

MAXIMUM LIFT COEFFICIENTS FOR AEROFOIL SECTIONS AT MACH NUMBERS FROM 0.60 TO 0.75

1. NOTATION AND UNITS

		SI	British
C_{Lmax}	maximum lift coefficient of symmetrical aerofoil section		
ΔC_{Lmax}	increment in maximum lift coefficient due to camber		
c	chord of aerofoil section	m	ft
M	free-stream Mach number		
y_5	thickness of aerofoil section at 5 per cent of chord line aft of leading edge	m	ft
τ	trailing-edge angle of aerofoil section	deg	deg

2. NOTES

Work is in progress at ESDU to update the maximum lift estimation for aerofoil sections. This Item will eventually be superseded as part of that work, but in the meantime it may be used as a guide to “conventional” aerofoils with simple camber lines. It should not be applied to aerofoils with reflex camber lines, particularly those with large amounts of rear camber (*e.g.* supercritical aerofoils).

Item No. 84026 gives the maximum lift coefficients that can be sustained by aerofoil sections at Mach numbers below about 0.4. The present Item gives corresponding data at Mach numbers within the range 0.60 to 0.75.

Figure 1 gives the maximum lift coefficients for symmetrical aerofoil sections in terms of the thickness of the aerofoil at 5 per cent chord and its trailing-edge angle. The increment in maximum lift coefficient due to camber of the aerofoil section may be estimated from the relation

$$\Delta C_{Lmax} \approx \frac{\text{percentage camber}}{9}, \quad (2.1)$$

where the camber is the ratio of the maximum ordinate of the aerofoil section mean line to the chord of the aerofoil.

For many aerofoil sections an increase in incidence well above that for the stall may cause a further increase in lift, particularly at high Mach numbers. Thus the maximum lift coefficient is defined here as the value of the lift coefficient when the slope of the lift curve is first zero.

Often it is the occurrence of buffeting or adverse stability changes, rather than the stall, that limits the lift coefficient usable in flight. These effects may occur at lift coefficients of the order of 90 per cent of the maximum.

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The curves have been obtained by correlating data from wind-tunnel tests made at Reynolds numbers between 0.5 and 4×10^6 , and the scatter of the data is within ± 0.1 of the appropriate curves. Scale effects may be appreciable but it is thought that they are unlikely to affect the qualitative trends of the curves presented here.

Data from tests on circular-arc and double-wedge aerofoils are in acceptable agreement with the curves and thus the Item should apply to aerofoil sections with sharp as well as to those with rounded leading-edges.

2.1 Some Comments on the Mechanism of the Stall

The stalling of an aerofoil section is associated with separation of the flow over a considerable fraction of the upper surface, and the value of C_{Lmax} depends on the effect of separation on the pressure distribution on the upper surface and the associated change in pressure distribution on the lower surface. The position of the separation point depends on the pressure distribution, which in turn depends on the section shape and incidence as well as on the free-stream Mach number. The section shape downstream of the separation point may in some cases play an important part because it influences the pressure recovery downstream of separation.

At low Mach numbers the most severe adverse pressure gradient occurs near the leading edge, and C_{Lmax} is often reached when separation occurs there; the nose shape is then usually the most important parameter. As the Mach number is raised, this type of flow persists at C_{Lmax} , although the flow pattern may be complicated by the presence of shock waves. When, however, the Mach number reaches a sufficiently high value, the flow at C_{Lmax} may remain attached to the nose and will then expand round it to produce a high local Mach number downstream. The boundary layer then separates at the shock wave which terminates the resultant region of supersonic flow. Under these circumstances the value of C_{Lmax} depends on the nose shape, since this has a large effect on the pressures in the region of supersonic flow, and also on the shape of the rear of the section, since this affects the pressure recovery after separation and the interaction between the flow on the two surfaces. Both these factors also affect the position of the shock wave. The flow pattern close to C_{Lmax} is thus a very complicated one and cannot be accurately described in terms of a small number of parameters. For this reason the accuracy of the curves given in this Item is not high.

3. DERIVATION

Miscellaneous test data.

4. EXAMPLE

Find the maximum lift coefficient at a Mach number of 0.7 of an aerofoil section for which the camber is 2.5 per cent, $\tau = 7.5^\circ$ and $y_5/c = 0.033$.

From Figure 1b for the symmetrical section

$$C_{Lmax} = 0.83.$$

From Equation (2.1) ΔC_{Lmax} due to camber $\approx 2.5/9 = 0.28$.

Thus the maximum lift coefficient of the aerofoil section

$$= 0.83 + 0.28 = 1.11 \approx 1.1.$$