



Aerofoil incidence for zero lift in subsonic two-dimensional flow

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AEROFOIL INCIDENCE FOR ZERO LIFT IN SUBSONIC TWO-DIMENSIONAL FLOW

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AEROFOIL INCIDENCE FOR ZERO LIFT IN SUBSONIC TWO-DIMENSIONAL FLOW

1. NOTATION AND UNITS

		SI	British
$(a_1)_T$	inviscid lift-curve slope of aerofoil	degree ⁻¹	degree ⁻¹
$(a_1)_v$	viscous lift-curve slope of aerofoil	degree ⁻¹	degree ⁻¹
c	aerofoil chord	m	ft
C_L	aerofoil lift coefficient		
k_M	compressibility correction factor		
k_{M0}	value of k_M when $z_{0.92} = z_{0.5}$ (i.e. $\log_{10} (z_{0.92}/z_{0.5}) = 0$)		
k_v	viscous correction factor, $[(\alpha_0)_v/(\alpha_0)_T]_{M=0}$		
M	free-stream Mach number		
P_j	coefficients in Equations (5.2), (5.3) and (5.4)		
R	Reynolds number based on free-stream conditions and c		
t	maximum thickness of aerofoil section	m	ft
x	chordwise distance aft from aerofoil leading edge, see Equation (5.1)	m	ft
x_j	value of x at chordwise station j , see Equation (5.5)	m	ft
z_c	aerofoil camber-line ordinate at station x	m	ft
z_{cj}	value of z_c at chordwise station j , see Equation (5.2)	m	ft
$z_{0.5}$	aerofoil camber-line ordinate at $x = 0.5c$, see Sketch 5.1	m	ft
$z_{0.92}$	aerofoil camber-line ordinate at $x = 0.92c$, see Sketch 5.1	m	ft
α	aerofoil incidence	degree	degree
$(\alpha_0)_T$	inviscid zero-lift incidence of aerofoil, Equation (5.2)	degree	degree
$(\alpha_0)_v$	viscous zero-lift incidence of aerofoil, Equation (5.6)	degree	degree
ϕ_j	angular parameter at chordwise station, j , Equation (5.5)	degree	degree

2. INTRODUCTION

This Item provides a means of predicting the viscous zero-lift incidence of aerofoil sections. The method of Pankhurst (Derivation 1) is first used to obtain the value of the inviscid zero-lift incidence. Viscous and compressibility correction factors are then applied to obtain the viscous zero-lift incidence.

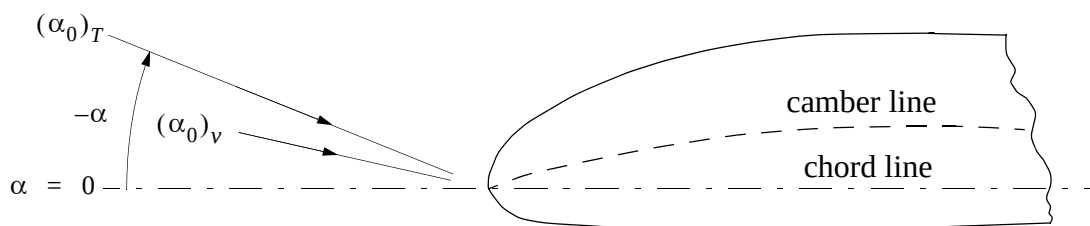
The VGK program* (Derivations 3 and 4) was used with a variety of aerofoils to obtain the viscous zero-lift incidence data used as the basis for this Item. With the program run in inviscid mode the values of $(\alpha_0)_T$ obtained were found to agree closely with those predicted by the method of Pankhurst, and to be virtually independent of aerofoil thickness distribution and Mach number.

The method may be used for aerofoils with simple or complex camber-line shapes provided that the flow remains subcritical, and the boundary layers are fully attached. A general discussion of the effects of viscosity and Mach number is given in Sections 3 and 4, respectively. The method of Pankhurst is given in Section 5.1, and Section 5.2 introduces the viscous and compressibility correction factors. Section 6 discusses accuracy and applicability. The Derivation and References are given in Section 7. Worked examples showing the application of the method are given in Section 8.

3. VISCOUS EFFECTS

As in inviscid flow, it is primarily the camber-line shape that determines the aerofoil zero-lift incidence; the aerofoil thickness distribution and hence trailing-edge angle have no significant effect.

On a positively cambered aerofoil the boundary layer on the upper surface is thicker than on the lower surface, especially over the rear part of the aerofoil. This effectively decreases the rear camber and the lift associated with camber. The *magnitude* of the zero-lift incidence, which is related to camber, is also reduced, so that $|(\alpha_0)_v| < |(\alpha_0)_T|$, see Sketch 3.1. Because boundary-layer thickness increases as Reynolds number decreases, the difference $|(\alpha_0)_T| - |(\alpha_0)_v|$ increases as Reynolds number decreases. This is illustrated in Sketch 3.2 for the NACA 65₁-412 aerofoil. Note that by their nature positively cambered aerofoils have negative zero-lift incidences.

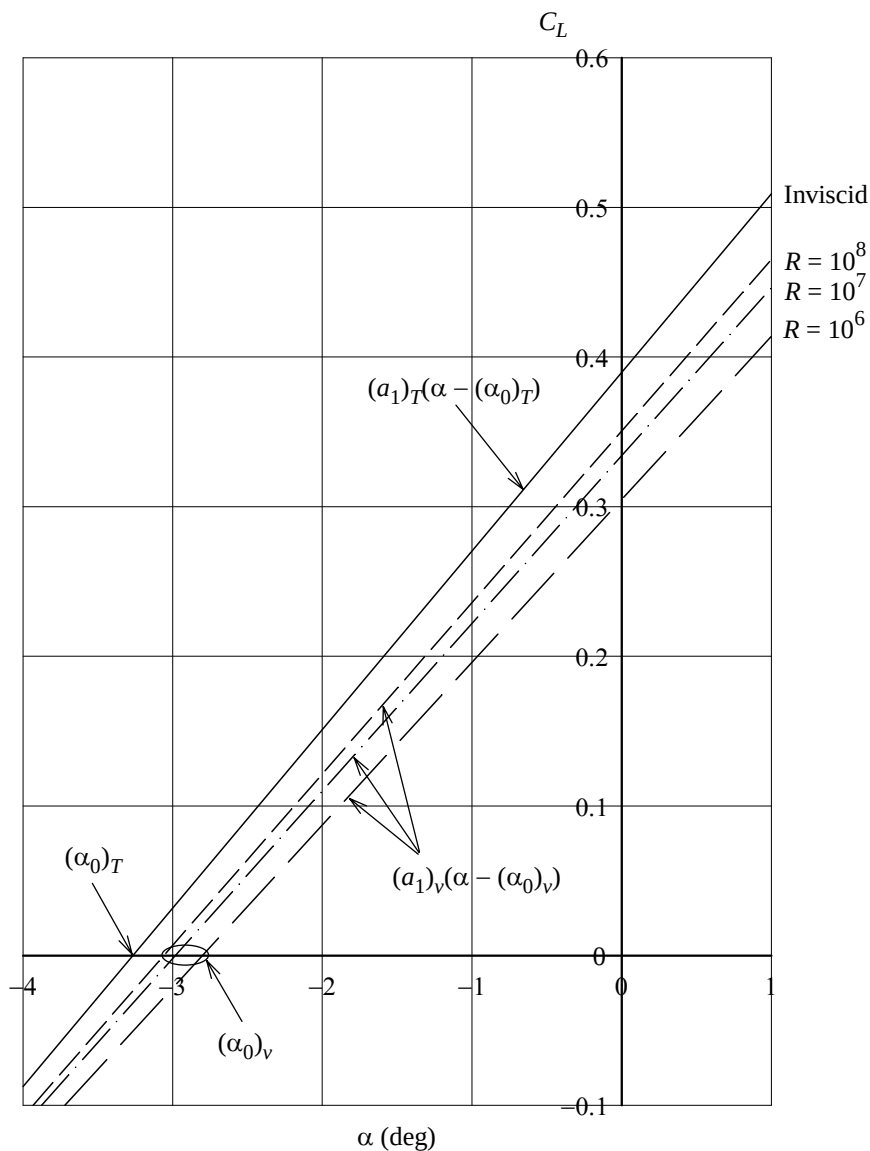


Sketch 3.1 Viscous and inviscid free-stream flow directions at zero lift

* The VGK (Viscous Garabedian and Korn) method is a computational method for determining two-dimensional attached flow past an aerofoil immersed in a subsonic free-stream, with allowance for viscous effects. The method couples finite difference solutions of inviscid flow about the aerofoil with solutions for the displacement effects of the boundary layers. External to the boundary layer the inviscid flow element is governed by the full potential equation. The viscous flow element representing the boundary layers is governed by integral equations. An iterative approach allows boundary-layer displacement and momentum-thickness distributions to be calculated, with allowance for the effects of the pressure field obtained from the inviscid flow element. The method of coupling the viscous and inviscid flow elements permits converged solutions to be obtained for flows with attached boundary layers.

The boundary-layer thickness at the aerofoil trailing edge is also affected by the extent and magnitude of the adverse pressure gradient near the trailing edge. Aerofoils for which there is a large amount of rear camber, for example those with double-hump camber lines, (Reference 6, see also Table 6.1), have a larger adverse pressure gradient near the trailing edge. The influence of viscosity is therefore greater, with the result that the magnitude of the zero-lift incidence decreases as rear camber increases.

Parametric studies with the VGK program showed that the zero-lift incidence was virtually independent of boundary-layer transition position. Thus, in incompressible flow, the effect of viscosity on zero-lift incidence is essentially dependent only on Reynolds number and the amount of aerofoil rear camber.



Sketch 3.2 Low incidence characteristics of NACA 651-412 aerofoil

4. EFFECT OF COMPRESSIBILITY

The effect of increased Mach number in subcritical flow is to cause a greater adverse pressure gradient near the aerofoil trailing edge. This thickens the upper-surface boundary layer and so reduces the effective camber of the aerofoil. The overall effect is that the magnitude of $(\alpha_0)_v$ decreases as Mach number rises, with a dependence on Reynolds number and aerofoil rear camber.

For a combination of low Reynolds number and a large amount of aerofoil rear camber the upper-surface boundary-layer thickness at the trailing edge will be large. The effect of increased Mach number is further thickening of the boundary layer, and if the thickness were to become too great separation would occur and the method would not apply.

This Item deals only with conditions where the flow over the aerofoil is wholly subsonic. Note that at the zero-lift incidence the onset of supercritical flow can occur at a free-stream Mach number other than the critical Mach number associated with the design incidence. For the majority of aerofoils used in the investigation the onset of supercritical flow at zero lift occurs at about $M = 0.6$, which can be taken as an upper limit on applicability for the method.

5. METHOD

5.1 Inviscid Method (Pankhurst)

The method of Pankhurst (Derivation 1) is based on the thin-aerofoil value of $(\alpha_0)_T$ derived by Glauert (Derivation 2) and expressed in degrees as,

$$(\alpha_0)_T = -\frac{180}{\pi^2} \int_0^1 \frac{z_c/c}{(x/c)^{1/2} [1 - (x/c)]^{3/2}} d\left(\frac{x}{c}\right). \quad (5.1)$$

Pankhurst evaluated Equation (5.1) by an approximate means using the trapezium rule with 14 stations and assuming that the tangent at the trailing edge of the camber line coincides with the straight line joining the last two points*. For the definition of chord line used in this Item, where the camber ordinates at the leading and trailing edge are zero, this procedure leads to:

$$(\alpha_0)_T = \sum_{j=1}^{12} P_j(z_{cj}/c), \quad (5.2)$$

$$\text{where} \quad P_j = -\frac{(\phi_{j+1} - \phi_{j-1})}{\pi[1 - (x_j/c)]} \quad \text{for } j = 1 \text{ to } 11, \quad (5.3)$$

$$\text{and} \quad P_{12} = -\frac{(2\phi_{13} - \phi_{12} - \phi_{11})}{\pi[1 - (x_{12}/c)]}, \quad (5.4)$$

$$\text{with} \quad \phi_j = \sin^{-1}(x_j/c)^{1/2} \text{ degrees.} \quad (5.5)$$

* Strictly speaking the method should therefore not be applied when this is not a reasonable approximation, for example when the slope (dz_c/dx) tends to infinity at the trailing edge.