ASSIGNMENT 1: Airfoil characteristics using different analysis methods

Assignment format

A PDF file report of at max **1500 words** (3 pages of text, see how to find the word count in Overleaf or Word). There is otherwise not a page limit, but you should *only* include content that you deem relevant to explain your findings. You should also fulfill the requirements specified below.
For each group, only **one** person should upload the final report.
The **deadline** for the report can be found on DTU Learn.

Report Objectives

The assignment aims to assess the advantages and limitations of using different analysis and design tools for airfoils. The tools to be applied are the thin airfoil theory, a panel method, and viscous-inviscid interaction using the open-source software Xfoil.

When you are handed the assignment, you are not expected to be able to solve all the tasks yet.

The report should accomplish the following:

1. In the same figure plot and compare the geometries of the NACA 2312 and 2324 airfoils, and the NACA 4412 and 4424 airfoils, both including also the chamber line.
   **Remember to use axis equal to ensure that the x and y axes have a 1:1 aspect.

2. Evaluate and compare the lift coefficient using the following five different methods:
   a. Thin airfoil theory
   b. Panel Method
   c. Xfoil (See the settings below)
      i. With free transition BL
      ii. With fixed transition BL

Here you should present a plot of the lift coefficient vs angle of attack, between -10 and 15 degrees, utilizing the 4 different methods/settings for the above 4 airfoils (one plot per airfoil). Make sure that in each plot the lines are readable as they can overlap.

Discuss what causes the differences between the different methods and check if and why some of those differences are more pronounced for some of the airfoils.

3. Evaluate and compare the pressure difference distribution ΔC_p as function of x/c, where $\Delta C_p \equiv \dfrac{p_{upper} - p_{lower}}{½\rho U_0^2}$, for AoA = 10° using the following five different methods:
   a. Thin airfoil theory
   b. Panel Method
   c. Xfoil (See the settings below)
      i. With free transition BL
      ii. With fixed transition BL

Here you should present a plot of pressure coefficient vs. x/c for utilizing the 4 different methods/settings for each of the for the 4 airfoils (1 plot per airfoil). Make sure that in each plot the lines are readable as they can overlap.

Discuss what causes the differences between the different methods and check if and why some of those differences are more pronounced for one of the airfoils.

4. Do the same as in the above question for dimensionless pressure distribution, $C_p \equiv \dfrac{p - p_0}{½\rho U_0^2}$, i.e. plot C_p as function of x/c, except for thin airfoil theory, where the pressure distribution is not known.

5. For the Xfoil results of question 2, save the drag coefficient and make the polar plot (Cl vs Cd).

Here you should present one figure for each airfoil, and each plot should contain a curve for each boundary layer transition method.

Discuss what causes the differences when using different transition models. Check and explain any differences in the trends between the two airfoils.

From the results obtained, fill in the following table:

Case	**Maximum Cl/Cd**	**The AoA at which Max Cl/Cd occurs**
NACA 2312 with free transition		
NACA 2312 with fixed transition		
NACA 2324 with free transition		
NACA 2324 with fixed transition		
NACA 4412 with free transition		
NACA 4412 with fixed transition		
NACA 4424 with free transition		
NACA 4424 with fixed transition		

Discuss the results from the table.

6. Discuss what would happen if the Reynolds number was significantly lower, for example, one order of magnitude lower. Would the differences between Xfoil and the other

methods be higher or lower? Would you expect the performance of the airfoil to improve? Which airfoil is more sensible to changes in the Reynolds number?
Do the same for different transition criteria by changing the N-parameter and the location of fixed transition.
TIP: You can simply run Xfoil with a different Reynolds number and/or transition criteria and see what happens.

Required figures and tables

- Plots comparing the airfoil contour/shape comparing the NACA 2312 and 2324 airfoils, and the NACA 4412 and 4424 airfoils, respectively. Remember to include the chamber lines and to use axis equal to ensure that the x and y axis have a 1:1 aspect ratio.
- Lift coefficient vs. Angle of Attack for the 4 different methods/settings, one plot for each of the four airfoils.
- Pressure difference coefficient vs. x/c for the 4 different methods/settings, one plot for each of the four airfoils.
- Pressure coefficient vs. x/c for 3 different methods/settings (all except thin airfoil theory), one plot for each of the four airfoils.
- Cl vs Cd for all the BL transition cases, one plot for each of the four airfoils.
- Fill out the table of question 5.

Xfoil Settings

When using Xfoil you should apply the following settings

- Reynolds number: $1.5*10^6$ (for all evaluations)
- Mach number: 0.0 (for all evaluations)

When evaluating the different Boundary Layer (BL) models assume free transition (Ncrit = 9) for the lower side of the airfoil, and the following transition criteria for the upper side:

1. Free transition (Ncrit = 9)
2. Fixed transition with $X_t = 0.1$

Background

Airfoil parametrization

There are different ways of defining and designing airfoils, and often they are categorized in series or families depending on their purpose and use.

The NACA airfoil families

One of the most popular airfoil series for aircraft is the NACA family. The early NACA airfoil series, the 4-digit, 5-digit, and modified 4-/5-digit, were generated using analytical equations that describe the camber (curvature) of the mean-line (geometric centerline) of the airfoil section as well as the section's thickness distribution along the length of the airfoil. Later families, including the 6-Series, are more complicated shapes derived using theoretical rather than geometrical methods.
The method to generate the geometry is illustrated in Fig. 1. The leading and trailing edges are defined as the forward and rearward extremities, respectively, and the chord line is defined as the straight line connecting them.

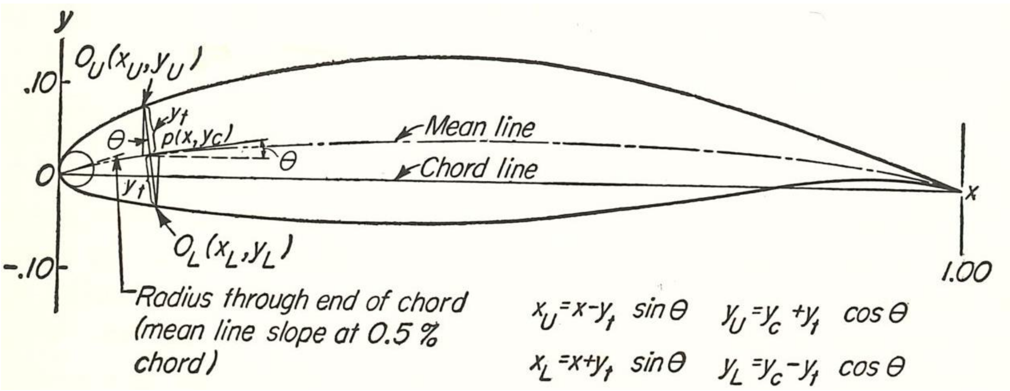

Figure 1. Definition of the geometry of NACA airfoil.
Courtesy: Abbott and Doenhoff: 'Theory of wing sections', Dover publications, 1958).

The geometry is designed by first defining the ordinates of the camber line, y_c, which is given as the mean line of the airfoil. Next, the coordinates of the upper and lower surface of the airfoil are given by an analytical expression defining the thickness distribution. The location of the coordinates are obtained by tacking the half thickness, y_t, and plotting it perpendicular to the camber line in both directions, with the upper one defining the upper surface and the lower one the lower surface, respectively. Let x_U and y_U represent the abscissa and ordinate, respectively, of a typical point at the upper surface, the upper surface is defined by the following relationship:

$$x_U = x - y_t \sin\theta$$
$$y_U = y_c + y_t \cos\theta$$

The corresponding expressions for the lower surface coordinates are

$$x_L = x + y_t \sin\theta$$
$$y_L = y_c - y_t \cos\theta$$

where $$\theta = \tan^{-1}\left(\frac{dy_c}{dx}\right).$$

NACA four-digit airfoils

The camber line for a four-digit NACA airfoil is given by:

$$y_c(x) = \begin{cases} \dfrac{m}{p^2}\left(2p\dfrac{x}{c} - \left(\dfrac{x}{c}\right)^2\right), & 0 \le x \le p \\ \dfrac{m}{(1-p)^2}\left(1 - 2p + 2p\dfrac{x}{c} - \left(\dfrac{x}{c}\right)^2\right), & p < x \le 1 \end{cases}$$

where:

- m is the maximum camber, measured in percentage of the chord
- p= is the location of the maximum camber, measured in percentage of the chord

A specific four-digit airfoil is referred to as NACA mpxx, where xx denotes the relative thickness of the airfoil in percentage of the chord length. Hence, as an example, a NACA 2312 airfoil has a maximum camber of 2%, which is located 30% away from the leading edge, and a thickness of 12%.

The thickness distribution of a four-digit NACA airfoil is described using the following equation:

$$y_t = 5t\left(0.2969\sqrt{x} - 0.1260x - 0.3516x^2 + 0.2843x^3 - 0.1015x^4\right),$$

where:

- t is the maximum thickness of the airfoil, measured in percentage of the chord
- x is the position along the chord (ranging from 0 at the leading edge to 1 at the trailing edge)
- y_t is the half-thickness, measured perpendicular from the mean camber line at a given point x