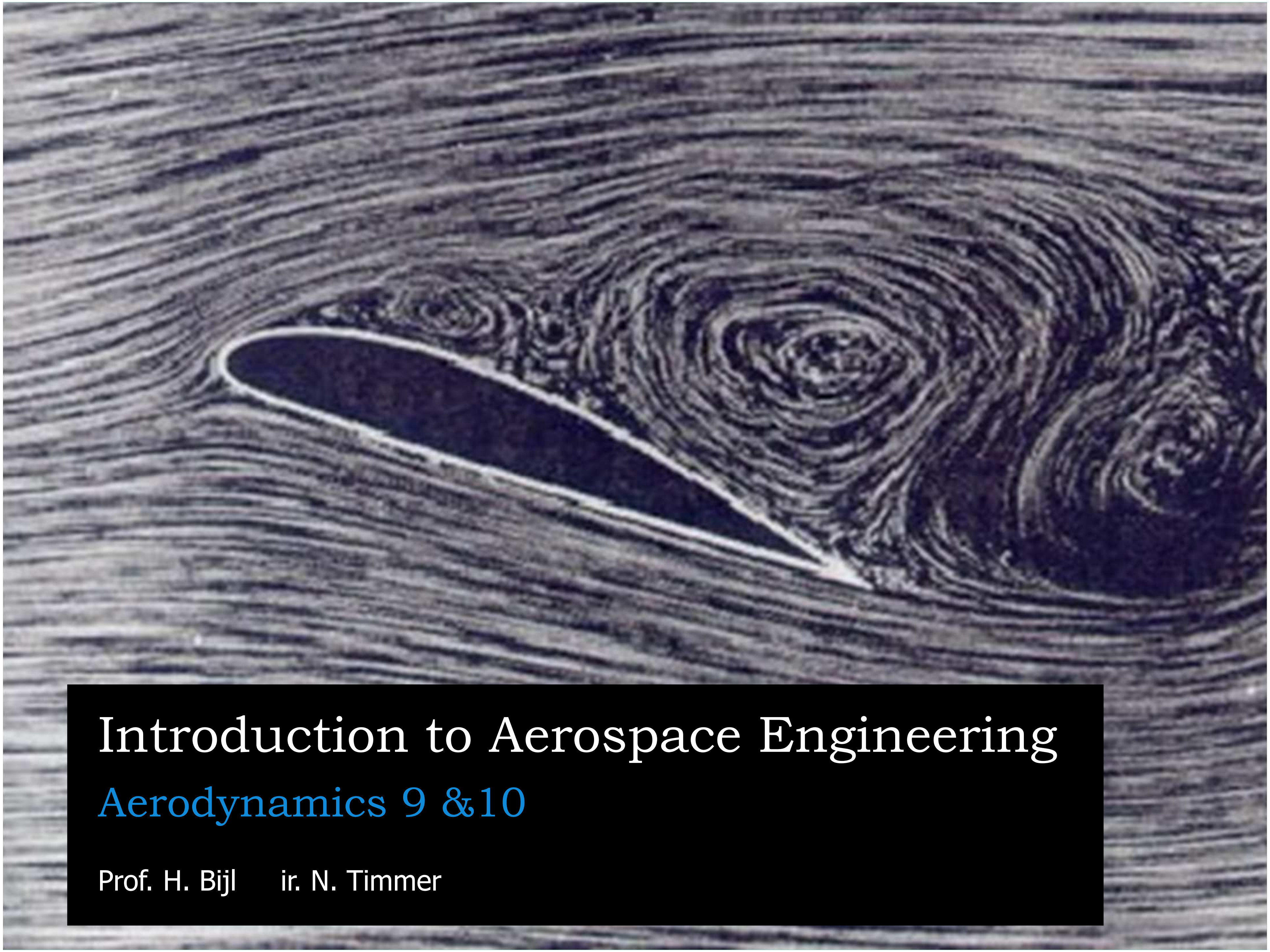
Introduction to Aerospace Engineering

Lecture slides



Introduction to Aerospace Engineering

Aerodynamics 9 &10

Prof. H. Bijl ir. N. Timmer

9 & 10

Airfoils and pressure distributions

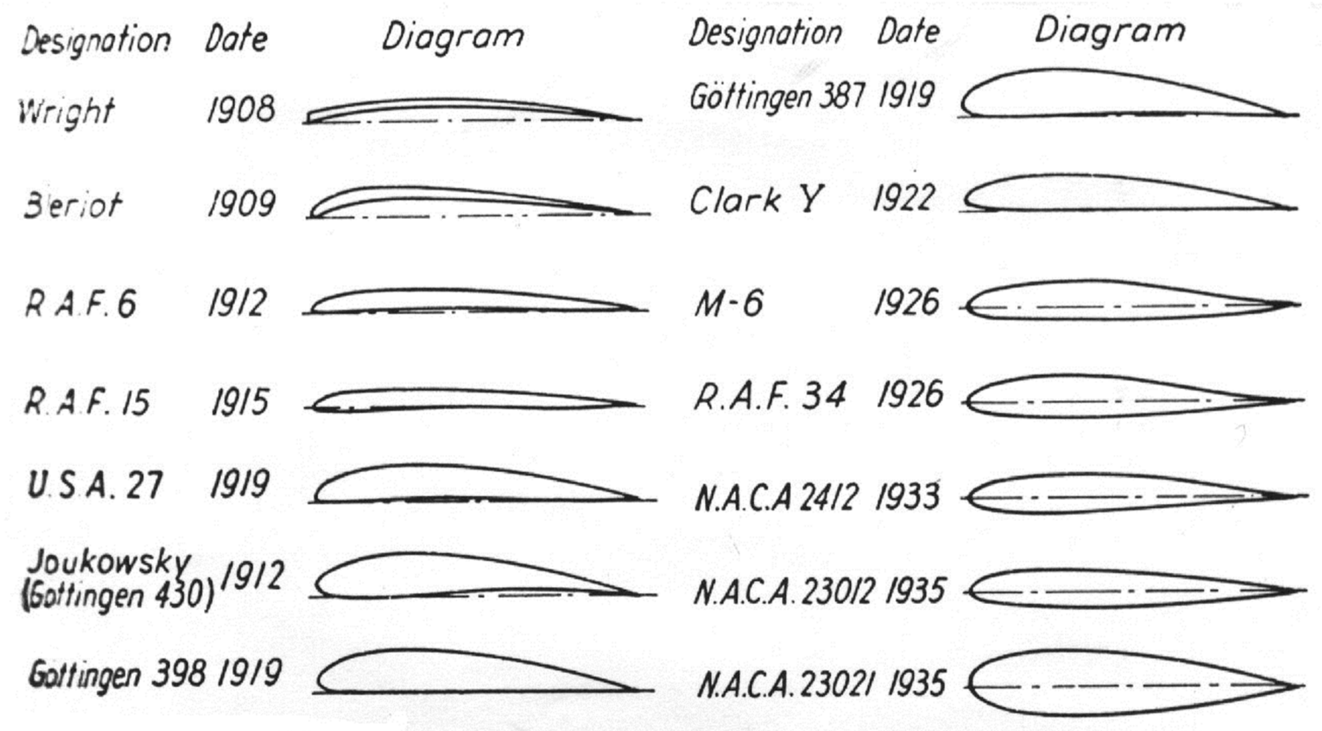
Anderson 5.1- 5.16 (exc. 5.11)

Topics

- Airfoils
- Pressure coefficient

History of wing profiles

Early years: many different airfoil descriptions



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

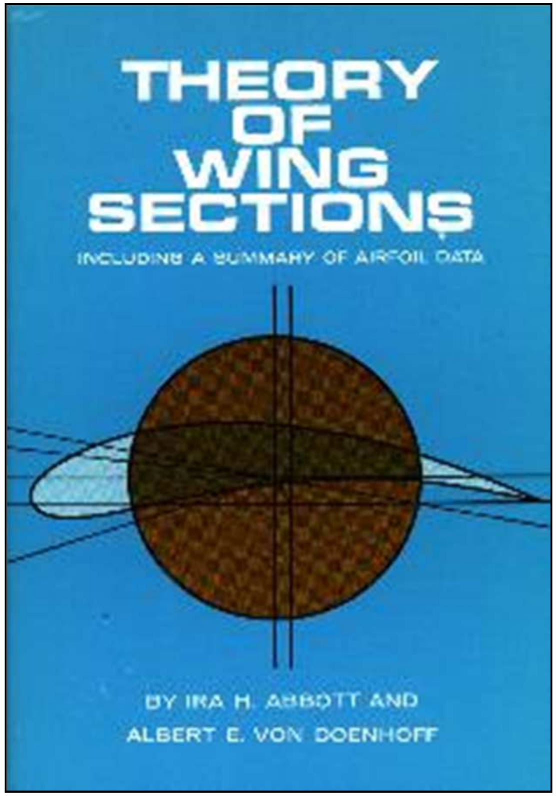
0.**4** of the chord; and

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

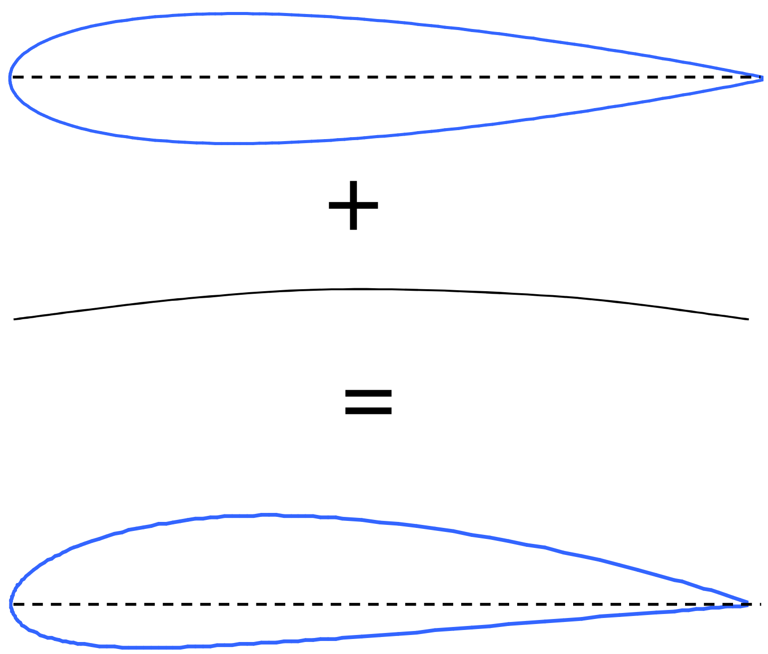
- The significance of wing-section characteristics
- Simple two-dimensional flows
- Theory of wing sections of finite thickness
- Theory of thin wing sections
- The effects of viscosity
- Families of wing sections
- Experimental characteristics of wing sections
- High-lift devices
- Effects of compressibility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

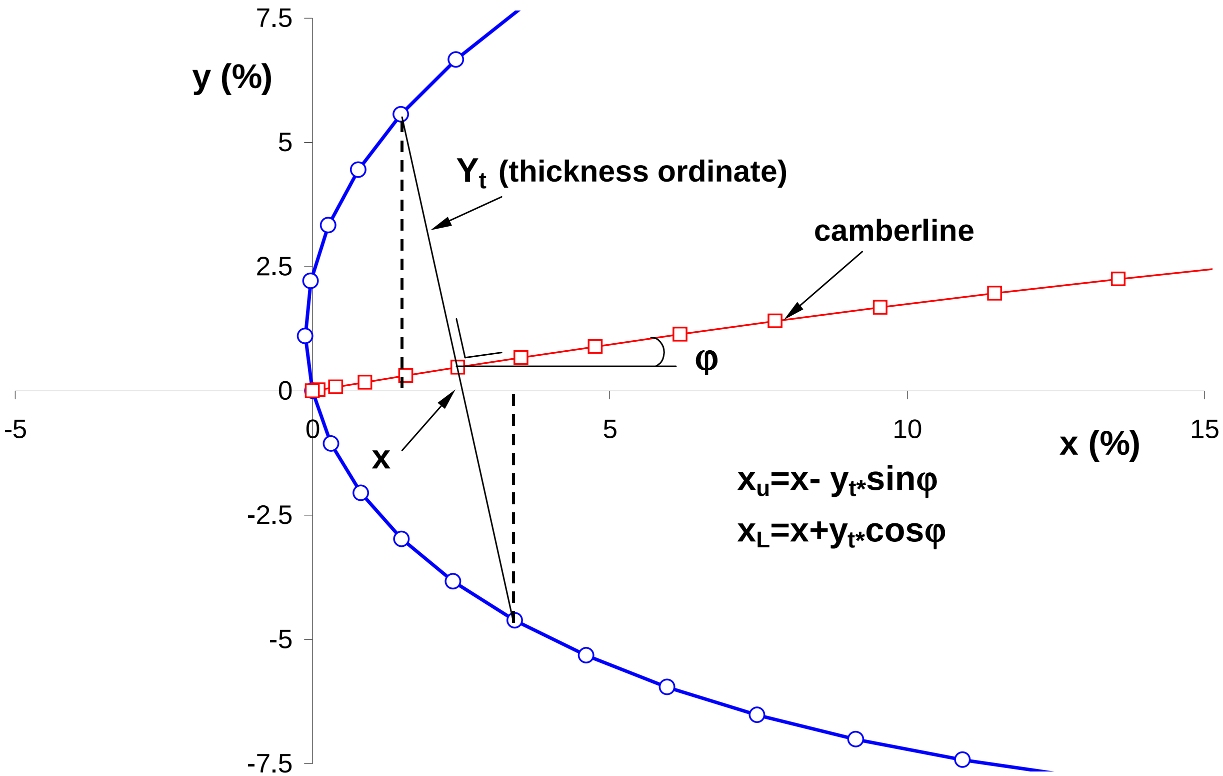
NACA airfoil development

- National advisory committee on aeronautics
- A systematic investigation of the effect of maximum thickness, thickness distrubution, 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

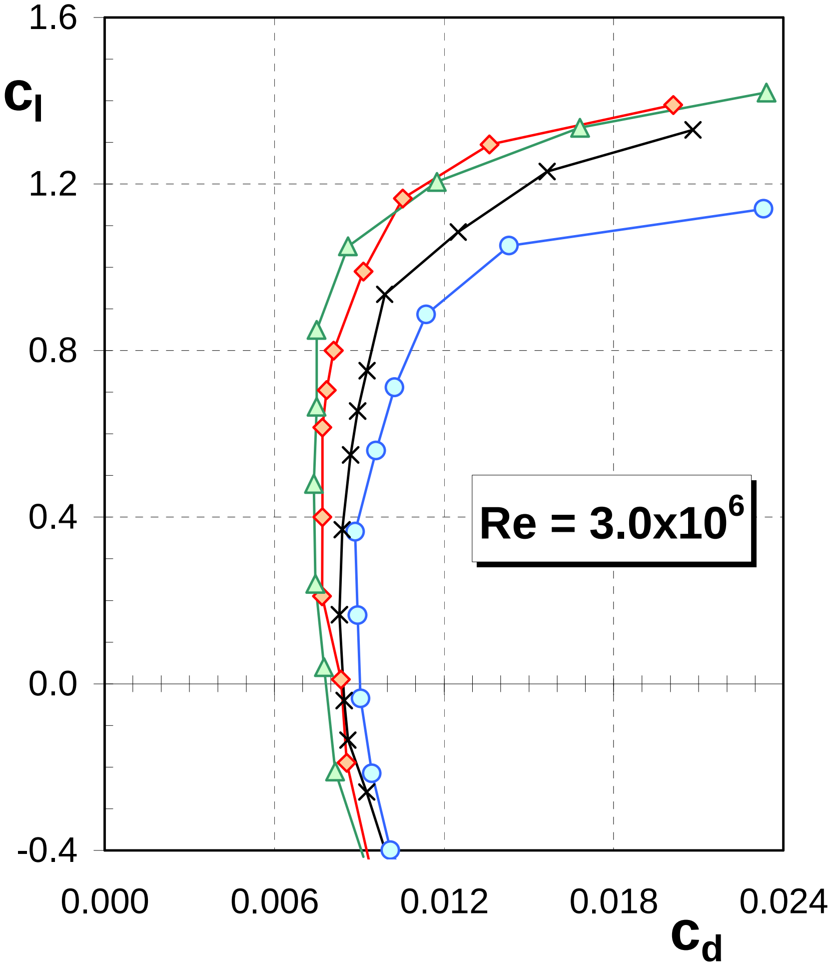
- Systematic in the sense that different camberlines were combined with different (symmetric) thickness distributions

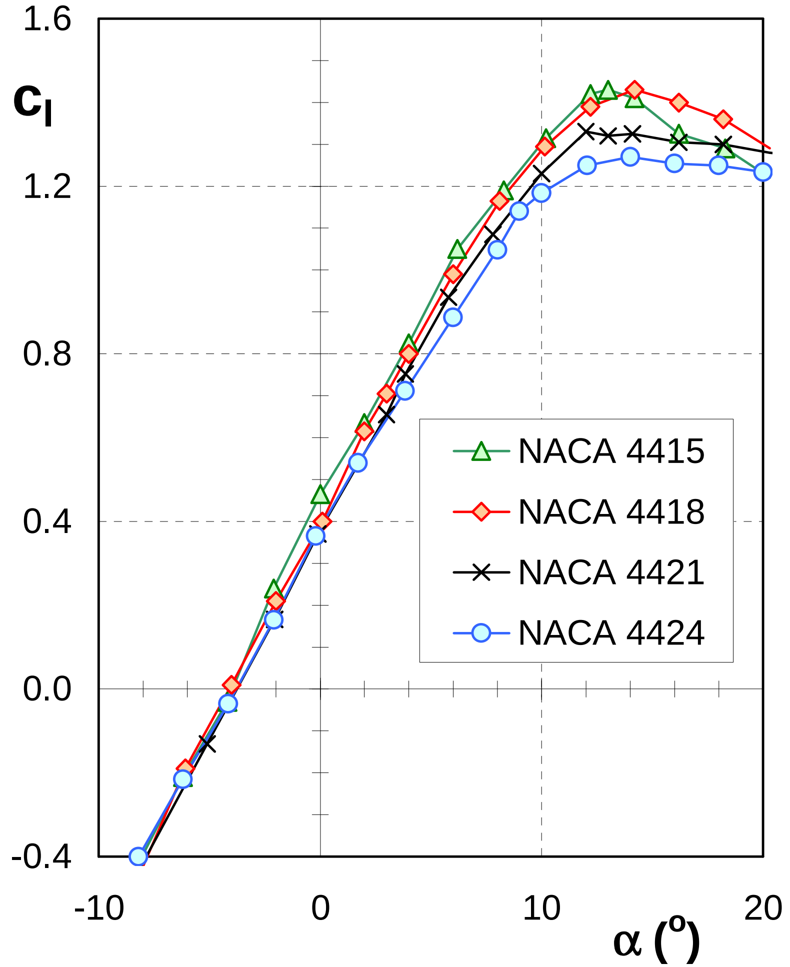




7.5
y (%)
5
Yt (thickness ordinate)
camberline
2.5
φ
0
-5
0
5
10
15
x
x (%)
xu=x- yt*sinφ
xL=x+yt*cosφ
-2.5
-5
-7.5

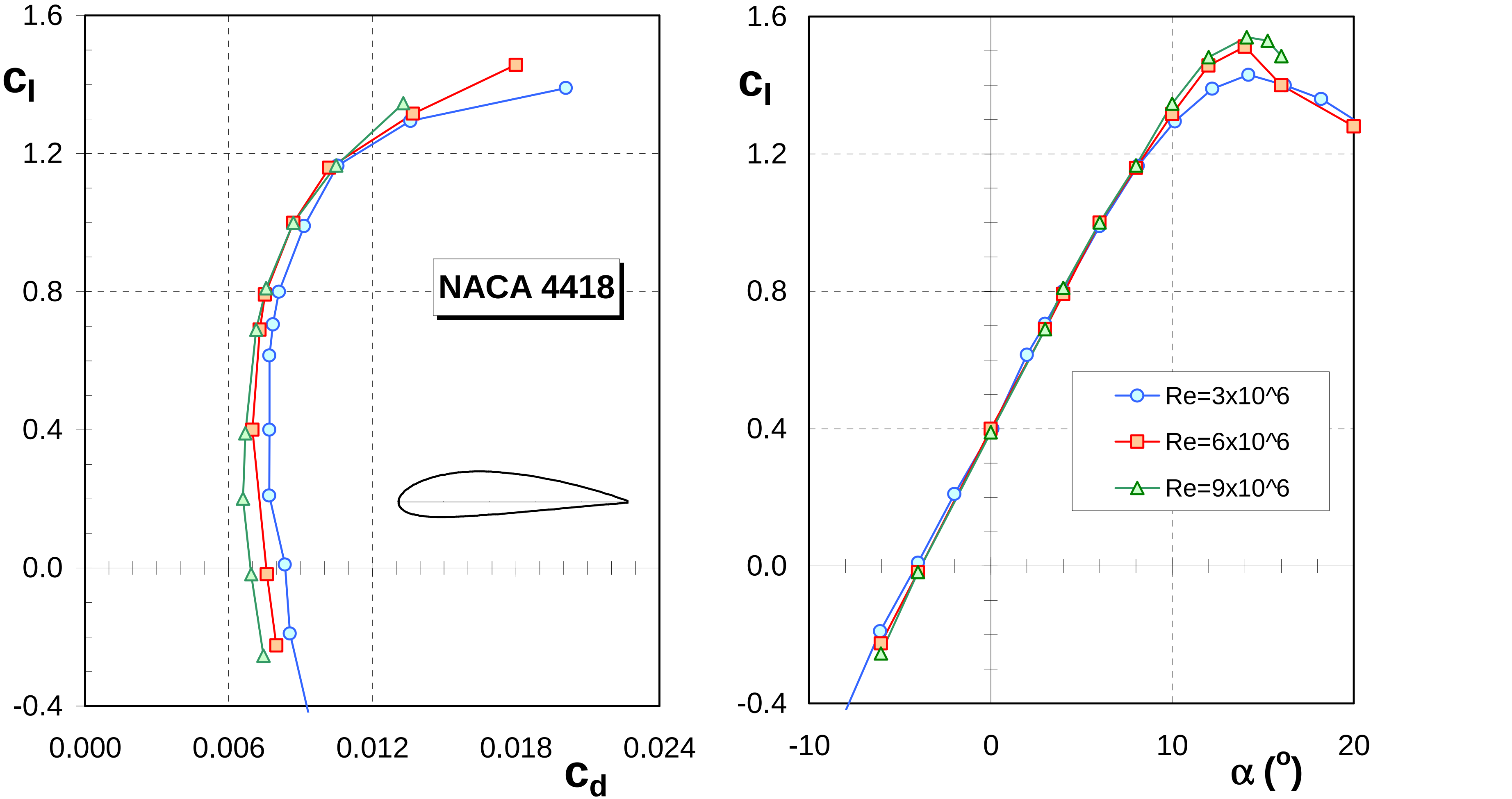
Effect of airfoil maximum thickness

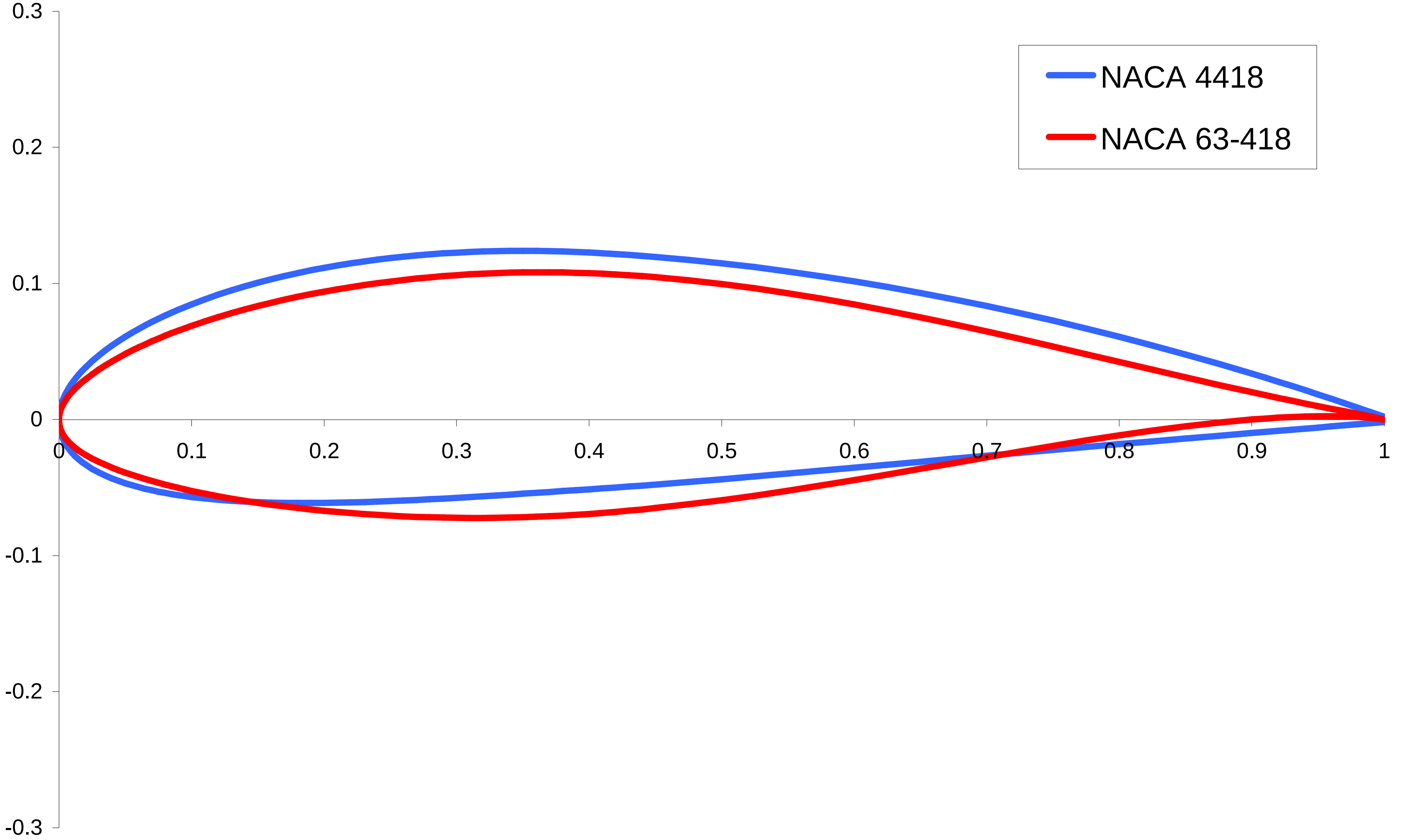




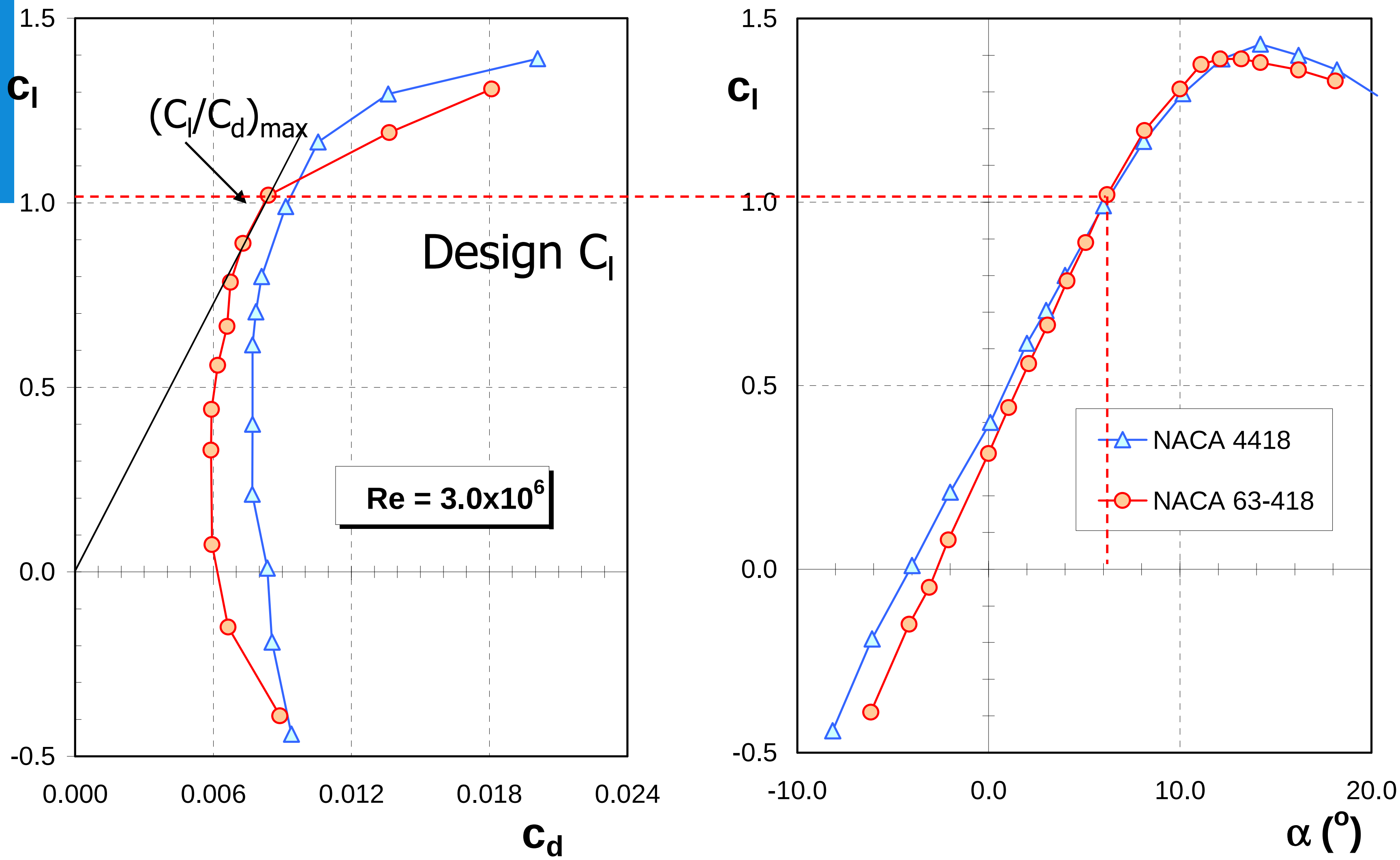
TUDelft

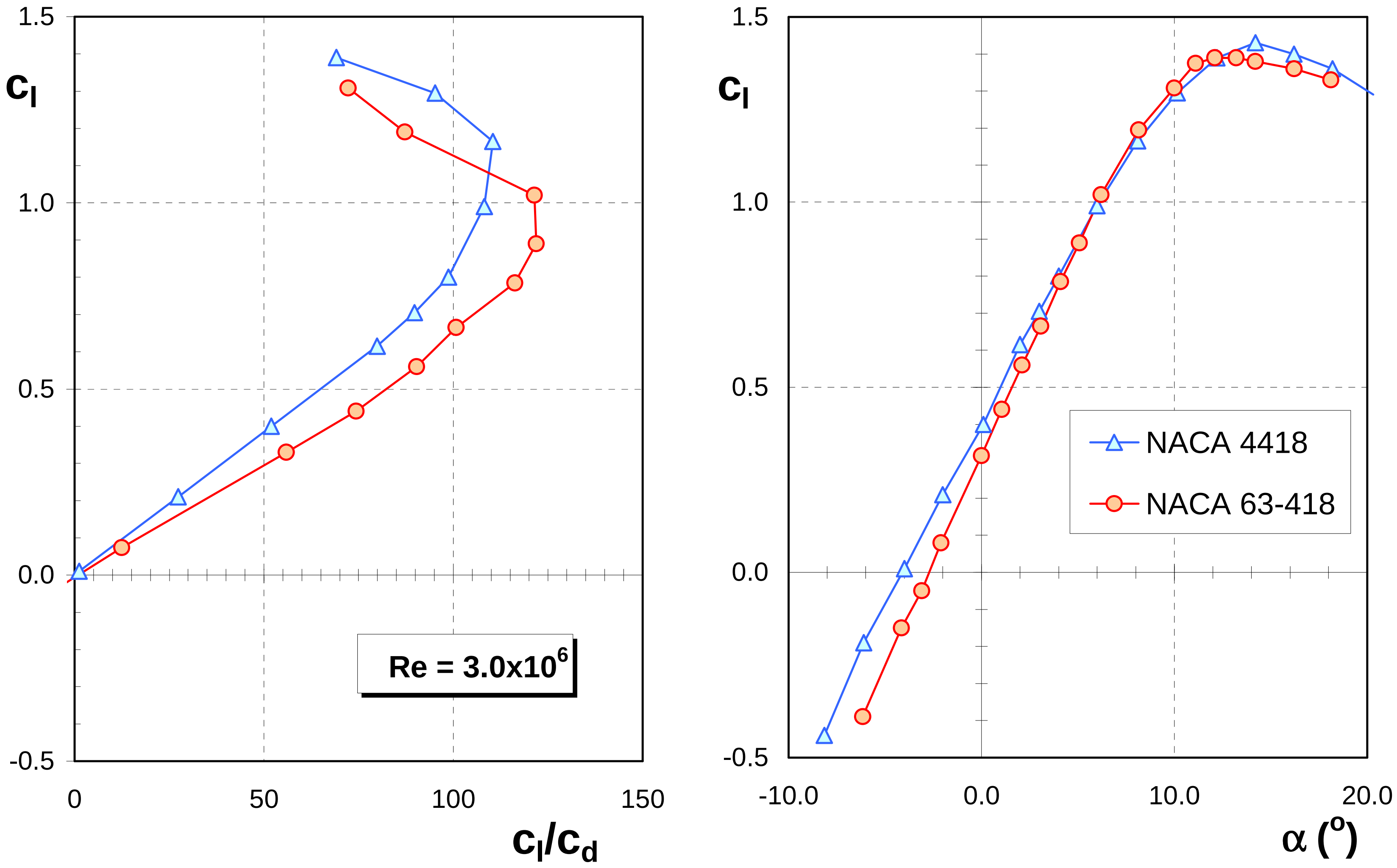
Effect of Reynolds number




NACA 4418
NACA 63-418
0.3
0.2
0.1
0
-0.1
-0.2
-0.3
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1

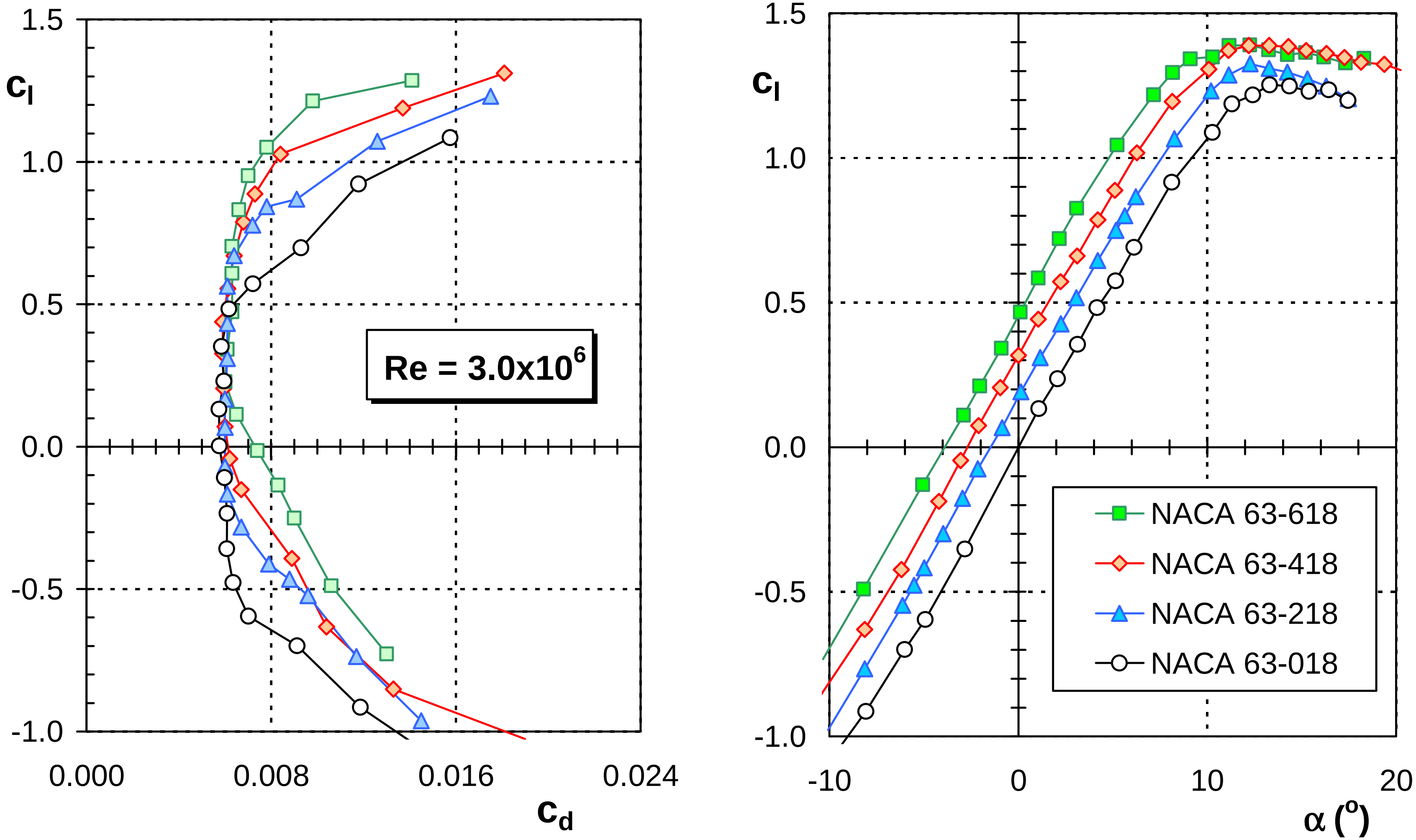
A turbulent and a laminar airfoil



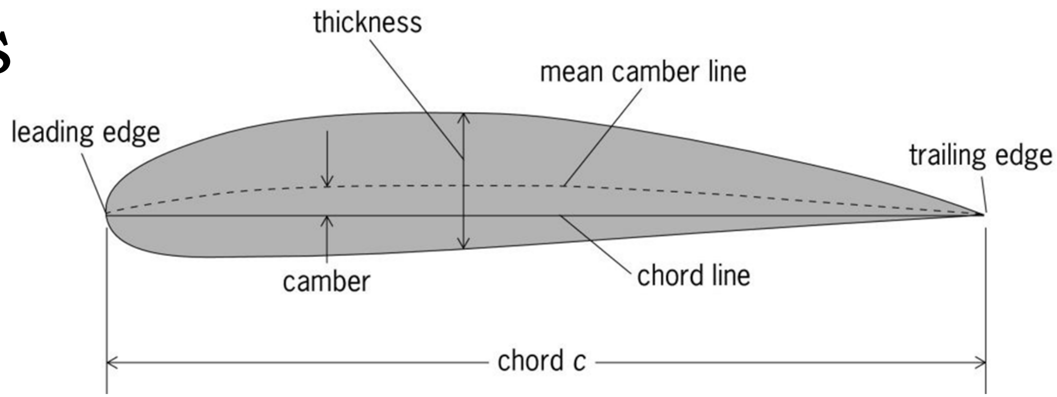


1.5
1.0
0.5
0.0
-0.5
c_l
0
50
100
150
c_l/c_d
Re = 3.0x10^6
1.5
1.0
0.5
0.0
-0.5
c_l
-10.0
0.0
10.0
20.0
α (^{o})
NACA 4418
NACA 63-418

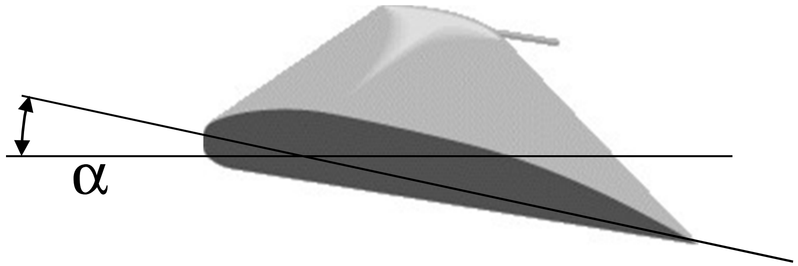
The effect of camber on the NACA 63-series



Lift-parameters



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

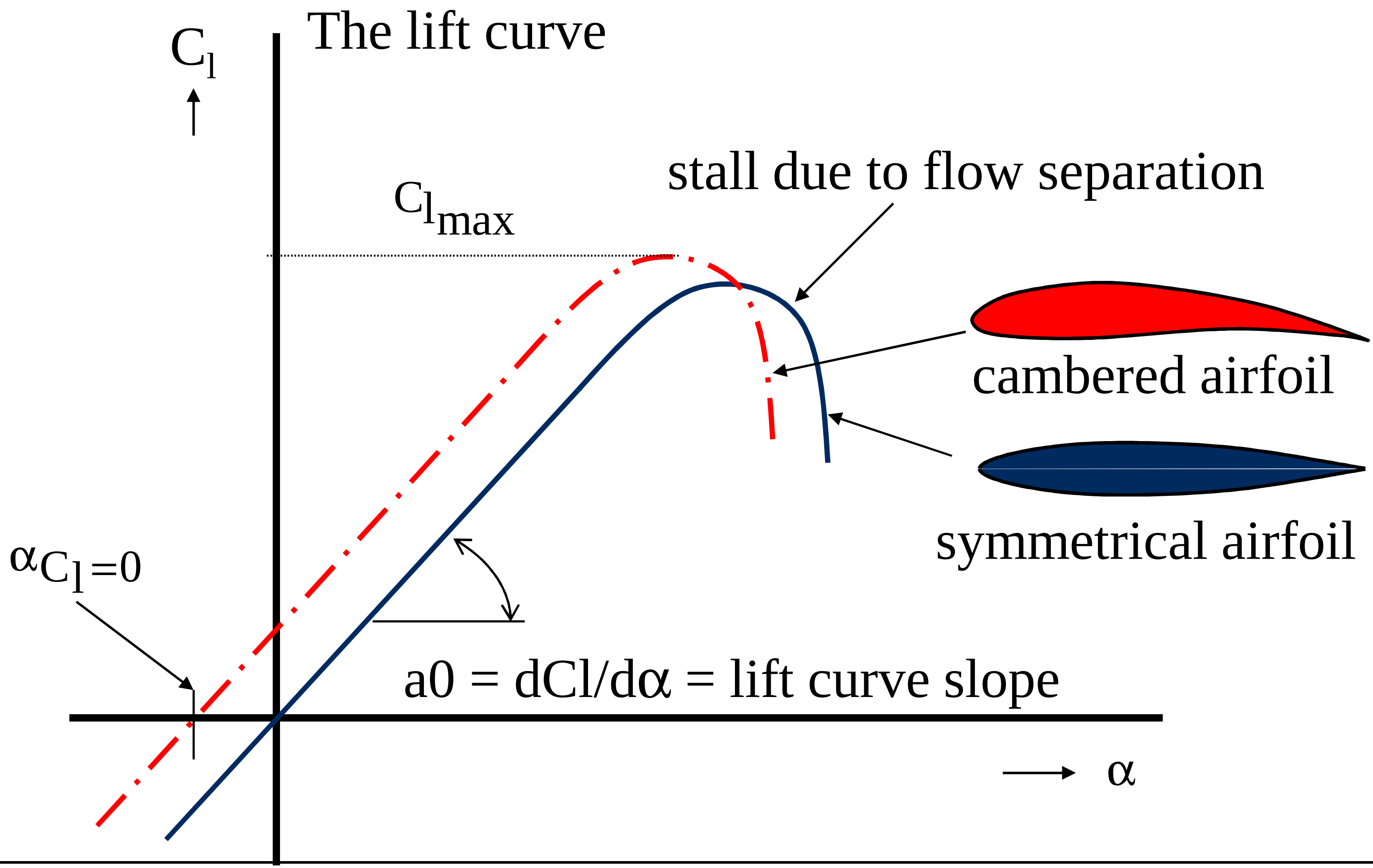


ρ (air density) depends on:
- altitude & temperature (atmosphere)

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

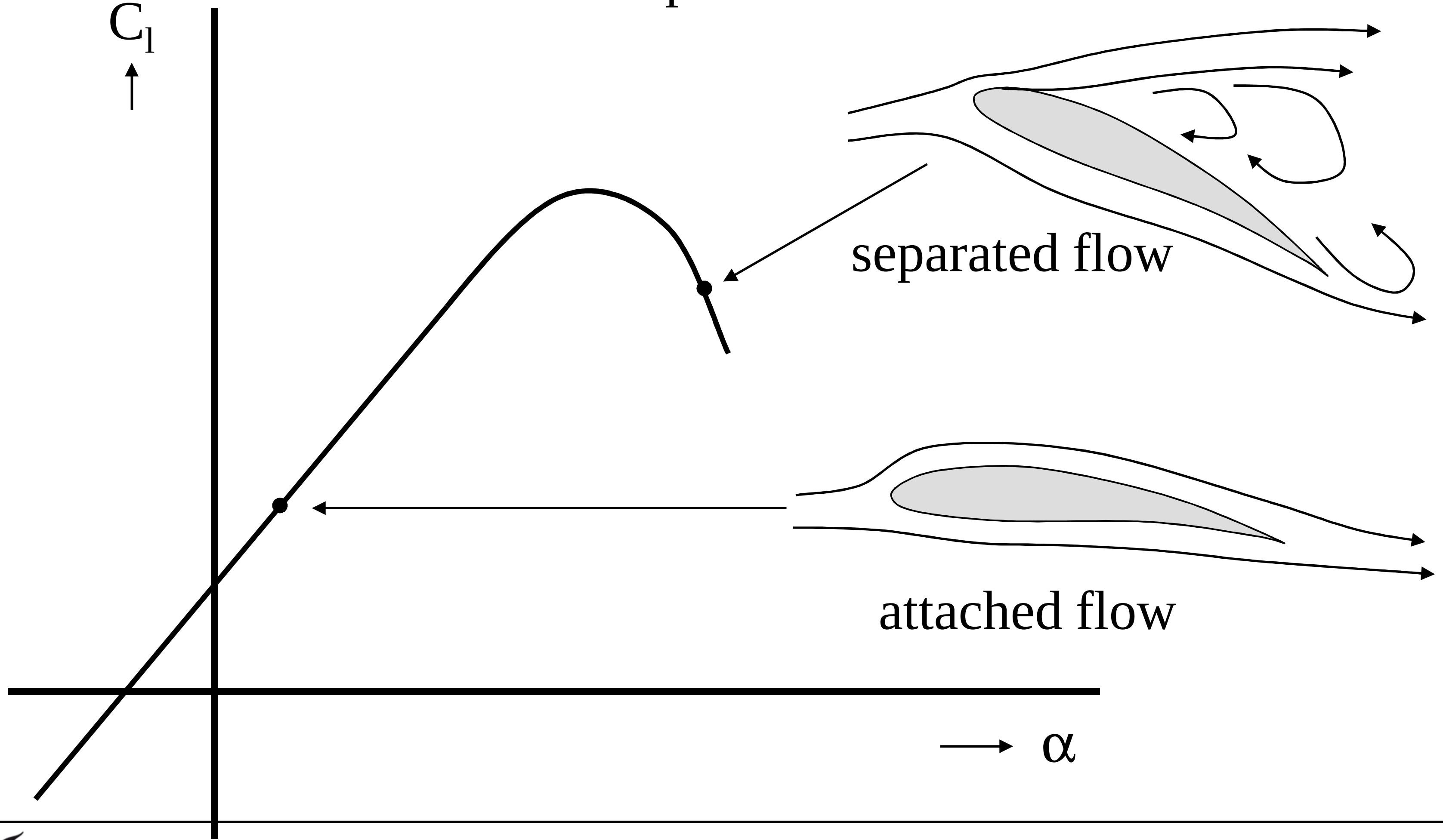
Lift due to **<u>pressure differences</u>** over the airfoil - Bernoulli

Airfoil data



Airfoils

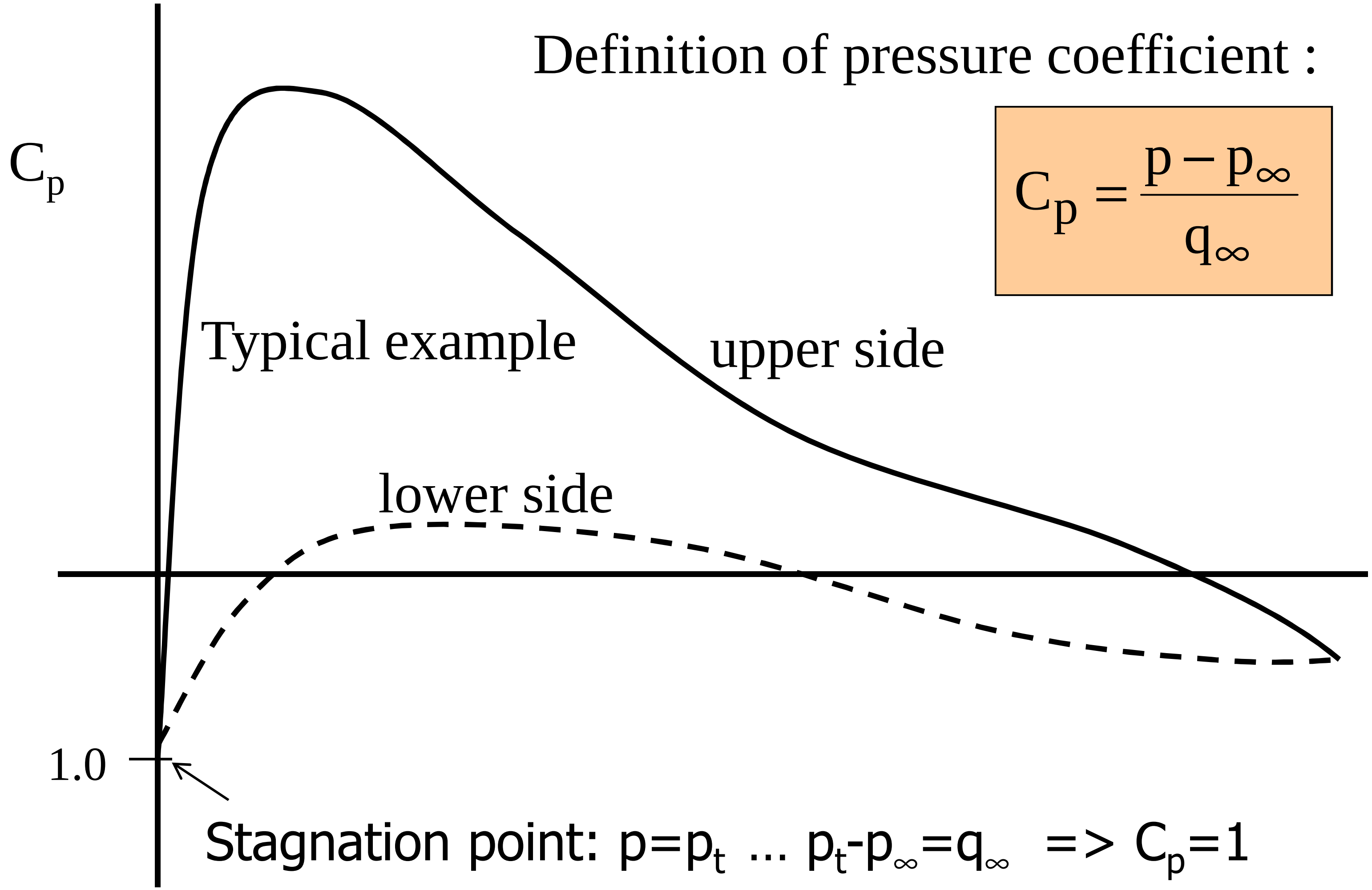
Stall due to flow separation



Pressure coefficient (negative is up)

Definition of pressure coefficient :

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



Stagnation point: $p=p_t$ … $p_t - p_\infty = q_\infty$ => $C_p=1$

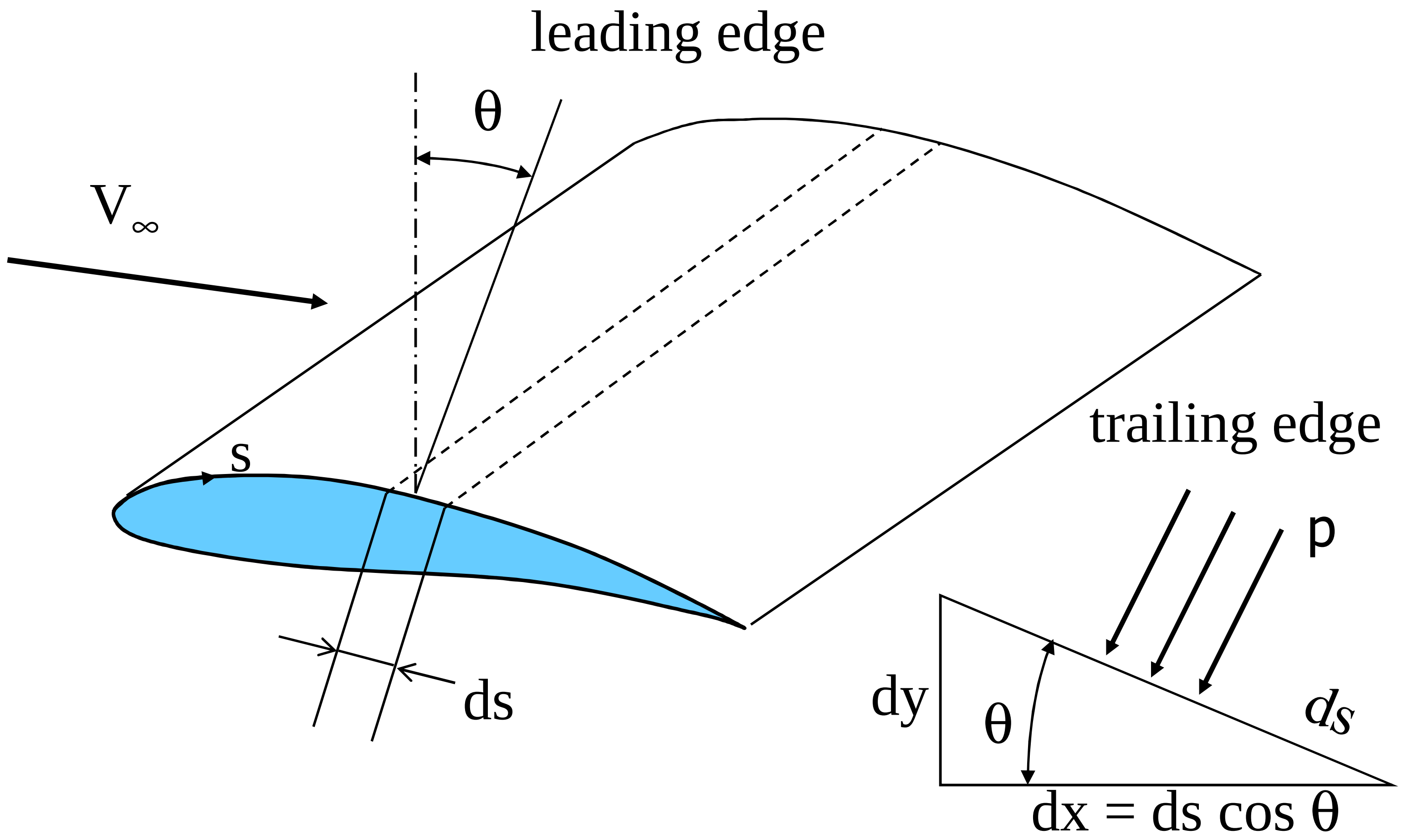
Example 5.6

- The pressure on a point on the wing of an airplane is 7.58×10^4 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: $p_\infty = 7.95.10^4$ N/m^2 $\rho_\infty = 1.0066$ kg/m^3

$$C_p = \frac{p - p_\infty}{q_\infty} \quad => \quad C_p = -1.50$$

Obtaining lift from pressure distribution



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 $\mathrm{ds}\cos\theta = \mathrm{dx}$ $\qquad N = \int_0^c p_l dx - \int_0^c p_u dx$

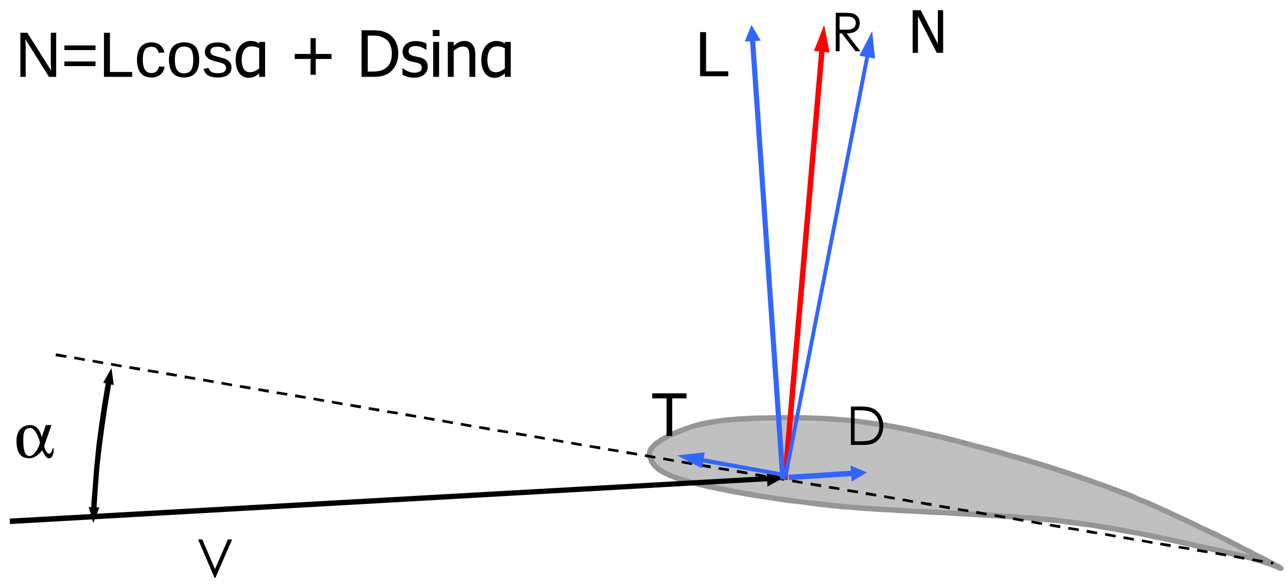
Write dimensionless force coefficient : $C_n = \dfrac{N}{\frac{1}{2}\rho V_\infty^2 c} = \dfrac{N}{q_\infty c}$

$$C_n = \int_0^1 \frac{p_l - p_\infty}{q_\infty} d\left(\frac{x}{c}\right) - \int_0^1 \frac{p_u - p_\infty}{q_\infty} d\left(\frac{x}{c}\right)$$

$$C_n = \int_0^1 \left(C_{p_l} - C_{p_u}\right) d\left(\frac{x}{c}\right)$$

$T = L\sin\alpha - D\cos\alpha$

$N = L\cos\alpha + D\sin\alpha$



α = angle of attack

Obtaining lift from normal force coefficient

$$L = N\cos\alpha - T\sin\alpha \qquad c_l = c_n\cos\alpha - c_t\sin\alpha$$

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

For small angle of attack $\alpha \le 5^o: \quad \cos\alpha \approx 1, \quad \sin\alpha \approx 0$

$$C_l \approx \frac{1}{c}\int_0^1 \left(C_{p_l} - C_{p_u}\right) d\left(x\right)$$

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

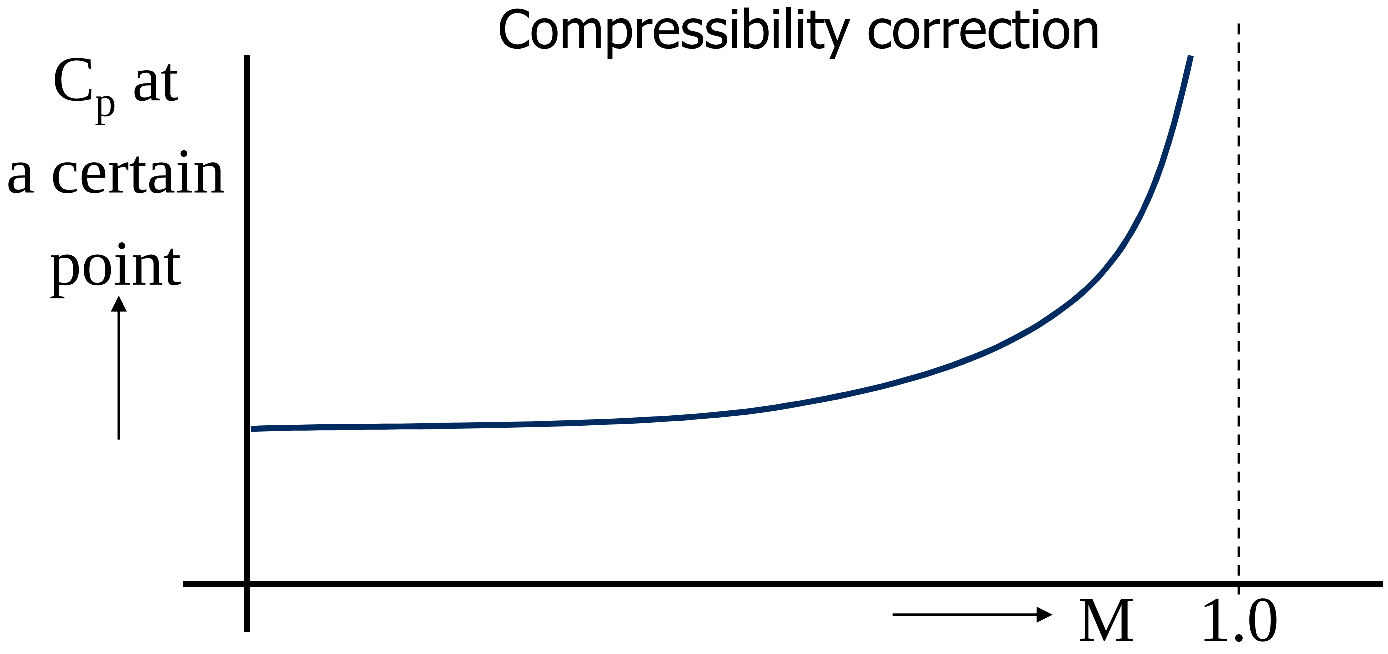
$$C_{p,u} = 1 - 300\left(\frac{x}{c}\right)^2 \qquad \text{for } 0 \le \left(\frac{x}{c}\right) \le 0.1$$

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

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

Calculate the normal force coefficient.

Compressibility correction of the pressure coefficient



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

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

Prandtl-Glauert Rule

Compressibility correction for lift coefficient

$$C_l \approx \frac{1}{c}\int_0^c \frac{\left(C_{p_l} - C_{p_u}\right)_o d\left(x\right)}{\sqrt{1-M_\infty^2}} =$$

$$C_{l,0} \equiv \frac{1}{c}\int_0^c \left(C_{p_l} - C_{p_u}\right)_o d\left(x\right)$$

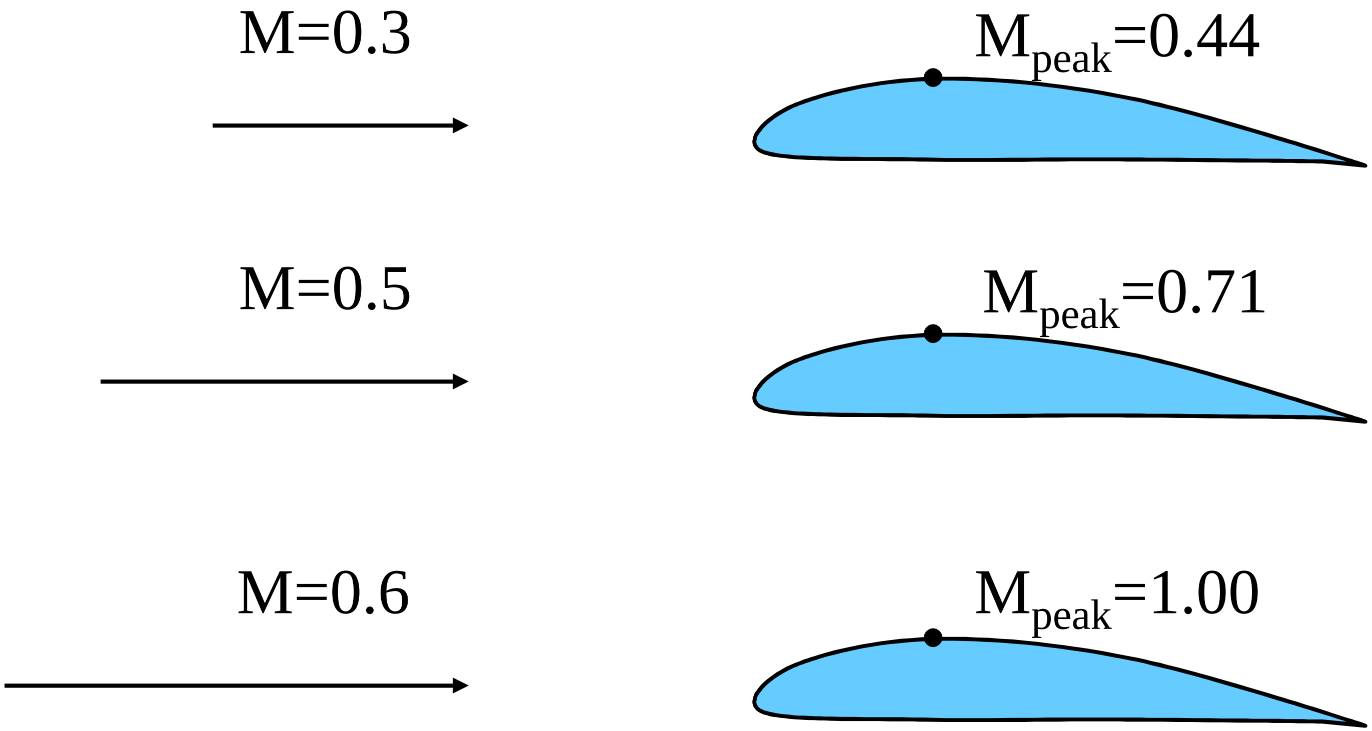
$$C_l = \frac{C_{l,0}}{\sqrt{1-M_\infty^2}}$$

Example 5.12

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

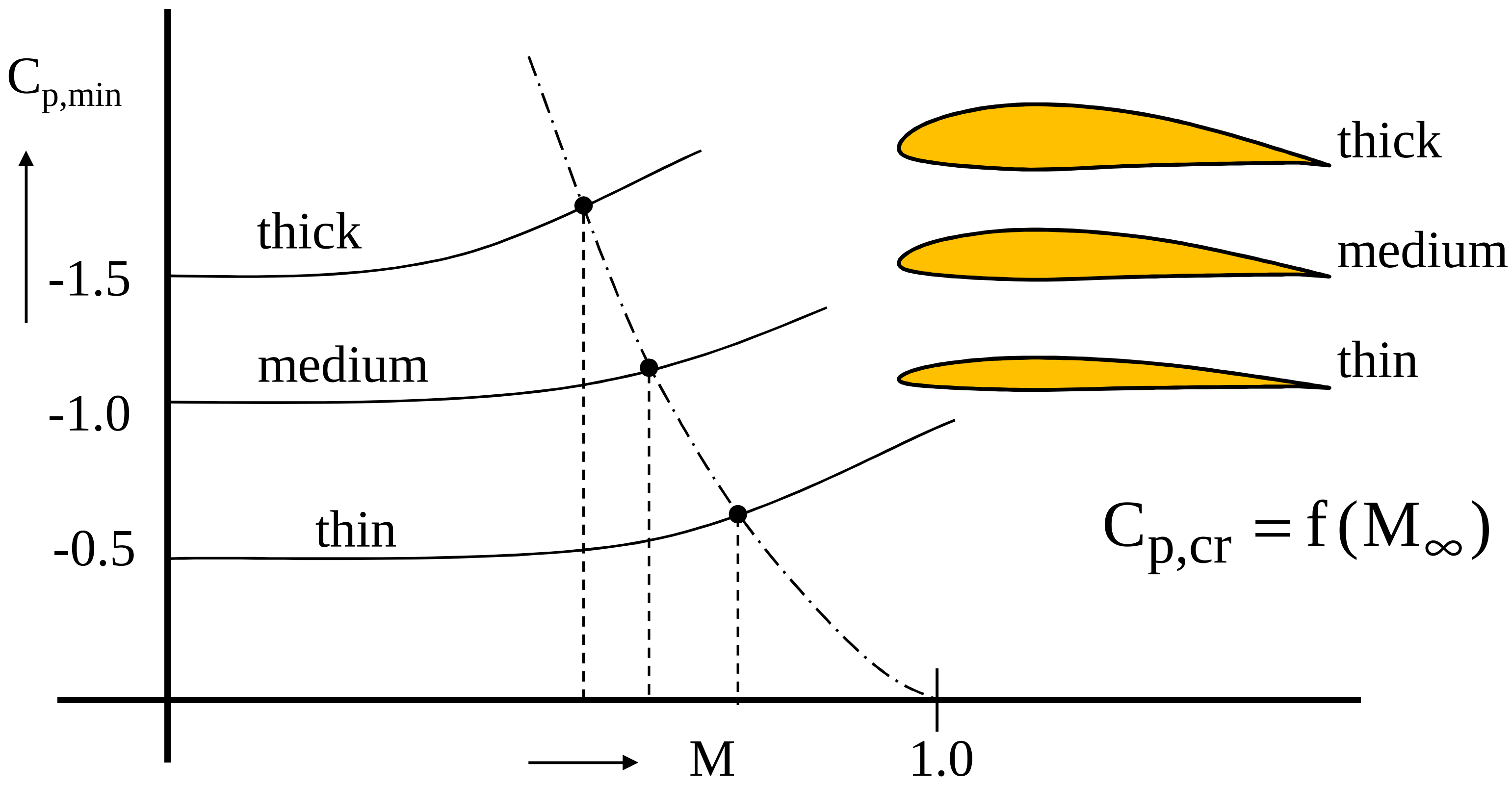
- - From App. D for angle of attack of 4^o, c_l = 0.83.However, these data were obtained at low speeds. Use the Prandtl-Glauert correction to calculate the lift coefficient at M=0.7

Critical Mach number and critical pressure coefficient



Critical Mach number for the airfoil

Critical pressure coefficient



Thicker airfoil reaches critical pressure coefficient at a lower value of M_∞