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Transonic data memorandum Aerofoils having a specified form of upper-surface pressure distribution: Details and comments on design

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TRANSONIC DATA MEMORANDUM 71020 AEROFOILS HAVING A SPECIFIED FORM OF UPPER-SURFACE PRESSURE DISTRIBUTION: DETAILS AND COMMENTS ON DESIGN

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TRANSONIC DATA MEMORANDUM 71020

AEROFOILS HAVING A SPECIFIED FORM OF UPPER-SURFACE PRESSURE DISTRIBUTION: DETAILS AND COMMENTS ON DESIGN

INTRODUCTION

One object of this Memorandum is to present details of a series of aerofoils that have been used for parametric studies of drag-rise Mach number¹, of profile drag at the drag-rise Mach number² and for a limited study of profile drag at off-design conditions³. This series, constituting a cross-section of infinite families of aerofoils, was designed by an inversion of the direct Lock, Wilby, Powell method⁴ to give a specified and particular form of upper-surface pressure distribution at the drag-rise Mach number in a manner that is especially appropriate for application to swept wings at high-subsonic and low-supersonic speeds. The particular form of upper-surface pressure distribution, which with a form of thickness distribution, design Reynolds number and design transition positions on the upper and lower surfaces defines the infinite families, is a level plateau ("flat roof-top") – corresponding to a local Mach number of 1.02 ($p_r = 0.515H$, see Section 2.1) and extending from the immediate vicinity of the leading edge to a chordwise station designated x_R (varied from $0.3c$ to $0.6c$) – followed by a linear rise to the trailing edge. Corresponding thickness distributions have been chosen from the RAE 101 to 104 series (t/c ranging from 6 per cent to 16 per cent), the design Reynolds number has been taken as 10^7 based on chord length and the design transition positions have been taken at $0.05c$ on both upper and lower surfaces.

A further object is to discuss the influence that the principal design variables have on the geometrical shape of the cambered aerofoils that have so far been derived to this specification. Furthermore, the opportunity is taken to draw attention to certain limitations having aerodynamic significance that result from the use of a restricted range and particular type of thickness distribution. Although these limitations do not detract from the usefulness of the aerofoils as a basis for parametric^{1,2,3} and initial design studies they are certainly such that modifications would in many cases be required at the detail design stage. At this stage the necessary modifications could be introduced through defining the aerofoil specification in greater detail and then employing the same inverse method of derivation to determine the new geometry.

This Memorandum supersedes T.D. Memor. 67010 (issued in 1967) in which a similar family of aerofoils was presented. This earlier family was designed using inviscid flow theory incorporating a description of compressibility effects thought at the time to be the best available. Now, an inversion of the method of Reference 4 has led to the derivation of the present family which, through the accounting for viscous effects and the use of more-recently derived and better substantiated⁴ compressibility factors, has greater realism than the former – although the limitations referred to above must still be borne in mind.

The principal difference between the aerofoils presented here and those presented in T.D. Memor. 67010 is that for a specified Mach number and thickness distribution the present series generate slightly less lift at the design condition, require less camber and incidence and have smaller values of trailing-edge pressure coefficient.

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1. NOTATION

C_{LD}	lift coefficient when free-stream Mach number is M_D
C_p	pressure coefficient
c	aerofoil chord
H	stagnation pressure in undisturbed stream
M_D	drag-rise Mach number as defined in Reference 5 (see also Section 2.1)
p	local static pressure
R_{ed}	design Reynolds number based on chord length and M_D
t	aerofoil maximum thickness
x	chordwise distance measured from leading edge
z	aerofoil ordinate measured normal to chord line
z_s	aerofoil camber-line ordinate measured normal to chord line
z'_s	aerofoil camber-line ordinate measured normal to datum parallel to free stream and passing through trailing edge
α	angle of incidence

Suffixes

l	denotes lower surface
R	identifies chordwise station defining end of roof-top
r	denotes local value on constant-pressure plateau
t	due to thickness distribution
te	pertaining to trailing edge
trd	identifies design transition position
u	denotes upper surface

2. DETAILS OF THE AEROFOILS

2.1 General Specification

The aerofoils in this Memorandum are, for reasons given in Reference 1, specified by the value of drag-rise Mach number (M_D), the form of the upper-surface pressure distribution at this Mach number, the form of the thickness distribution, the thickness/chord ratio and values of Reynolds number and upper and lower surface transition position at the design condition; the associated lift coefficient (C_{LD}) is implicit in this specification.

It follows from the empirical criterion[†] for the onset of rapid drag-rise developed in Reference 5 that if an aerofoil is designed at a certain free-stream Mach number to have a constant pressure p equal to $0.515H$ over the forward part of the upper surface and embracing the crest, then that Mach number is, in fact, the drag-rise Mach number. It also follows from this design feature that the supercritical drag-creep⁵ (prior to the rapid drag-rise and encountered when an aerofoil pressure distribution and shape combine in a particular way) is automatically avoided at the design condition.

The form of the upper-surface pressure distribution is completed by defining the chordwise extent of the roof-top (the variable rearward extent being designated x_R), and the pressure distribution in the vicinity of x_R ; the subsequent pressure recovery and the value of trailing-edge pressure being implicit in the design process and compatible with conditions in a real (*i.e.* viscous) flow. Precise details of these features will be found in Section 6.2 of Reference 1.

The specification is completed by the choice of thickness form and here the RAE 101 to 104 series is used⁶. Principal features of this series of thickness forms are given in Table 6.1. For simplicity in demonstrating the form of aerofoil implicit in this type of specification, and in presenting the parametric studies of References 1, 2 and 3, the thickness distribution associated with a particular upper-surface pressure distribution is that having the same roof-top limit[‡]. The choice of thickness form and of the relative positions of the roof-top limit need not, of course, be restricted in these ways.

2.2 Aerofoil Numbering System

Any aerofoil corresponding to the general specification of Section 2.1 is primarily defined by the thickness form, and the values of t/c , x_R/c and M_D ; the values ascribed at the design condition to Reynolds number based on chord length and M_D , and to the location of transition position on upper and lower surfaces having lesser influence. Any aerofoil may be identified by these seven quantities using, for example, a designation of the following type.

$$102 - 12 - 40 - 75 \quad (7.0, 05, 05)$$

where the seven numbers taken in sequence have the following significance:

102 denotes RAE 102 series

$$12 = 100 t/c$$

$$40 = 100 x_R/c$$

$$75 = 100 M_D$$

$$7.0 = \text{index of } R_{ed} \text{ expressed to a base of } 10$$

$$05 = 100 (x/c)_{trd} - \text{upper surface}$$

$$05 = 100 (x/c)_{trd} - \text{lower surface.}$$

[†] Shock-wave drag will be encountered when the pressure at the crest of an aerofoil, calculated on the basis of equivalent shock-free flow, falls below $0.515H$ (*i.e.* local Mach number at the crest increases above 1.02).

[‡] Strictly, it is only according to linear theory for incompressible flow that the RAE series has a flat roof-top form from the leading edge to the station x_R at zero incidence. According to exact theory the station x_R corresponds to the point of minimum pressure in incompressible flow, but the effect of increasing Mach number can be such as to relocate the minimum pressure ahead of x_R (see Figure 4).

This numbering system is used throughout this Memorandum, except that for convenience the bracketed numbers are omitted because they are not varied from the values above for the members of the series presented here.

2.3 Particular Examples

The principal features of the particular members of the series chosen for presentation here are contained in Table 6.2. The values of t/c , x_R/c and M_D are implicit in the designation of each aerofoil. Values of C_{LD} and α are tabulated; the former have been extracted from Figures 2 to 5 of Reference 1 and the latter are defined by

$$\alpha = \sin^{-1}(z'_s/c)_{x=0}$$

calculated for viscous compressible flow.

Upper- and lower-surface ordinates and chordwise pressure distributions (expressed in coefficient form) are given in Tables 6.3 to 6.6 for $x_R/c = 0.3(0.1)0.6$; the values are given for the points $x/c = (1 + \cos\phi)/2$ where $\phi = v\pi/16$ with $v = 1, 2, \dots, 15$.

The aerofoil camber lines are not given in tabular form because there is no case for using any particular one in combination with a thickness distribution other than that used in its derivation.

The procedure used to derive the ordinates of the camber line and the value of the associated incidence, which, at the known value of M_D , would generate the required upper-surface pressure distribution when combined with the chosen thickness form and thickness/chord ratio at the design Reynolds number and for the design transition positions is outlined in Section 6.3.1 of Reference 1. The relevant equations are given in Appendix A of Reference 1.

3. THE INFLUENCE OF SYSTEMATIC CHANGES IN THE PRIMARY DESIGN VARIABLES

The object of this Section, in conjunction with Figures 1 to 6, is to use small groups of the aerofoils so far derived to illustrate how the shapes of the aerofoils and their pressure distributions are influenced by systematic changes in any one of three primary design variables, M_D , t/c and x_R/c , as the other variables are held constant. This procedure also illustrates the “exchange rate” between each of these variables and the dependent variable[†] C_{LD} , because the value of C_{LD} is derived directly from the specification once the aerofoil geometry has been established. Furthermore, one small group of aerofoils for which C_{LD} is approximately constant has been selected (Figure 2) and this allows the “exchange rate” between M_D and t/c to be illustrated for a particular value of x_R/c .

Because the upper-surface pressure distribution and the thickness distribution are both specified, the primary influence of the design variables is on the shape of the camber line. The basic principle that can be invoked to allow a qualitative assessment to be made of the effect of the variables is that the resultant upper-surface pressure distribution is produced essentially by adding to the local pressure due to thickness alone (at zero incidence) the (negative) increments in pressure due to camber and incidence in such a way that the specified upper-surface pressure distribution is achieved. These pressure increments approximate to half the local loading (not exactly so because of non-linear effects, particularly near the leading edge) and hence are directly responsible for determining not only the design incidence and the shape of the camber line but the lift coefficient C_{LD} as well. With the specification used here (and in References 1, 2 and 3), the upper-surface pressure distribution depends mainly on the two parameters M_D and x_R/c – the effect of t/c ,

[†] At the detailed design stage any two of the four variables (including C_{LD}) can be specified and the other two determined from the charts presented in Figures 2 to 5 of Reference 1. The aerofoil can then be derived by inverting the method of Reference 4 (see Appendix A of Reference 1).

which influences the pressure recovery to the trailing edge[†], being a secondary one – and it is clear from Figure 1 (see also Section 3.1) that the value of $(-C_{pr})$ falls dramatically with increasing M_D . Conversely, the contribution to the upper-surface distribution arising from the thickness distribution when expressed in the form $(-C_{pt})$ is an increasing function of t/c and – because of the effect of compressibility – of M_D also. It may thus be appreciated that variations in any of the primary design parameters can affect significantly the shape of the area between the specified upper-surface pressure distribution and the curve of $(-C_{pt})$ [‡]. This, with the ensuing changes in the shape of the camber line, is discussed in the following sub-sections.

3.1 Drag-Rise Mach Number

The strongest influence of increasing M_D stems from the fall in the value of $(-C_{pr})$ on the constant-pressure plateau on the upper surface (see Figures 1 and 2). This occurs because the velocity on the plateau is specifically related to a local Mach number of 1.02 ($p_r = 0.515H$) and cannot increase with increasing free-stream velocity. The pressure coefficient $(-C_{pr})$ is, in fact, fixed according to the equation

$$C_{pr} = \frac{0.515(1 + 0.2M_D^2)^{3.5} - 1}{0.7M_D^2}, \quad (3.1)$$

derived from the exact isentropic relationship for the condition $p_r = 0.515H$ and taking the ratio of the specific heats of air as 1.4.

The influence of M_D when x_R/c and t/c are held constant (at 0.40 and 0.10, respectively) is illustrated in Figure 1 using aerofoils

$$\begin{aligned} 102 - 10 - 40 - 65, \\ 102 - 10 - 40 - 70, \\ 102 - 10 - 40 - 75. \end{aligned}$$

Examination of the pressure distributions shows how the region between the curve for the upper surface and that due to thickness alone decreases in size and alters in shape as M_D is increased, due both to the decrease in $(-C_{pr})$ and to the increase in $(-C_{pt})$ caused by the normal compressibility effect. Thus at a low value (0.65) of M_D (Figure 1a) the loading can be thought of as compounded of a component of “flat-plate” type (associated with incidence in the absence of camber) and a component that is almost constant for $0 \leq x \leq x_R$ and then decreases linearly to the trailing edge. The latter component leads to a camber line that closely resembles the standard NACA camber line having the appropriate value of a (0.4 in this case). As the design Mach number, M_D , is increased the loading tends to be concentrated towards the leading edge and, to a lesser extent, over the rear of the aerofoil until at the upper limiting value^{††} of M_D (in this case $M_D = 0.780$) there is zero loading near the station x_R/c . The consequent change in the camber line is one that leads to a marked reduction in camber ordinate over the forward portion of the aerofoil, accompanied by a less marked reduction over the rear of the aerofoil. This progression leads, at $M_D = 0.75$ (Figure 1c), to a camber line of “S” type having negative camber over the forward portion of the aerofoil. There is at the same time a rapid decrease in the available lift coefficient C_{LD} , from 0.92 (at $M_D = 0.65$) to 0.28 (at $M_D = 0.75$).

[†] In the detail design stage the form of the upper-surface pressure recovery and the trailing-edge pressure may themselves be treated as design variables, through, for example, local modification to the thickness form.

[‡] For convenience, because they are presented only for illustrative purposes, the curves of C_{pt} in Figures 1 to 4 and 6 have been calculated for inviscid flow and faired into the values of C_{pte} for the derived aerofoils. The effect of Reynolds number and transition position on the pressure distribution due to thickness is small except in the vicinity of the trailing edge. In practice, of course, the pressure distribution due to thickness will not have the same value of C_{pte} as the derived aerofoil even when calculated for the same Reynolds number and transition positions.

^{††} The upper limiting value of M_D is that at which, for any given thickness distribution and t/c ratio, the velocities on the lower surface become sufficiently supersonic to provoke drag-rise.