Fluids – Lecture 19 Notes

1. Airfoils – Overview

Reading: Anderson 4.1–4.3

Airfoils – Overview

3-D wing context

The cross-sectional shape of a wing or other streamlined surface is called an *airfoil*. The importance of this shape arises when we attempt to model or approximate the flow about the 3-D surface as a collection of 2-D flows in the cross-sectional planes.

3–D Wing

2–D Airfoil section flows
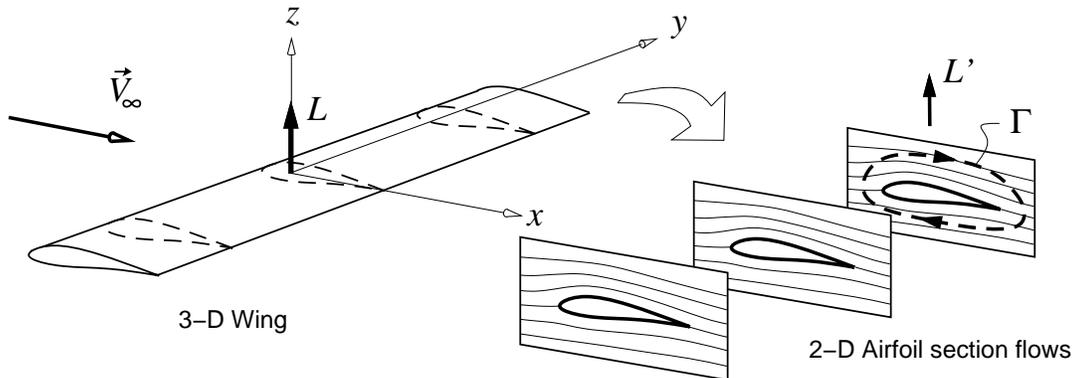
In each such 2-D plane, the airfoil is the aerodynamic body shape of interest. 2-D section properties become functions of the spanwise coordinate y. Examples are $L'(y)$, $\Gamma(y)$, etc. Quantities of interest for the whole wing cn then be obtained by integrating over all the sectional flows. For example,

$$L = \int_{-b/2}^{b/2} L'(y)\, dy$$

where b is the wing span. The airfoil shape is therefore an important item of interest, since it is key in defining the individual section flows.

It must be stressed that the 2-D section flows are <u>not</u> completely independent, but rather they influence each other's effective angle of attack, or the apparent $\vec{V}_\infty$ direction in each 2-D plane. Fortunately this complication does not prevent us from treating each 2-D plane as though it was truly independent, since the angle of attack corrections can be added separately later.

Nomenclature

The figure below shows the key terms used when dealing with airfoil geometry. The *Mean Camber Line* is defined to lie halfway between the upper and lower surfaces.

Maximum Thickness

Leading Edge

Maximum Camber

Mean Camber Line

Chord Line

Trailing Edge

Chord c

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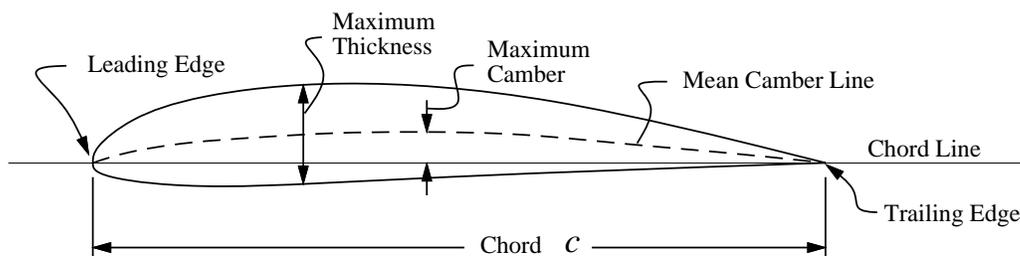
Aerodynamic Characterization

The freestream velocity vector $\vec{V}_\infty$ is defined by its magnitude $V_\infty = |\vec{V}_\infty|$, and the angle of attack α it makes with the airfoil's chord line. The overall aerodynamic loads on the airfoil are the resultant force/span vector $\vec{R}'$ and the moment/span M', by convention taken about the quarter-chord location. The resultant force is resolved into a lift force L' and drag force D' perpendicular and parallel to $\vec{V}_\infty$.
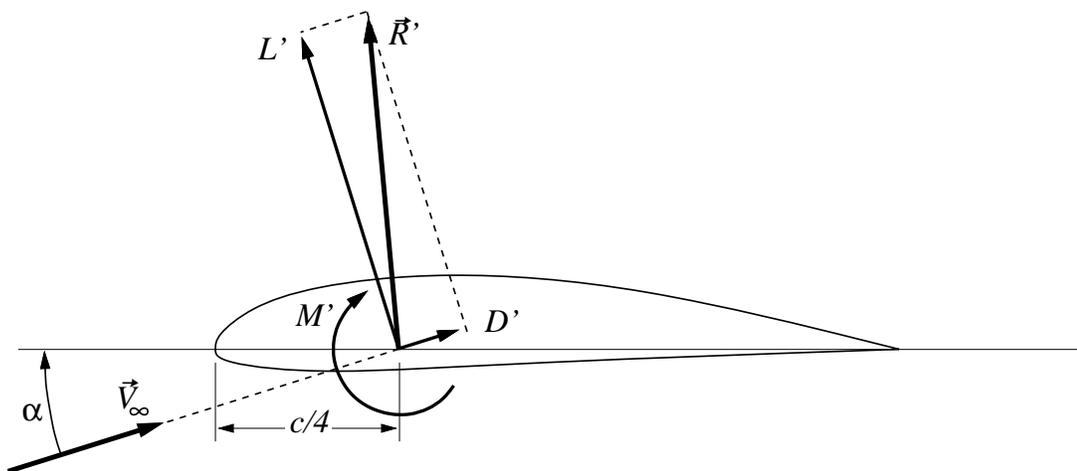
The forces and moment are more conveniently nondimensionalized using the freestream dynamic pressure $q_\infty \equiv \frac{1}{2}\rho_\infty V_\infty^2$ and the chord c, giving the lift, drag, and moment coefficients.

$$c_\ell \equiv \frac{L'}{q_\infty\, c} \quad , \qquad c_d \equiv \frac{D'}{q_\infty\, c} \quad , \qquad c_m \equiv \frac{M'}{q_\infty\, c^2}$$

Dimensional analysis reveals that these will depend only on the angle of attack α, the Reynolds number $Re \equiv \rho_\infty V_\infty c/\mu_\infty$, the Mach number $M_\infty \equiv V_\infty/a_\infty$, and on the airfoil shape.

$$c_\ell\, ,\, c_d\, ,\, c_m \;=\; f\,(\,\alpha\, ,\, Re\, ,\, M_\infty\, ,\, \text{airfoil shape}\,)$$

For low speed flows, M_∞ has virtually no effect. And for a given airfoil shape, we therefore have

$$c_\ell\, ,\, c_d\, ,\, c_m \;=\; f\,(\,\alpha\, ,\, Re\,) \qquad\qquad\qquad \text{(low speed flow, given airfoil)}$$

Typical $c_\ell(\alpha)$ and $c_m(\alpha$ curves for any given Re have a number of important features, as shown in the figure. For moderate angles of attack, the $c_\ell(\alpha)$ curve is nearly linear, and very closely matches the one predicted by potential-flow theory (e.g. a panel method). At some larger angle of attack, c_ℓ curve reaches a maximum value of $c_{\ell_{\max}}$ and then decreases. For α's beyond $c_{\ell_{\max}}$ the airfoil is said to be *stalled*, and exhibits varying amounts of separated flow. An analogous situation occurs for large negative α's.

Within the linear region, the $c_\ell(\alpha)$ curve can be closely approximated with a linear fit.

$$c_\ell(\alpha) \;=\; a_0\,(\alpha - \alpha_{\text{L}=0}) \qquad\qquad\qquad \text{(away from stall)}$$

Here, a_0 is the *lift-curve slope*, and $\alpha_{\text{L}=0}$ is the *zero-lift angle* These can be measured or computed reasonably accurately with a potential-flow method.
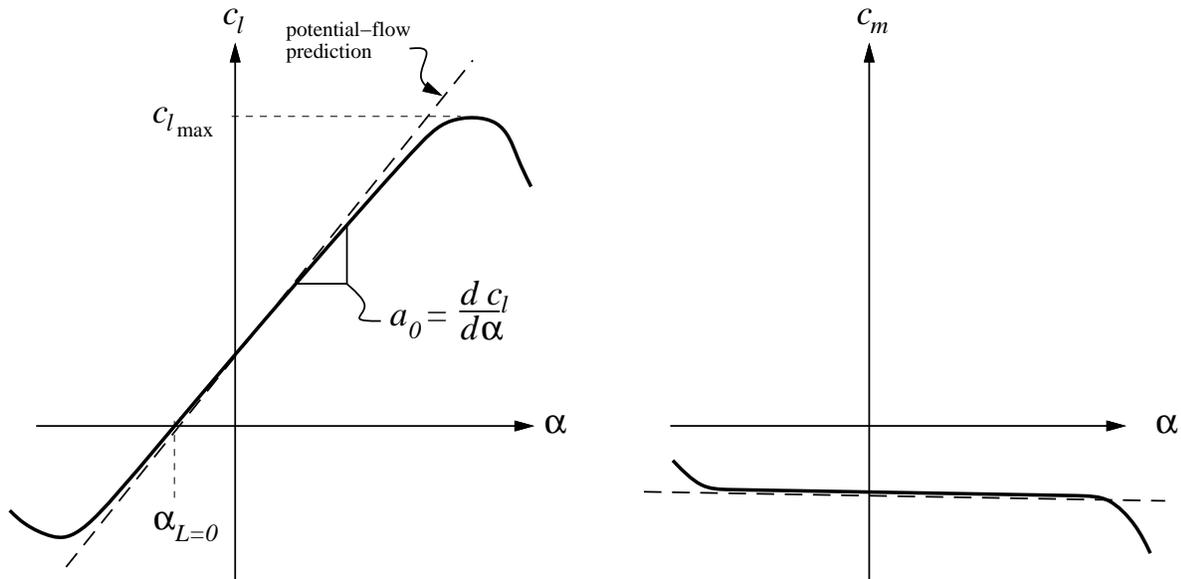
The moment coefficient $c_m(\alpha)$, when defined about the quarter-chord point, is very nearly constant away from stall. Again this is predicted well by potential-flow methods. Past stall, the $c_m(\alpha)$ curve deviates sharply from its constant value.

The drag coefficient c_d can be plotted versus α, as shown in the figure on the left. However, a more useful and more standard way is to plot c_ℓ vs c_d, with α simply a dummy parameter along the curve. This plot is called a *drag polar*, and is shown in the figure on the right.
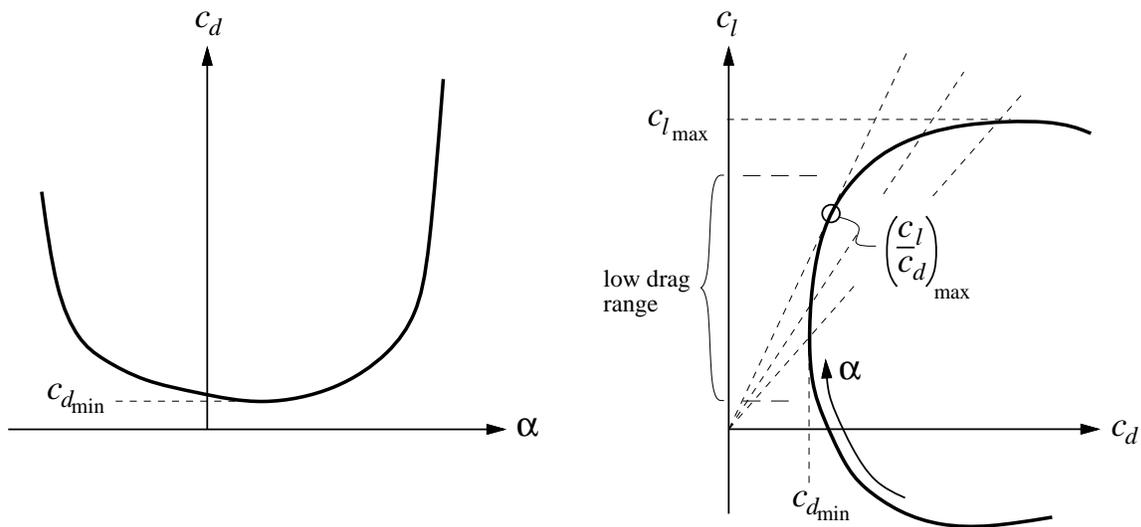
One reason for using the drag polar format is that when evaluating the aerodynamic performance of an airfoil, the α values are not really relevant. All that matters is the drag and how it compares to lift. The drag polar format compares these directly, and hence summarizes the most important features of the airfoil's drag characteristics in one plot. One such feature is the *maximum lift-to-drag ratio*, or $(c_\ell/c_d)_{\max}$, which is where a line from the origin lies tangent to the polar curve. The $c_{\ell_{\max}}$ and $c_{d_{\min}}$ values are also directly visible. An aerodynamicist might also note the *low-drag range* of lift coefficients where the airfoil naturally wants to operate.

It must be stressed that c_d values are roughly 100 times smaller than typical maximum c_ℓ values. Hence, the c_d axis on a polar plot is greatly enlarged.

A sample polar plot and $c_\ell(\alpha; Re)$ and $c_m(\alpha; Re)$ curves for an actual sailplane airfoil are shown below, for two different Reynolds numbers.