AE 343: Aerodynamic laboratory
Low speed flow past airfoil

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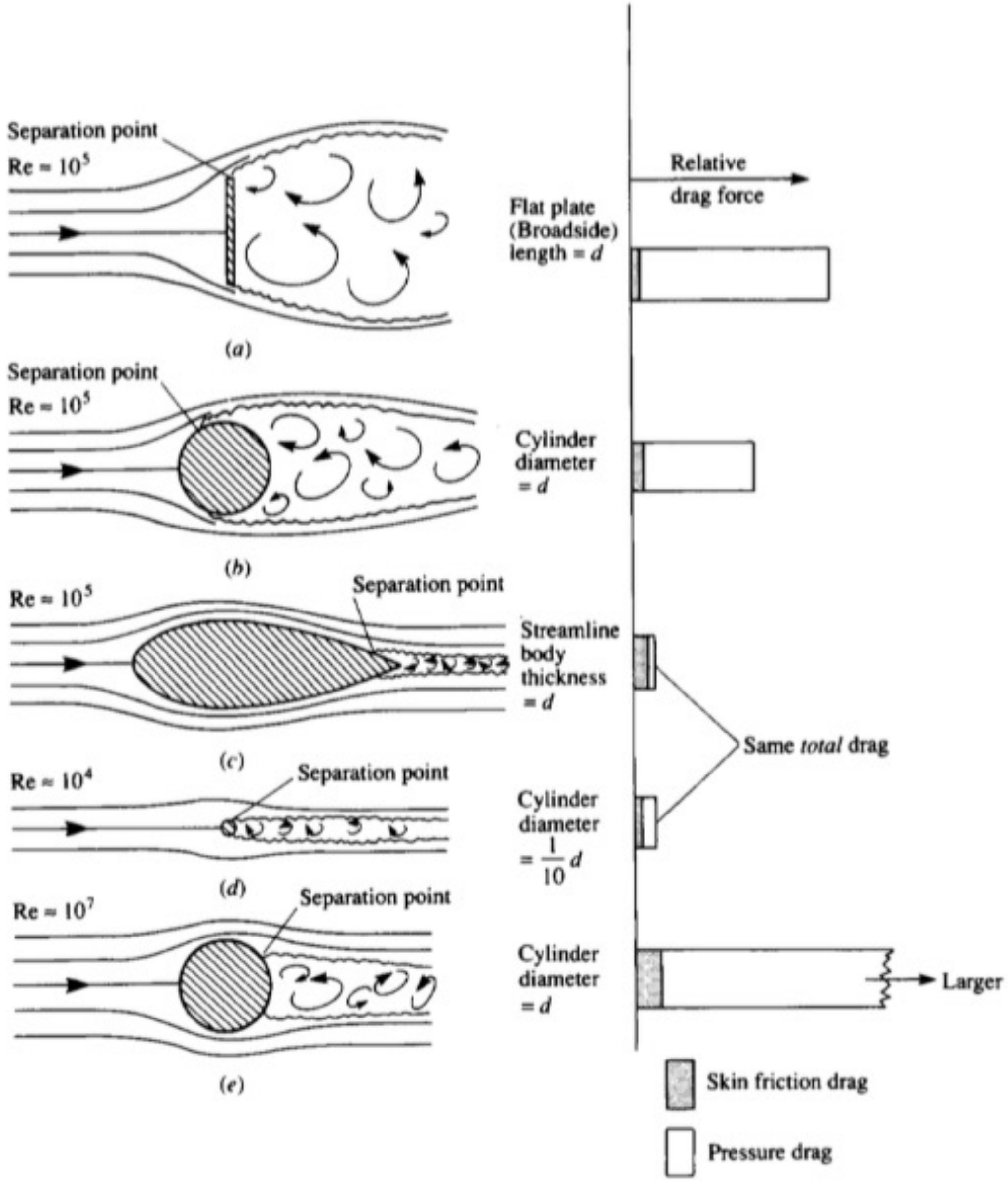
Background

1. Bluff and streamlined bodies
2. Nomenclature (chord, camber, sweep etc.)
3. Aerodynamic forces & moments
4. Estimation of aerodynamic coefficients
5. Aerodynamic center & center of pressure
6. Drag estimation from wake velocity measurements

Bluff and streamlined bodies



Pressure drag dominates for a **blunt/bluff** body

Skin-friction drag dominates for a **streamlined** body

Types of drag

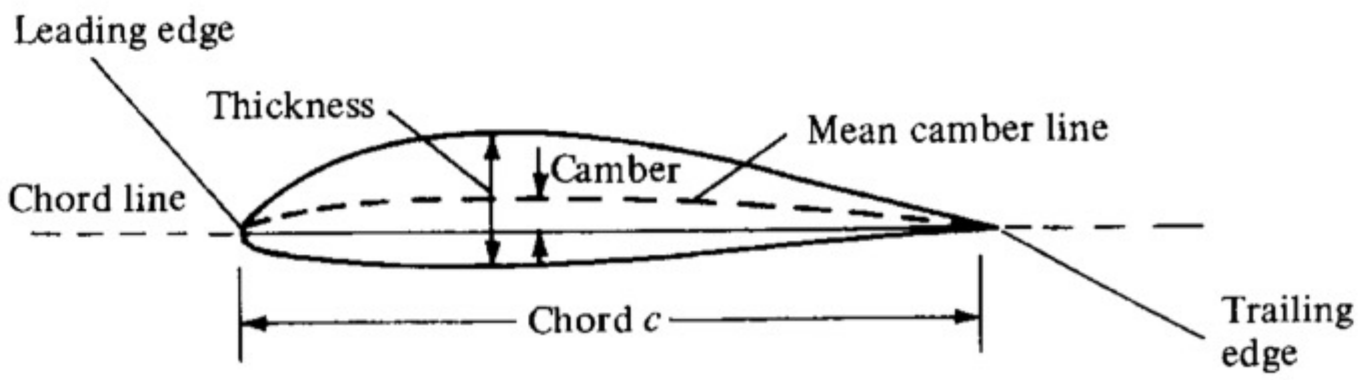
1. Zero-lift drag / profile drag c_{d0} (skin-friction drag + form drag)
2. Drag due to lift / induced drag
3. Wave drag

In low speed flows at small angles of attack, our objective is to **reduce skin friction drag by delaying the transition to turbulence**

Some factors affecting turbulence transition

1. Surface roughness
2. Atmospheric turbulence and engine noise
3. Pressure gradient
4. Surface heating/cooling
5. Compressibility (Mach number effects)
6. Suction/blowing on the surface

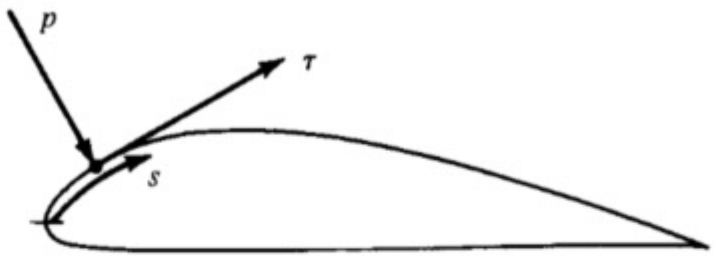
Airfoil geometry and nomenclature



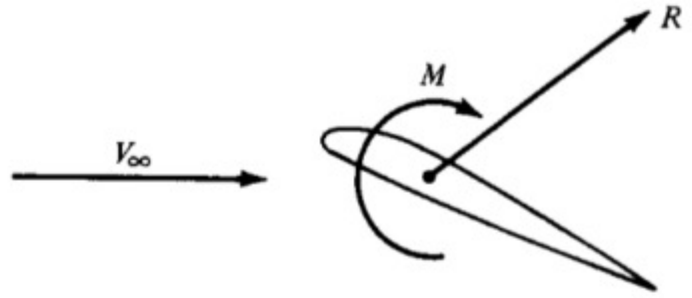
4-digit series, NACA 2412

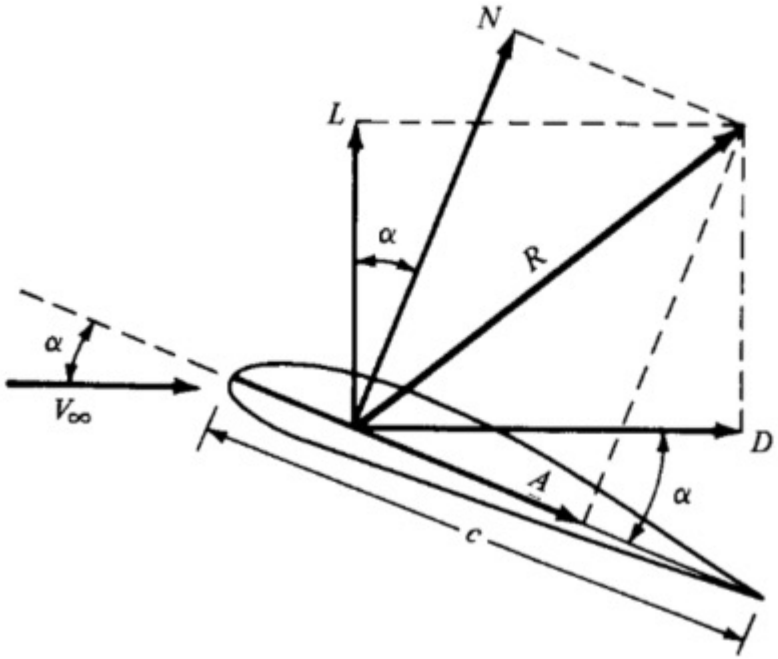
1st digit - max. camber in 100ths of chord.
2nd digit - location of max. camber along chord from LE in 10ths of chord.
Last 2 digits - max. thickness in 100ths of chord.

Forces and moments acting on an airfoil



$p = p(s)$ = surface pressure distribution
$\tau = \tau(s)$ = surface shear stress distribution

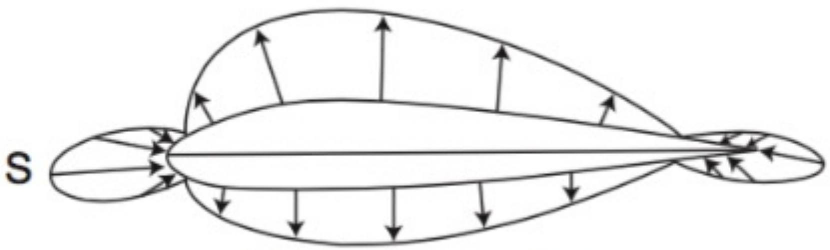




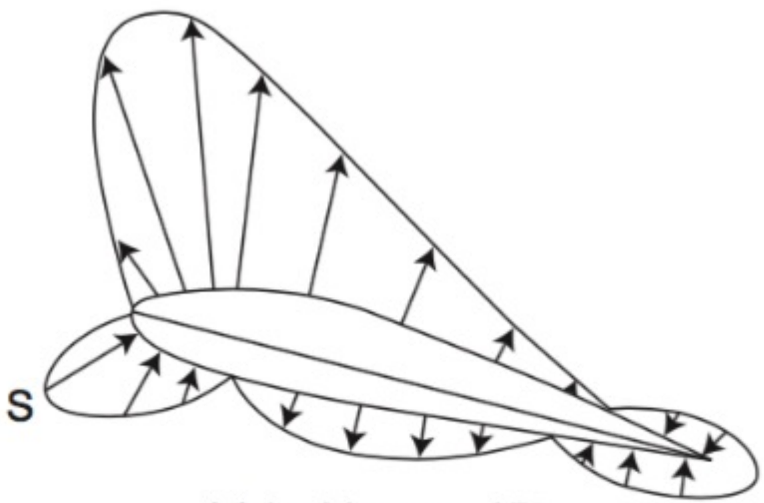
$$L = N\cos\alpha - A\sin\alpha$$

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

Pressure and skin-friction coefficients



(a) Incidence = 0°



(c) Incidence = 15°

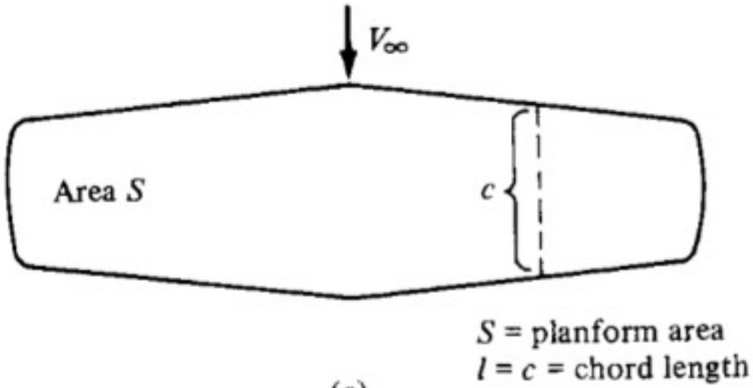
Pressure coefficient

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

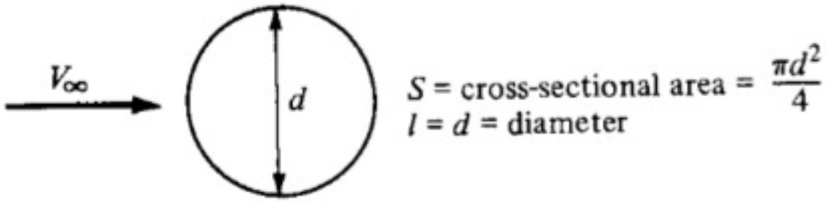
Skin-friction coefficient

$$C_f = \frac{\tau}{q_\infty}$$

Force and moment coefficients



(a)



(b)

$$C_F = \frac{F}{q_\infty S}$$

$$C_M = \frac{M}{q_\infty S c}$$

$$q_\infty = \frac{1}{2}\rho_\infty V_\infty^2$$

Estimating aerodynamic coefficients

$$C_n = \frac{1}{c}\left[\int_0^c (C_{p,l} - C_{p,u})dx + \int_0^c \left(c_{f,u}\frac{dy_u}{dx} + c_{f,l}\frac{dy}{dx}\right)dx\right]$$

$$\approx \frac{1}{c}\left[\int_0^c (C_{p,l} - C_{p,u})dx\right]$$

$$C_a = \frac{1}{c}\left[\int_0^c \left(C_{p,u}\frac{dy_u}{dx} - C_{p,l}\frac{dy_l}{dx}\right)dx + \int_0^c (C_{f,u} + C_{f,l})\,dx\right]$$

$$\approx \frac{1}{c}\left[\int_0^c \left(C_{p,u}\frac{dy_u}{dx} - C_{p,l}\frac{dy_l}{dx}\right)dx\right] ???$$

Lift and drag coefficients

$$C_l = C_n \cos\alpha - C_a \sin\alpha$$

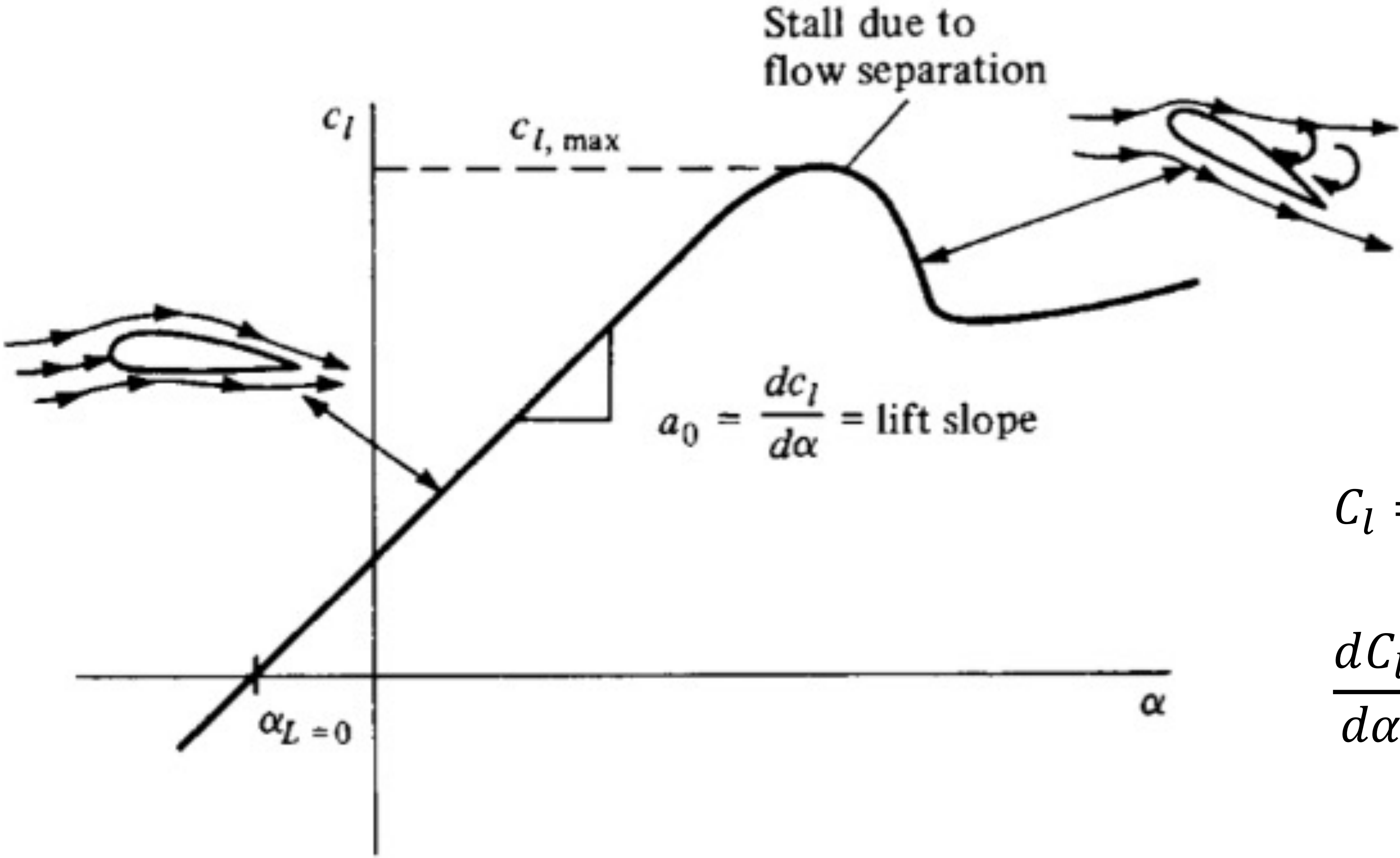
$$C_{d,p} = C_n \sin\alpha + C_a \cos\alpha$$

Moment coefficient about leading edge

$$C_{m_{LE}} = \frac{1}{c^2}\left[\int_0^c (C_{p,u} - C_{p,l})x\,dx + \int_0^c \left(y_u C_{p,u}\frac{dy_u}{dx} - y_l C_{p,l}\frac{dy_l}{dx}\right)dx\right]$$

Classical thin airfoil theory

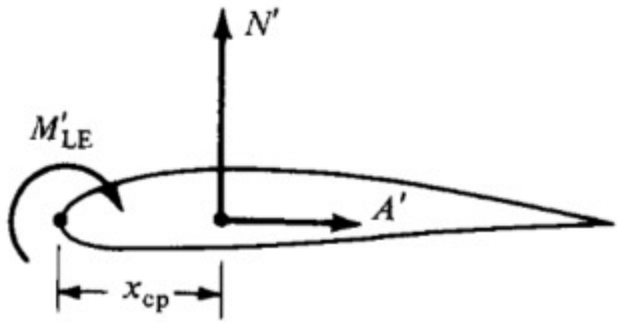
Cambered airfoil



$$C_l = 2\pi(\alpha + \alpha_{L=0})$$

$$\frac{dC_l}{d\alpha} = 2\pi$$

Center of pressure (x_{cp})



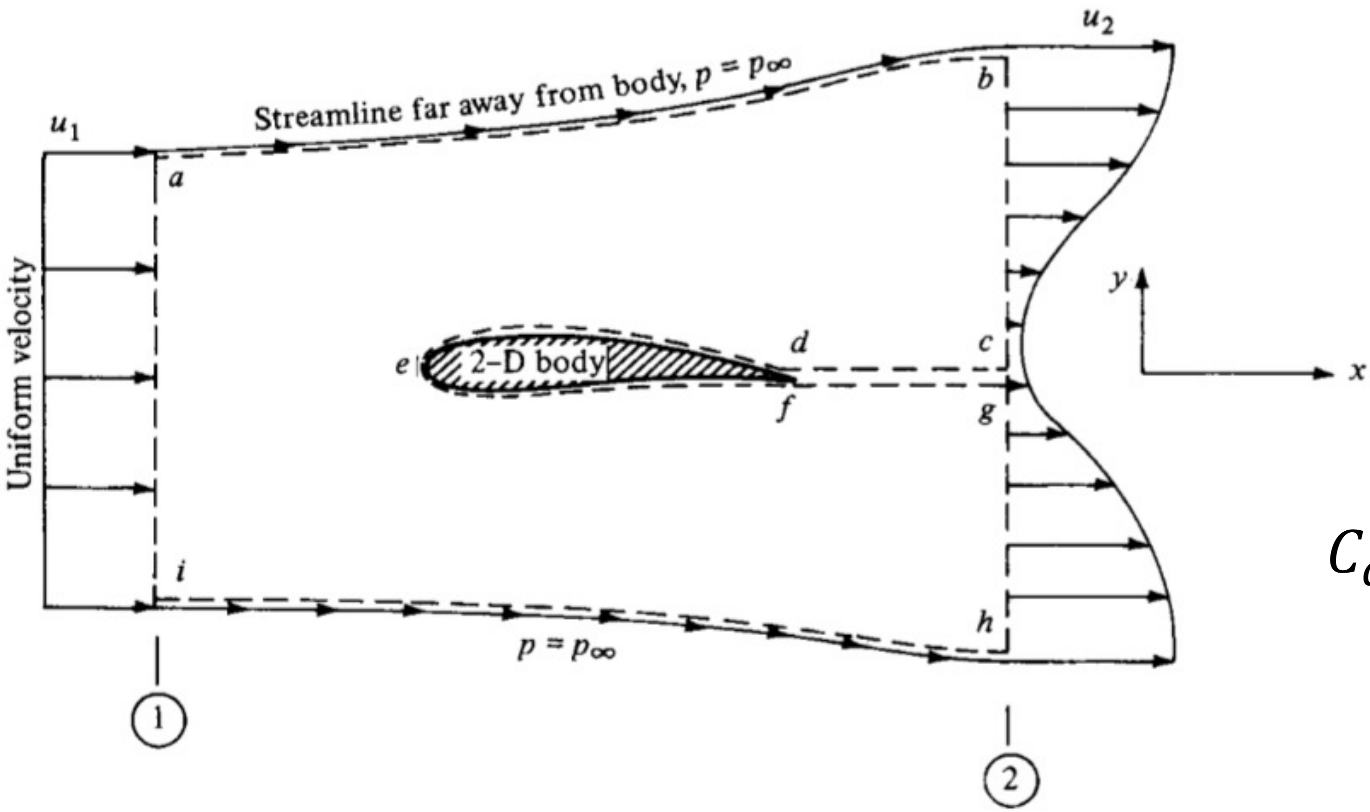
$$M'_{LE} = -\frac{c}{4}N' + M_{\frac{c}{4}} = -x_{cp}N'$$

Aerodynamic center (x_{ac})

$$\frac{dC_l}{d\alpha} = a_0 \qquad \frac{dC_{m,c/4}}{d\alpha} = m_0$$

$$\frac{x_{ac}}{c} = \frac{-m_0}{a_0} + 0.25$$

Estimating total drag coefficient from velocity measurements



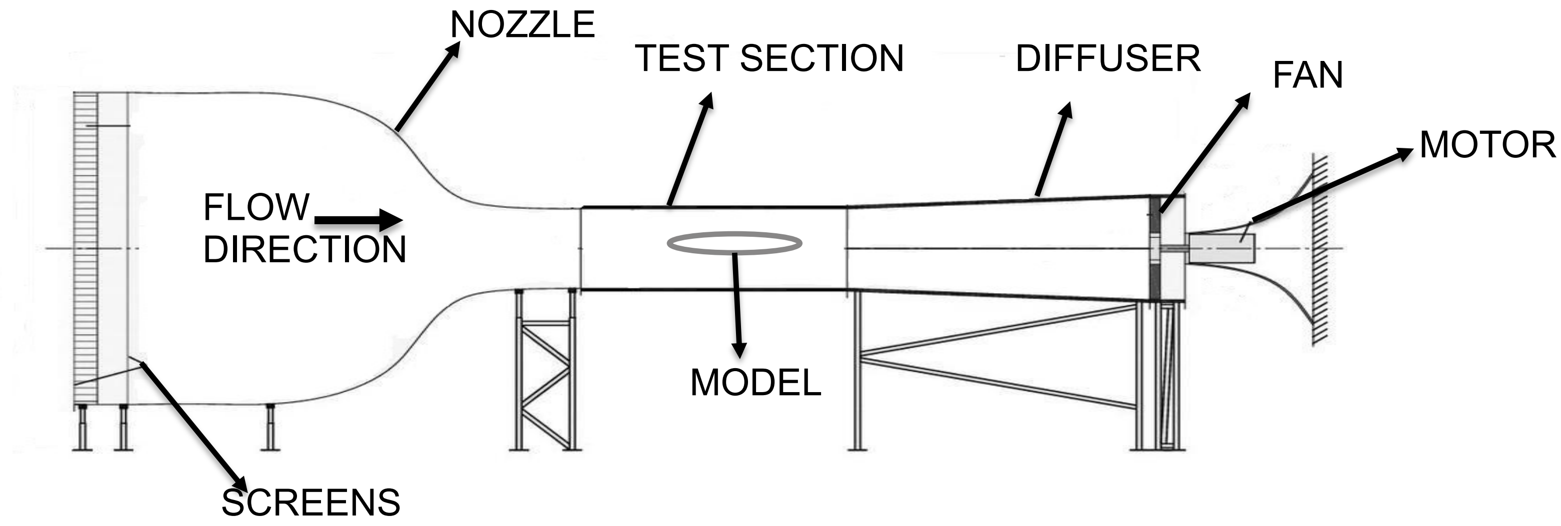
$$C_d = \frac{2}{c}\int_h^b \frac{V}{V_\infty}\left(1 - \frac{V}{V_\infty}\right) dy$$

$C_{d,f}$ (skin friction drag coefficient) obtained as difference of C_d and $C_{d,p}$

Experimental setup

Suction type wind tunnel, airfoil, pressure transducer, spirit level

Area ratio to accelerate the flow



Experimental procedure

Study and note geometrical features of the airfoil (chord c, thickness, camber, position of max. camber/thickness etc.) along with the experimental set-up (check uploaded video)

Note ambient conditions (p_0, T_0). Set required test section (free-stream) velocity V_∞ based on contraction of wind tunnel and differential pressure reading obtained using digital manometer between start of test section and reservoir.

Check the test matrix for pressure measurements and wake data; i.e. freestream velocity V_∞ and angle of attack α (eg: $\alpha =$ -6 to + 10 degrees in steps of 2 degrees and $V_\infty = 20$ m/s). Check with TA for the particular specification

Based on (differential) pressure data obtained from the manometer, C_p to be calculated versus x/c for all points on the top and bottom surface of the airfoil in the test matrix $(C_{p,u}, C_{p,l})$

Experimental procedure

C_p to be integrated to obtain C_l, $C_{d,p}$ (pressure drag coefficient) for all points in the test matrix. Do not use small angle approximations for this integration

Similarly $C_{m,LE}$ (about LE), $C_{m,c/4}$ (about $c/4$) should be computed from C_p for all the points

Get X_{cp} based on C_n and $C_{m,LE}$ for all points in test matrix

Obtain velocity profile at wake from total and static pressure data for all points in test matrix. Pitot-static probe to be used for obtaining wake velocity data. If a pitot probe is used, static pressure can be approximately assumed to be constant along the vertical in the wake for calculating velocity profile

C_d (total) to be obtained for all points in test matrix by integrating loss in momentum in the wake

$C_{d,f}$ (skin friction drag coefficient) obtained as difference of C_d and $C_{d,p}$

Report writing

Discuss the experimental setup, airfoil geometry

Report ambient data and test section velocity

Present representative C_p plots and wake profile (normalized appropriately) for some points in the test matrix (eg: one negative, 0 and one positive angle of attack)

Explain methodology and expressions for calculation of C_l, C_d, C_m, X_{cp} based on test data.

Plot the following:

1. C_l versus α for the specified velocity. Obtain lift curve slope. Plot theoretical value from thin airfoil theory on the same plot and compare. Also compare with known data in the literature for the airfoil.
2. $C_d, C_{d,p}, C_{d,f}$ versus α for all the points in the test matrix at the specified velocity. Get parametric dependence of various drag coefficients with angle of attack. Compare with expected theoretical values and literature on the airfoil (include in plots)
3. $C_{m,LE}, C_{m,c/4}$ versus α for all the points in the test matrix at the specified velocity. Obtain the moment slope. Plot theoretical value from thin airfoil theory on the same plot and compare
4. X_{cp} versus α for all the points in the test matrix at the specified velocity.

Report writing

Estimate the location of the aerodynamic center of the airfoil from the slope of the lift and moment curves. Compare it with theoretical expectation from thin airfoil theory

Include error analysis in all your computed results to correct significant digits. Error analysis means error in your measurements, not the discrepancy between theory and experiment.

In case of mismatch with theoretical values, include reasons for the observed discrepancy