Introduction to Aerospace Engineering

Lecture slides
9 & 10

Airfoils and pressure distributions
Anderson 5.1-5.16 (exc. 5.11)
Topics

- Airfoils
- Pressure coefficient
History of wing profiles

Early years: many different airfoil descriptions

<table>
<thead>
<tr>
<th>Designation</th>
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<th>Diagram</th>
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History of wing profiles

National Advisory Committee for Aeronautics (NACA) – 1915
supported research & development (at Langley)
a.o. aerodynamics (first open windtunnels)
airfoils (large overview in 1933)
specific nomenclature to describe:
2412 means:
  2% camber at
  0.4 of the chord; and
  12% thickness/chord ratio (or 0.12)

retractable landing gear; engine nacelles, propellors, etc.
Airfoil Data

A well known source for airfoil data:

“Theory of Wing Sections”, by Abott & von Doenhoff

Contents

• The significance of wing-section characteristics
• Simple two-dimensional flows
• Theory of wing sections of finite thickness
• Theory of thin wing sections
• The effects of viscosity
• Families of wing sections
• Experimental characteristics of wing sections
• High-lift devices
• Effects of compressibility at subsonic speeds
NACA airfoil development

- National advisory committee on aeronautics
- A systematic investigation of the effect of maximum thickness, thickness distribution, camber, camber distribution, Reynolds number, L.E. roughness and flap deflection on the performance of airfoils for aeronautical application
- Most known are NACA 44xx, NACA 230xx, 63 and 64 series
NACA airfoil development

- Systematic in the sense that different camberlines were combined with different (symmetric) thickness distributions
\[ x_u = x - y_t \cdot \sin \varphi \]
\[ x_L = x + y_t \cdot \cos \varphi \]
Effect of airfoil maximum thickness

Re = 3.0\times10^6

\[ C_l \]

\[ C_d \]

\[ \alpha \] (°)

NACA 4415
NACA 4418
NACA 4421
NACA 4424
Effect of Reynolds number

NACA 4418

\( \alpha \) (°)

\( c_l \)

\( c_d \)

\( \text{Re} = 3 \times 10^6 \)

\( \text{Re} = 6 \times 10^6 \)

\( \text{Re} = 9 \times 10^6 \)
A turbulent and a laminar airfoil

Design $C_l$

$Re = 3.0 \times 10^6$

$\left( C_l/C_d \right)_{\text{max}}$

$C_l$

$C_d$

$\alpha$ ($^\circ$)

NACA 4418

NACA 63-418
Re = $3.0 \times 10^6$

- NACA 4418
- NACA 63-418
The effect of camber on the NACA 63-series

Re = $3.0 \times 10^6$

NACA 63-618
NACA 63-418
NACA 63-218
NACA 63-018
Lift-parameters

\( C_l \) (Lift coefficient) depends on (a.o.)
- airfoil or wing profile
- angle of attack (\( \alpha \))

\( \rho \) (air density) depends on:
- altitude & temperature (atmosphere)

\( V \) (air speed) and \( S \) (wing area) are design parameters

Lift due to **pressure differences** over the airfoil - Bernoulli
Airfoil data

The lift curve

\[ C_l \]

\[ C_{l_{\text{max}}} \]

\[ \alpha C_1 = 0 \]

\[ a_0 = \frac{dC_l}{d\alpha} = \text{lift curve slope} \]

stall due to flow separation

cambered airfoil

symmetrical airfoil
Airfoils

Stall due to flow separation

Attached flow

Separated flow

$C_1$ vs. $\alpha$
Pressure coefficient (negative is up)

Definition of pressure coefficient:

\[ C_p = \frac{p - p_\infty}{q_\infty} \]

Typical example

Stagnation point: \( p = p_t \) ... \( p_t - p_\infty = q_\infty \) \( \Rightarrow C_p = 1 \)
Example 5.6

• The pressure on a point on the wing of an airplane is $7.58 \times 10^4$ N/m$^2$. The airplane is flying with a velocity of 70 m/s at conditions associated with standard altitude of 2000m. Calculate the pressure coefficient at this point on the wing.

2000 m: $p_\infty = 7.95 \times 10^4$ N/m$^2$  $\rho_\infty = 1.0066$ kg/m$^3$

$$C_p = \frac{p - p_\infty}{\rho_\infty q_\infty}$$

$=>$  $C_p = -1.50$
Obtaining lift from pressure distribution

\[ V_\infty \]

\[ \theta \]

[Diagram showing a wing with leading and trailing edges, pressure distribution, and the equation \( dx = ds \cos \theta \).]
Obtaining lift from pressure distribution

Normal force per meter span: 

\[ N = \int_{LE}^{TE} p_l \cos \theta \, ds - \int_{LE}^{TE} p_u \cos \theta \, ds \]

with \( ds \cos \theta = dx \)

\[ N = \int_{0}^{c} p_l \, dx - \int_{0}^{c} p_u \, dx \]

Write dimensionless force coefficient:

\[ C_n = \frac{N}{\frac{1}{2} \rho V_\infty^2 c} = \frac{N}{q_\infty c} \]

\[ C_n = \int_{0}^{1} \left( C_{p_l} - C_{p_u} \right) d \left( \frac{x}{c} \right) \]
\[ T = L \sin \alpha - D \cos \alpha \]
\[ N = L \cos \alpha + D \sin \alpha \]

\( \alpha = \text{angle of attack} \)
Obtaining lift from normal force coefficient

\[ L = N \cos \alpha - T \sin \alpha \quad c_l = c_n \cos \alpha - c_t \sin \alpha \]

\[ \frac{L}{q_\infty c} = \frac{N}{q_\infty c} \cos \alpha - \frac{T}{q_\infty c} \sin \alpha \]

For small angle of attack \( \alpha \leq 5^\circ \): \( \cos \alpha \approx 1, \quad \sin \alpha \approx 0 \)

\[ C_l \approx \frac{1}{C} \int_0^1 (C_{p_l} - C_{p_u}) \, d(x) \]
Example 5.11

Consider an airfoil with chord length $c$ and the running distance $x$ measured along the chord. The leading edge is located at $x/c=0$ and the trailing edge at $x/c=1$. The pressure coefficient variations over the upper and lower surfaces are given as

$$C_{p,u} = 1 - 300 \left( \frac{x}{c} \right)^2 \quad \text{for } 0 \leq \left( \frac{x}{c} \right) \leq 0.1$$

$$C_{p,u} = -2.2777 + 2.2777 \left( \frac{x}{c} \right) \quad \text{for } 0.1 \leq \left( \frac{x}{c} \right) \leq 1.0$$

$$C_{p,l} = 1 - 0.95 \left( \frac{x}{c} \right) \quad \text{for } 0 \leq \left( \frac{x}{c} \right) \leq 1.0$$

Calculate the normal force coefficient.
Compressibility correction of the pressure coefficient

Approximate theoretical correction (valid for $0 < M < 0.7$):

$$C_p = \frac{C_{p,0}}{\sqrt{1 - M_\infty^2}}$$

Prandtl-Glauert Rule
Compressibility correction for lift coefficient

\[
C_l \approx \frac{1}{c} \int_{0}^{c} \frac{(C_{p_l} - C_{p_u})_o \ d(x)}{\sqrt{1 - M_\infty^2}} = \\
C_{l,0} \equiv \frac{1}{c} \int_{0}^{c} (C_{p_l} - C_{p_u})_o \ d(x) \\
C_l = \frac{C_{l,0}}{\sqrt{1 - M_\infty^2}}
\]
Example 5.12

- Consider an NACA 4412 airfoil at an angle of attack of 4°. If the free-stream Mach number is 0.7, what is the lift coefficient?

- From App. D for angle of attack of 4°, \( c_l = 0.83 \). However, these data were obtained at low speeds. Use the Prandtl-Glauert correction to calculate the lift coefficient at \( M = 0.7 \).
Critical Mach number and critical pressure coefficient

M = 0.3

M = 0.5

M = 0.6

Critical Mach number for the airfoil

$M_{\text{peak}} = 0.44$

$M_{\text{peak}} = 0.71$

$M_{\text{peak}} = 1.00$
Critical pressure coefficient

\[ C_{p,\text{cr}} = f(M_\infty) \]

Thicker airfoil reaches critical pressure coefficient at a lower value of \( M_\infty \)