.40c$ and $0.80c$. The boundary layer may be assumed to be wholly turbulent at a Reynolds number $R_c = 10^7$, and the depth of the grooves, h , is defined by $h/c = 5 \times 10^{-4}$. The wing section corresponds to aerofoil 102 – 10 – 40 – 75, whilst the cruise condition corresponds to this aerofoil operating at its drag-rise condition, for which $M_\infty = M_D = 0.75$.

Item No. 75028 (Reference 9) is used to obtain the values of C_{Dfp} , the drag coefficient increment due to a groove in uniform flow. The parameter C_{Dfp} is based on groove width, w (which is assumed to be independent of location) and the local kinetic pressure, q_{1E} , at the edge of the boundary layer at the groove location, x/c , in the absence of the groove.

In order to use Item No. 75028, it is necessary to know R_h , the Reynolds number based on groove depth and local flow conditions at the edge of the boundary layer, in the absence of the groove. If it can be assumed, as in Appendix A of Item No. 75028, that

$$\rho \propto T^{-1} \text{ and } \mu \propto T^{0.8} \quad (7.1)$$

where μ is the dynamic viscosity, then the kinematic viscosity, ν , is given by

$$\nu = \frac{\mu}{\rho} \propto T^{1.8}. \quad (7.2)$$

Therefore, since

$$R_c = \frac{u_\infty c}{\nu_\infty}, \quad (7.3)$$

then

$$R_h = R_c \times \frac{h}{c} \times \frac{u_{1E}}{u_\infty} \times \frac{\nu_\infty}{\nu_{1E}}, \quad (7.4)$$

which, with the use of Equation (7.2), becomes

$$R_h = R_c \times \frac{h}{c} \times \frac{u_{1E}}{u_\infty} \times \left(\frac{T_{1E}}{T_\infty} \right)^{-1.8}. \quad (7.5)$$

The values of u_{1E}/u_∞ and T_{1E}/T_∞ , together with M_{1E}/M_∞ and hence M_{1E} (required later) at the groove locations are obtainable to satisfactory accuracy, for the present purpose, from linear interpolations* within the data in Table 8.20(a). They are given in Table 7.1.

* The program for the method of Item No. 91028 can provide calculations corresponding to interpolations based on the inbuilt curve-fitting procedure.

TABLE 7.1

x_E/c	u_{1E}/u_∞	T_{1E}/T_∞	M_{1E}/M_∞	M_{1E}
0.15	1.31	0.92	1.36	1.02
0.40	1.29	0.92	1.34	1.01
0.80	1.04	0.99	1.05	0.79

Use of Equation (7.5) with $R_c = 10^7$, $h/c = 5 \times 10^{-4}$ and the values of u_{1E}/u_∞ and T_{1E}/T_∞ , given in Table 7.1, gives the values of R_h in Table 7.2.

TABLE 7.2

x_E/c	R_h
0.15	7.61×10^3
0.40	7.49×10^3
0.80	5.29×10^3

The values of C_f , the local skin friction coefficient at the groove location, in the absence of the groove, are also required for use in Item No. 75028. In the absence of calculated values for the aerofoil, the use of Item No. 68020 (Reference 7) for flat plates is an acceptable approximation. The values of R_x , the Reynolds number based on x and local conditions at the edge of the boundary layer, in the absence of the groove, are obtained from Equation (7.5) with x/c substituted for h/c , giving the values shown in Table 7.3, which, with the corresponding values of M_{1E} from Table 7.1, give the values of C_f obtained from Item No. 68020, also shown in Table 7.3.

TABLE 7.3

$x/c (= x_E/c)$	R_x	C_f
0.15	2.3×10^6	0.00287
0.40	6.0×10^6	0.00248
0.80	8.5×10^6	0.00240

Then use of Item No. 75028 for grooves normal to the flow (or swept less than 30°) with the values of M_{1E} from Table 7.1, R_h from Table 7.2 and C_f from Table 7.3, gives the values of C_{Dfp} in Table 7.4.

TABLE 7.4

x_E/c	C_{Dfp}
0.15	0.0099
0.40	0.0083
0.80	0.0072

Use is made of Equation (3.1) in Item No. 90029 (Reference 10), which for an aerofoil means that the excrescence reference area A_{ref} is replaced by w and the wing reference area S_{ref} is replaced by c and the factor $K = 1$ (if the geometry and orientation of the grooves in the present example are taken as identical to those for which data are given in Item No. 75028). Thus,

$$C_D = C_{Dfp} \times \frac{w}{c} \times m_d \quad (7.6)$$

where C_D is the drag coefficient increment due to a groove, based on aerofoil chord, c , and free-stream kinetic pressure, q_∞ .

The drag penalties due to the skin grooves, for a given (fixed) value of w/c , are therefore proportional to the product $C_{Dfp} \times m_d$.

Figure 11, with $M_D = 0.75$, $x_R/c = 0.40$ and $t/c = 0.10$ gives, for the upper surface, the values of m_d shown in Table 7.5, together with the values of $C_D/(w/c)$ from Equation (7.6) obtained using the C_{Dfp} values from Table 7.4.

TABLE 7.5

x_E/c	m_d	$C_D/(w/c) = C_{Dfp} m_d$
0.15	1.86	0.0184
0.40	2.12	0.0176
0.80	1.17	0.0084

From Table 7.5 it can be seen that, for grooves of the same width at the locations considered, those within the roof-top, *i.e.* $x_E/c \leq 0.40$, have the most serious implications in terms of drag penalty, with more than twice the drag of that at $0.8c$.

8. TABLES

TABLE 8.1

AEROFOIL : 101 - 16 - 30 - 60

ALPHAD = 3.95

CLD = 0.97

MD = 0.60

(a) UPPER SURFACE

xE/c	CpE	M1E	M1E/Minf	u1E/uinf	T1E/Tinf	RHO1E/ RHOinf	THETA E/ THETA T	md
.00000	1.0000	.1632	.2720	.2809	1.0663	1.1741	.0000	.0000
.00961	-1.3620	1.0218	1.7030	1.6037	.8868	.7406	.0020	2.0669
.03806	-1.3620	1.0218	1.7030	1.6037	.8868	.7406	.0135	3.0160
.08427	-1.3620	1.0218	1.7030	1.6037	.8868	.7406	.0286	3.5028
.14640	-1.3620	1.0218	1.7030	1.6037	.8868	.7406	.0466	3.8628
.22220	-1.3620	1.0218	1.7030	1.6037	.8868	.7406	.0669	4.1520
.30870	-1.3320	1.0121	1.6868	1.5911	.8897	.7467	.0906	4.2741
.40250	-1.1550	.9558	1.5930	1.5166	.9064	.7822	.1288	3.7887
.50000	-.9410	.8894	1.4823	1.4260	.9256	.8242	.1801	3.1699
.59750	-.7260	.8237	1.3729	1.3338	.9439	.8656	.2457	2.5844
.69130	-.5200	.7612	1.2687	1.2435	.9607	.9045	.3292	2.0724
.77780	-.3300	.7033	1.1722	1.1577	.9755	.9399	.4345	1.6477
.85360	-.1630	.6516	1.0860	1.0796	.9881	.9705	.5629	1.3135*
.91570	-.0260	.6083	1.0139	1.0129	.9981	.9953	.7078	1.0669*
.96190	.0750	.5757	.9595	.9620	1.0054	1.0135	.8495	.9010*
.99040	.1380	.5549	.9248	.9293	1.0098	1.0247	.9590	.8044*
1.00000	.1590	.5479	.9131	.9183	1.0113	1.0285	1.0000	.7733*

(b) LOWER SURFACE

xE/c	CpE	M1E	M1E/Minf	u1E/uinf	T1E/Tinf	RHO1E/ RHOinf	THETA E/ THETA T	md
.00000	1.0000	.1632	.2720	.2809	1.0663	1.1741	.0000	.0000
.00961	1.0460	.1159	.1931	.1997	1.0691	1.1819	.0365	.0009
.03806	.4970	.4270	.7116	.7237	1.0343	1.0879	.0219	.1395
.08427	.1550	.5492	.9154	.9204	1.0110	1.0277	.0599	.4445
.14640	-.0350	.6112	1.0187	1.0174	.9975	.9937	.1092	.7471
.22220	-.1220	.6388	1.0646	1.0599	.9911	.9779	.1730	.9641
.30870	-.1520	.6482	1.0803	1.0743	.9889	.9725	.2495	1.0947
.40250	-.0280	.6090	1.0150	1.0139	.9980	.9950	.3970	.9541
.50000	.0760	.5753	.9589	.9615	1.0054	1.0136	.5649	.8287
.59750	.1670	.5452	.9087	.9140	1.0118	1.0299	.7558	.7179
.69130	.2310	.5235	.8725	.8796	1.0163	1.0412	.9382	.6432
.77780	.2740	.5086	.8477	.8559	1.0193	1.0488	1.0975	.5952
.85360	.2890	.5034	.8390	.8475	1.0203	1.0515	1.1921	.5819*
.91570	.3060	.4974	.8291	.8379	1.0215	1.0545	1.2846	.5645*
.96190	.2010	.5337	.8895	.8958	1.0142	1.0359	1.0564	.7085*
.99040	.1660	.5455	.9092	.9146	1.0118	1.0297	1.0064	.7619*
1.00000	.1590	.5479	.9131	.9183	1.0113	1.0285	1.0000	.7733*

* Values of m_d are of increasingly doubtful validity for excrescence locations aft of $x_E/c = 0.8$ (see Section 5.1 of Item No. 91028).

TABLE 8.2

AEROFOIL : 101 - 06 - 30 - 65

ALPHAD = 3.27

CLD = 0.98

MD = 0.65

(a) UPPER SURFACE

x_E/c	C_{pE}	$M1E$	$M1E/M_{inf}$	$u1E/u_{inf}$	$T1E/T_{inf}$	$RHO1E/RHO_{inf}$	$THETA E/THETA T$	m_d
.00000	1.0000	.1887	.2902	.3012	1.0768	1.2033	.0000	.0000
.00961	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0029	1.5587
.03806	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0191	2.2699
.08427	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0404	2.6360
.14640	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0659	2.9068
.22220	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0945	3.1243
.30870	-1.0380	1.0107	1.5549	1.4755	.9005	.7696	.1282	3.2060
.40250	-.9120	.9654	1.4853	1.4200	.9141	.7989	.1775	2.9411
.50000	-.7500	.9085	1.3976	1.3484	.9309	.8360	.2419	2.5512
.59750	-.5880	.8525	1.3115	1.2762	.9469	.8724	.3200	2.1709
.69130	-.4330	.7994	1.2298	1.2060	.9616	.9068	.4133	1.8276
.77780	-.2890	.7501	1.1541	1.1394	.9748	.9382	.5231	1.5312
.85360	-.1630	.7068	1.0874	1.0798	.9860	.9653	.6458	1.2917*
.91570	-.0600	.6711	1.0324	1.0297	.9949	.9873	.7724	1.1103*
.96190	.0170	.6440	.9908	.9915	1.0014	1.0036	.8879	.9832*
.99040	.0640	.6273	.9651	.9677	1.0054	1.0135	.9698	.9092*
1.00000	.0800	.6216	.9563	.9595	1.0067	1.0168	1.0000	.8847*

(b) LOWER SURFACE

x_E/c	C_{pE}	$M1E$	$M1E/M_{inf}$	$u1E/u_{inf}$	$T1E/T_{inf}$	$RHO1E/RHO_{inf}$	$THETA E/THETA T$	m_d
.00000	1.0000	.1887	.2902	.3012	1.0768	1.2033	.0000	.0000
.00961	.8130	.3142	.4834	.4986	1.0635	1.1664	.0209	.0308
.03806	.5240	.4515	.6946	.7090	1.0420	1.1084	.0614	.1534
.08427	.4170	.4953	.7619	.7747	1.0338	1.0866	.1398	.2566
.14640	.3630	.5165	.7946	.8063	1.0296	1.0756	.2386	.3343
.22220	.3340	.5277	.8119	.8229	1.0273	1.0696	.3547	.3922
.30870	.3110	.5365	.8254	.8359	1.0255	1.0649	.4778	.4428
.40250	.3140	.5354	.8237	.8342	1.0257	1.0655	.6290	.4641
.50000	.3060	.5384	.8284	.8387	1.0251	1.0638	.7629	.4927
.59750	.2950	.5426	.8348	.8448	1.0242	1.0616	.8841	.5223
.69130	.2760	.5498	.8459	.8554	1.0227	1.0576	.9776	.5597
.77780	.2480	.5603	.8620	.8708	1.0204	1.0519	1.0370	.6076
.85360	.2080	.5751	.8848	.8924	1.0172	1.0436	1.0547	.6717*
.91570	.1710	.5887	.9057	.9121	1.0142	1.0359	1.0617	.7332*
.96190	.1390	.6004	.9236	.9290	1.0116	1.0292	1.0592	.7877*
.99040	.0980	.6152	.9464	.9503	1.0082	1.0206	1.0194	.8548*
1.00000	.0800	.6216	.9563	.9595	1.0067	1.0168	1.0000	.8847*

* Values of m_d are of increasingly doubtful validity for excrescence locations aft of $x_E/c = 0.8$ (see Section 5.1 of Item No. 91028).

TABLE 8.3

AEROFOIL : 101 - 10 - 30 - 65

ALPHAD = 2.98

CLD = 0.82

MD = 0.65

(a) UPPER SURFACE

x_E/c	C_{pE}	$M1E$	$M1E/M1inf$	$u1E/u1inf$	$T1E/T1inf$	$RHO1E/RHO1inf$	$THETA E/THETA T$	md
.00000	1.0000	.1887	.2902	.3012	1.0768	1.2033	.0000	.0000
.00961	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0027	1.5775
.03806	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0175	2.2973
.08427	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0369	2.6678
.14640	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0601	2.9419
.22220	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0863	3.1620
.30870	-1.0380	1.0107	1.5549	1.4755	.9005	.7696	.1171	3.2447
.40250	-.9050	.9629	1.4814	1.4170	.9148	.8005	.1630	2.9548
.50000	-.7340	.9029	1.3891	1.3413	.9325	.8396	.2240	2.5360
.59750	-.5640	.8442	1.2988	1.2654	.9492	.8778	.2988	2.1343
.69130	-.4010	.7884	1.2130	1.1913	.9646	.9138	.3898	1.7740
.77780	-.2500	.7368	1.1335	1.1211	.9783	.9466	.4989	1.4665
.85360	-.1180	.6912	1.0634	1.0581	.9899	.9749	.6233	1.2202*
.91570	-.0090	.6532	1.0049	1.0045	.9992	.9981	.7558	1.0330*
.96190	.0710	.6248	.9613	.9641	1.0060	1.0150	.8773	.9052*
.99040	.1210	.6069	.9337	.9384	1.0101	1.0254	.9668	.8294*
1.00000	.1380	.6007	.9242	.9295	1.0115	1.0290	1.0000	.8043*

(b) LOWER SURFACE

x_E/c	C_{pE}	$M1E$	$M1E/M1inf$	$u1E/u1inf$	$T1E/T1inf$	$RHO1E/RHO1inf$	$THETA E/THETA T$	md
.00000	1.0000	.1887	.2902	.3012	1.0768	1.2033	.0000	.0000
.00961	.8710	.2807	.4318	.4462	1.0677	1.1779	.0144	.0190
.03806	.3800	.5099	.7844	.7965	1.0309	1.0790	.0339	.2222
.08427	.1930	.5807	.8933	.9005	1.0160	1.0404	.0845	.4329
.14640	.1030	.6134	.9436	.9477	1.0086	1.0217	.1488	.5931
.22220	.0620	.6280	.9662	.9687	1.0052	1.0131	.2267	.7036
.30870	.0410	.6355	.9777	.9794	1.0034	1.0086	.3133	.7838
.40250	.0970	.6155	.9469	.9508	1.0081	1.0204	.4483	.7491
.50000	.1380	.6007	.9242	.9295	1.0115	1.0290	.5830	.7220
.59750	.1730	.5880	.9046	.9111	1.0144	1.0363	.7184	.6957
.69130	.1940	.5803	.8928	.8999	1.0161	1.0407	.8362	.6831
.77780	.1990	.5785	.8899	.8972	1.0165	1.0417	.9223	.6885
.85360	.1860	.5832	.8973	.9042	1.0154	1.0390	.9650	.7162*
.91570	.1790	.5858	.9012	.9079	1.0148	1.0375	1.0058	.7339*
.96190	.1640	.5913	.9097	.9158	1.0136	1.0344	1.0166	.7612*
.99040	.1450	.5982	.9203	.9258	1.0121	1.0304	1.0048	.7927*
1.00000	.1380	.6007	.9242	.9295	1.0115	1.0290	1.0000	.8043*

* Values of m_d are of increasingly doubtful validity for excrescence locations aft of $x_E/c = 0.8$ (see Section 5.1 of Item No. 91028).

TABLE 8.4

AEROFOIL : 101 - 14 - 30 - 65

ALPHAD = 2.74

CLD = 0.65

MD = 0.65

(a) UPPER SURFACE

x _E /c	C _{pE}	M _{1E}	M _{1E} /M _{inf}	u _{1E} /u _{inf}	T _{1E} /T _{inf}	RHO _{1E} / RHO _{inf}	THETA _E / THETA _T	md
.00000	1.0000	.1887	.2902	.3012	1.0768	1.2033	.0000	.0000
.00961	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0025	1.5925
.03806	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0162	2.3192
.08427	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0341	2.6932
.14640	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0556	2.9699
.22220	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0798	3.1921
.30870	-1.0380	1.0107	1.5549	1.4755	.9005	.7696	.1083	3.2756
.40250	-.8980	.9604	1.4776	1.4139	.9156	.8021	.1517	2.9610
.50000	-.7210	.8984	1.3821	1.3356	.9338	.8426	.2098	2.5227
.59750	-.5440	.8374	1.2883	1.2564	.9511	.8822	.2817	2.1026
.69130	-.3740	.7792	1.1988	1.1789	.9671	.9197	.3708	1.7280
.77780	-.2170	.7254	1.1160	1.1055	.9812	.9537	.4793	1.4115
.85360	-.0800	.6780	1.0431	1.0396	.9932	.9830	.6054	1.1602*
.91570	.0330	.6383	.9821	.9834	1.0028	1.0070	.7421	.9707*
.96190	.1160	.6087	.9364	.9410	1.0097	1.0244	.8698	.8418*
.99040	.1680	.5898	.9074	.9137	1.0140	1.0352	.9655	.7655*
1.00000	.1850	.5836	.8979	.9047	1.0153	1.0388	1.0000	.7413*

(b) LOWER SURFACE

x _E /c	C _{pE}	M _{1E}	M _{1E} /M _{inf}	u _{1E} /u _{inf}	T _{1E} /T _{inf}	RHO _{1E} / RHO _{inf}	THETA _E / THETA _T	md
.00000	1.0000	.1887	.2902	.3012	1.0768	1.2033	.0000	.0000
.00961	.9450	.2320	.3568	.3696	1.0730	1.1925	.0116	.0089
.03806	.3210	.5327	.8196	.8303	1.0263	1.0669	.0200	.2415
.08427	-.0100	.6535	1.0054	1.0050	.9992	.9979	.0509	.6188
.14640	-.1830	.7137	1.0980	1.0893	.9842	.9610	.0911	.9555
.22220	-.2610	.7405	1.1393	1.1263	.9773	.9442	.1420	1.1910
.30870	-.2850	.7488	1.1519	1.1375	.9752	.9390	.2025	1.3298
.40250	-.1500	.7023	1.0805	1.0735	.9871	.9681	.3174	1.1576
.50000	-.0400	.6641	1.0216	1.0199	.9966	.9915	.4457	1.0120
.59750	.0550	.6305	.9700	.9723	1.0046	1.0116	.5886	.8859
.69130	.1210	.6069	.9337	.9384	1.0101	1.0254	.7226	.8024
.77780	.1640	.5913	.9097	.9158	1.0136	1.0344	.8368	.7507
.85360	.1780	.5862	.9018	.9084	1.0148	1.0373	.9047	.7385*
.91570	.2030	.5770	.8877	.8951	1.0168	1.0425	.9870	.7089*
.96190	.1900	.5818	.8950	.9021	1.0157	1.0398	.9880	.7310*
.99040	.1860	.5832	.8973	.9042	1.0154	1.0390	.9965	.7391*
1.00000	.1850	.5836	.8979	.9047	1.0153	1.0388	1.0000	.7413*

* Values of m_d are of increasingly doubtful validity for excrescence locations aft of $x_E/c = 0.8$ (see Section 5.1 of Item No. 91028).

TABLE 8.5

AEROFOIL : 101 - 16 - 30 - 65

ALPHAD = 2.64

CLD = 0.57

MD = 0.65

(a) UPPER SURFACE

x _E /c	C _{pE}	M _{1E}	M _{1E} /M _{inf}	u _{1E} /u _{inf}	T _{1E} /T _{inf}	RHO _{1E} / RHO _{inf}	THETA _E / THETA _T	md
.00000	1.0000	.1887	.2902	.3012	1.0768	1.2033	.0000	.0000
.00961	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0024	1.5993
.03806	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0155	2.3291
.08427	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0328	2.7048
.14640	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0535	2.9826
.22220	-1.0680	1.0216	1.5717	1.4887	.8972	.7625	.0768	3.2058
.30870	-1.0380	1.0107	1.5549	1.4755	.9005	.7696	.1042	3.2896
.40250	-.8950	.9594	1.4760	1.4125	.9159	.8028	.1464	2.9642
.50000	-.7150	.8963	1.3789	1.3329	.9344	.8439	.2028	2.5163
.59750	-.5350	.8343	1.2835	1.2523	.9520	.8842	.2734	2.0884
.69130	-.3620	.7751	1.1925	1.1734	.9682	.9223	.3614	1.7077
.77780	-.2030	.7206	1.1086	1.0989	.9825	.9567	.4689	1.3888
.85360	-.0630	.6721	1.0340	1.0312	.9946	.9867	.5958	1.1338*
.91570	.0520	.6316	.9717	.9738	1.0044	1.0110	.7344	.9430*
.96190	.1370	.6011	.9248	.9300	1.0114	1.0288	.8659	.8128*
.99040	.1890	.5821	.8956	.9026	1.0157	1.0396	.9629	.7377*
1.00000	.2070	.5755	.8854	.8930	1.0171	1.0434	1.0000	.7125*

(b) LOWER SURFACE

x _E /c	C _{pE}	M _{1E}	M _{1E} /M _{inf}	u _{1E} /u _{inf}	T _{1E} /T _{inf}	RHO _{1E} / RHO _{inf}	THETA _E / THETA _T	md
.00000	1.0000	.1887	.2902	.3012	1.0768	1.2033	.0000	.0000
.00961	.9740	.2102	.3233	.3352	1.0750	1.1982	.0109	.0060
.03806	.3200	.5331	.8202	.8308	1.0262	1.0667	.0157	.2335
.08427	-.0930	.6826	1.0501	1.0459	.9921	.9803	.0398	.6978
.14640	-.3320	.7649	1.1767	1.1595	.9709	.9288	.0709	1.1764
.22220	-.4440	.8031	1.2356	1.2110	.9606	.9043	.1114	1.5288
.30870	-.4740	.8134	1.2514	1.2247	.9578	.8977	.1615	1.7211
.40250	-.2850	.7488	1.1519	1.1375	.9752	.9390	.2666	1.4219
.50000	-.1310	.6957	1.0704	1.0644	.9888	.9722	.3902	1.1803
.59750	-.0020	.6507	1.0011	1.0010	.9998	.9996	.5333	.9861
.69130	.0890	.6184	.9514	.9549	1.0075	1.0187	.6738	.8579
.77780	.1510	.5960	.9169	.9227	1.0126	1.0317	.7995	.7753
.85360	.1790	.5858	.9012	.9079	1.0148	1.0375	.8808	.7417*
.91570	.2170	.5718	.8797	.8876	1.0179	1.0454	.9801	.6929*
.96190	.2000	.5781	.8894	.8967	1.0166	1.0419	.9685	.7198*
.99040	.2040	.5766	.8871	.8946	1.0169	1.0427	.9896	.7161*
1.00000	.2070	.5755	.8854	.8930	1.0171	1.0434	1.0000	.7125*

* Values of m_d are of increasingly doubtful validity for excrescence locations aft of $x_E/c = 0.8$ (see Section 5.1 of Item No. 91028).