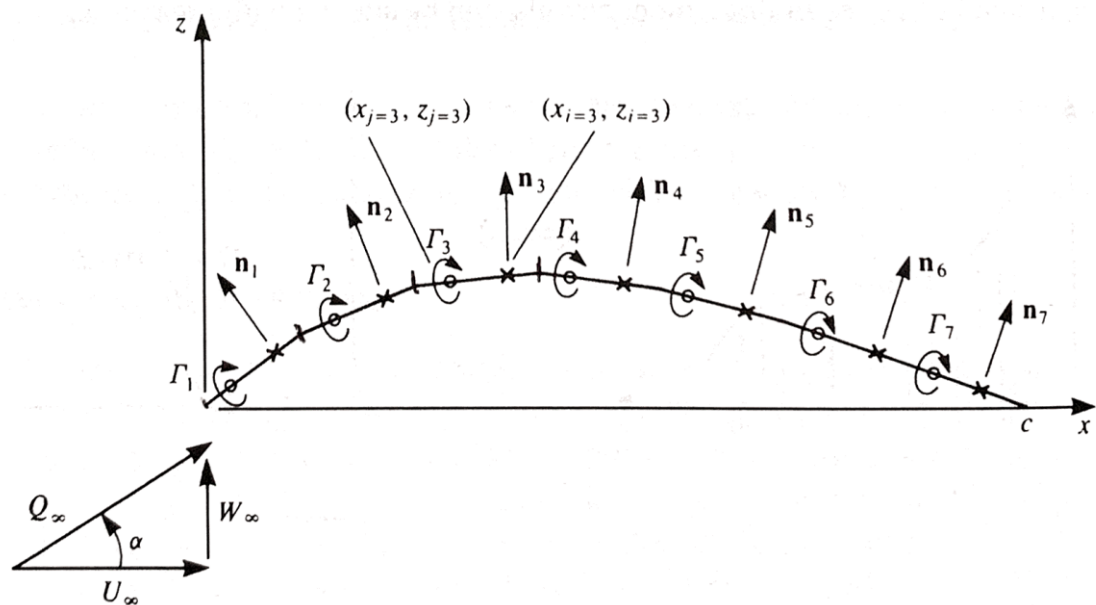


# Assignment 1: Potential flow over a thin airfoil

Academic year: 2024-2025

This document is a **fillable form**. Use it to deliver the assignment as part of the final report that contains all 5 assignments.

|                 |  |
|-----------------|--|
| Full Name:      |  |
| Student Number: |  |
| Date:           |  |



## Requirements

- Personal / group task: personal or group (max. 2 persons)
- **Handwritten drawings** must contain a visible **date and signature** to its side
- Be as complete as necessary, as succinct as possible

## Introduction

For the analysis of airfoils, it is essential to obtain the velocity and the pressure distribution over the surface. Based on this the lift and the pitching moment can be

obtained quite accurately using a potential flow approach for low angles of attack. The drag remains zero due to the absence of viscosity. The drag of the airfoil can be determined if also the development of the boundary is calculated. This requires an interaction between the inviscid (potential flow) and viscous (boundary layer) calculation which is outside the scope of this assignment.

To get a basic understanding of the capabilities and limitations of potential flow calculations on airfoils, this assignment is aimed at developing a very simple flat panel model that can predict the characteristics of thin airfoils. In this assignment you are requested to write a program that calculates the potential flow over 2-dimensional airfoils with zero thickness (according to the so-called “**thin airfoil theory**”).

## Preparation

Write a clear and concise program that, based on panel methods, can calculate the pressure distribution ( $\Delta C_p$  vs.  $x$ ) and the lift coefficient,  $C_l$ , of **an airfoil with zero thickness** following the panel method described in Katz and Plotkin, “Low-Speed Aerodynamics” section 9.8. You may use the programming language/environment of your preference (MATLAB, Python or Fortran).

Feel free to apply **Artificial Intelligence** (AI) like ChatGPT to write your program (make sure you check the correct working of the program). In any case make sure to add a flow diagram as well as enough comment lines to describe the working principle. Add a listing at the end of the assignment.

The code, should be able to compute the pressure distribution over the camber line of a 4-digits NACA airfoil, given by:

$$z_c(x) = \begin{cases} \frac{m}{p^2}(2px - x^2), & 0 \leq x \leq p \\ \frac{m}{(1-p)^2}(1 - 2p + 2px - x^2), & p \leq x \leq 1 \end{cases}$$

Where  $p$  is the location of maximum camber and  $m$  the maximum camber, with respect to the camber line. The first digit of a NACA 4 digits profile represents maximum camber,  $m$ , times 100 while the second one represents the location of the maximum camber,  $p$ , times 10.

For Task 2, you will need a **reference airfoil**. You can get your reference airfoil by adding up the three last digits of your student number. The table below describes which airfoil you should be used as the reference:

| Sum of digits | Airfoil   | Sum of digits | Airfoil   | Sum of digits | Airfoil   |
|---------------|-----------|---------------|-----------|---------------|-----------|
| 0-1           | NACA 2424 | 10-11         | NACA 2412 | 20-21         | NACA 4412 |

|     |           |       |           |       |           |
|-----|-----------|-------|-----------|-------|-----------|
| 2-3 | NACA 2415 | 12-13 | NACA 1412 | 22-23 | NACA 1410 |
| 4-5 | NACA 4415 | 14-15 | NACA 4418 | 24-25 | NACA 2418 |
| 6-7 | NACA 2408 | 16-17 | NACA 2421 | 26-27 | NACA 2410 |
| 8-9 | NACA 4421 | 18-19 | NACA 1408 |       |           |

Experimental data from these airfoils are available on Abbott, “Theory of wing sections, including a summary of airfoil data” (see document provided via Brightspace).

XFOIL is a panel method solver available in:

<https://web.mit.edu/drela/Public/web/xfoil/>.

You will be required to use its **inviscid solution** to complete the assignment Task 2.

For the details on the theory please refer to standard textbooks like:

- J. Katz and A. Plotkin, “Low-Speed Aerodynamics” (Cambridge Aerospace Series) (section 9.8)
- J.J. Bertin, “Aerodynamics for Engineers”
- A.M. Kuethe and C. Chow, “Foundations of Aerodynamics: Bases of Aerodynamic Design”

## Task 1 – Theoretical background

Once the program has been established perform the following tasks:

1. In the **field underneath** paste a picture with a flow diagram that describes your solver and the essential steps taken from the input geometry of the camber line and angle of attack to the pressure distributions and lift coefficient.

2. Describe the theory that is being implemented along with the equations being solved. First, please input handwritten figures with the respective equations and comments of the following:

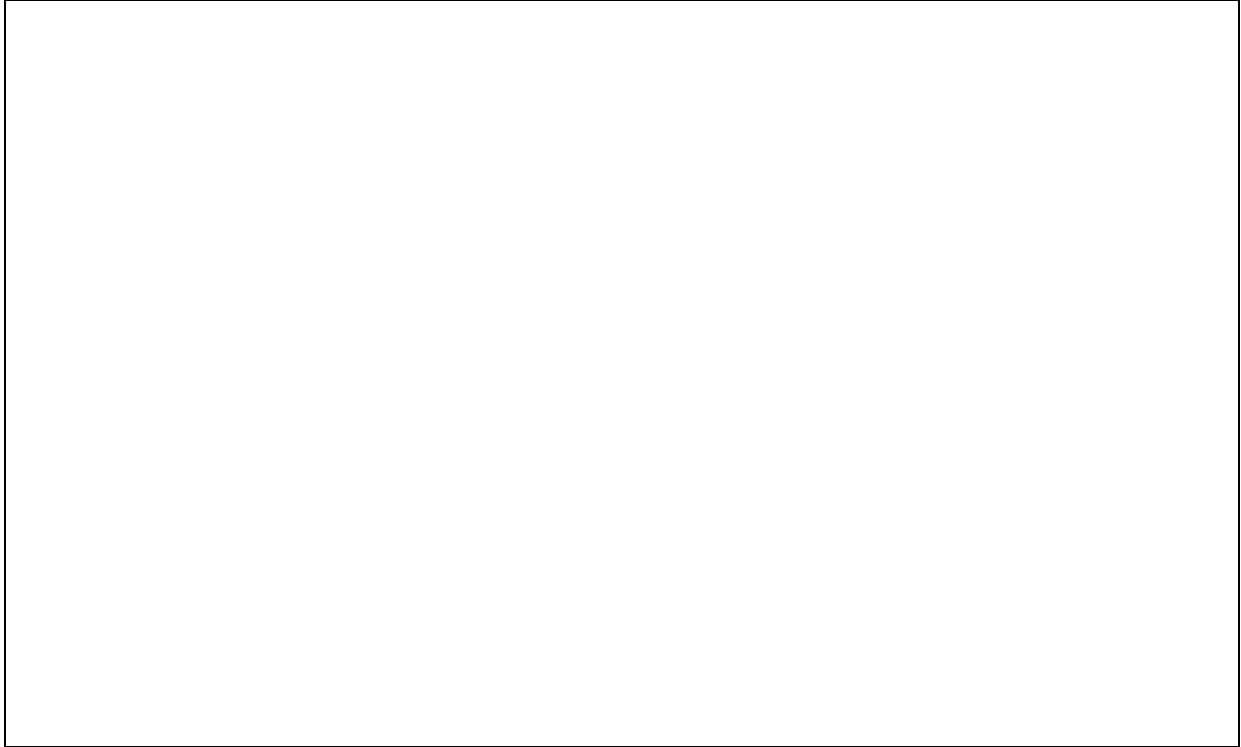
- a. Describe the induced velocity at a point in space  $(x, z)$  due to the presence of a point vortex of strength  $\Gamma_i$  at location  $(x_i, z_i)$ .

- b. Write the non-penetration condition for this point assuming a uniform inflow in the x direction  $(U_\infty, 0)$  and  $N$  vortices.

- c. Describe in the box below the so called “Kutta condition” and necessary condition imposed to  $\Gamma_i$  at the trailing edge.

- d. Describe the equations that provide the pressure coefficient ( $\Delta C_p$ ), the lift ( $C_l$ ) and pitching moment coefficients ( $C_{m_{\frac{1}{4}c}}$ ) of the airfoil at the one-quarter chord)

3. Use the code to compute the derivative of the lift gradient,  $\frac{dC_l}{d\alpha}$ , of an airfoil:
- Do so for different number of panels around the airfoil.
  - In the box below, plot the obtained derivative as a function of number of elements including the expected theoretical derivative from thin-airfoil theory.

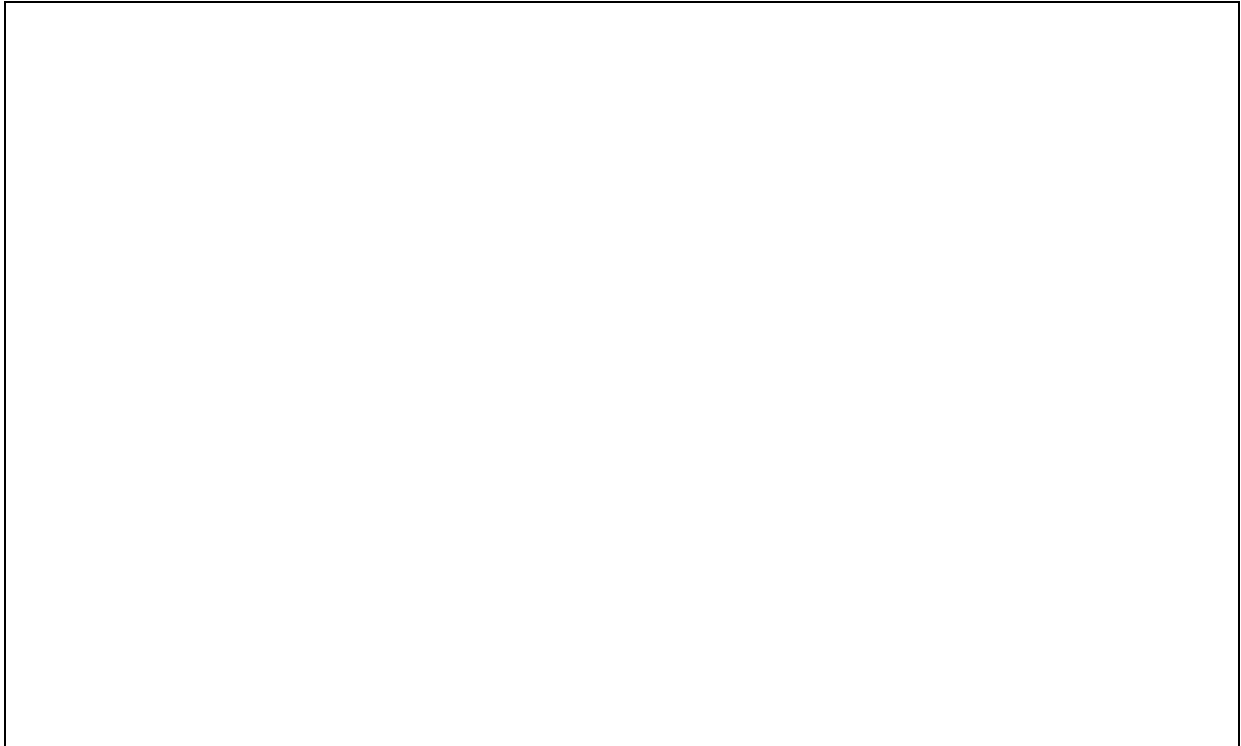


- Add a short description on the convergence rate and minimum number of panels required (max. 100 words).



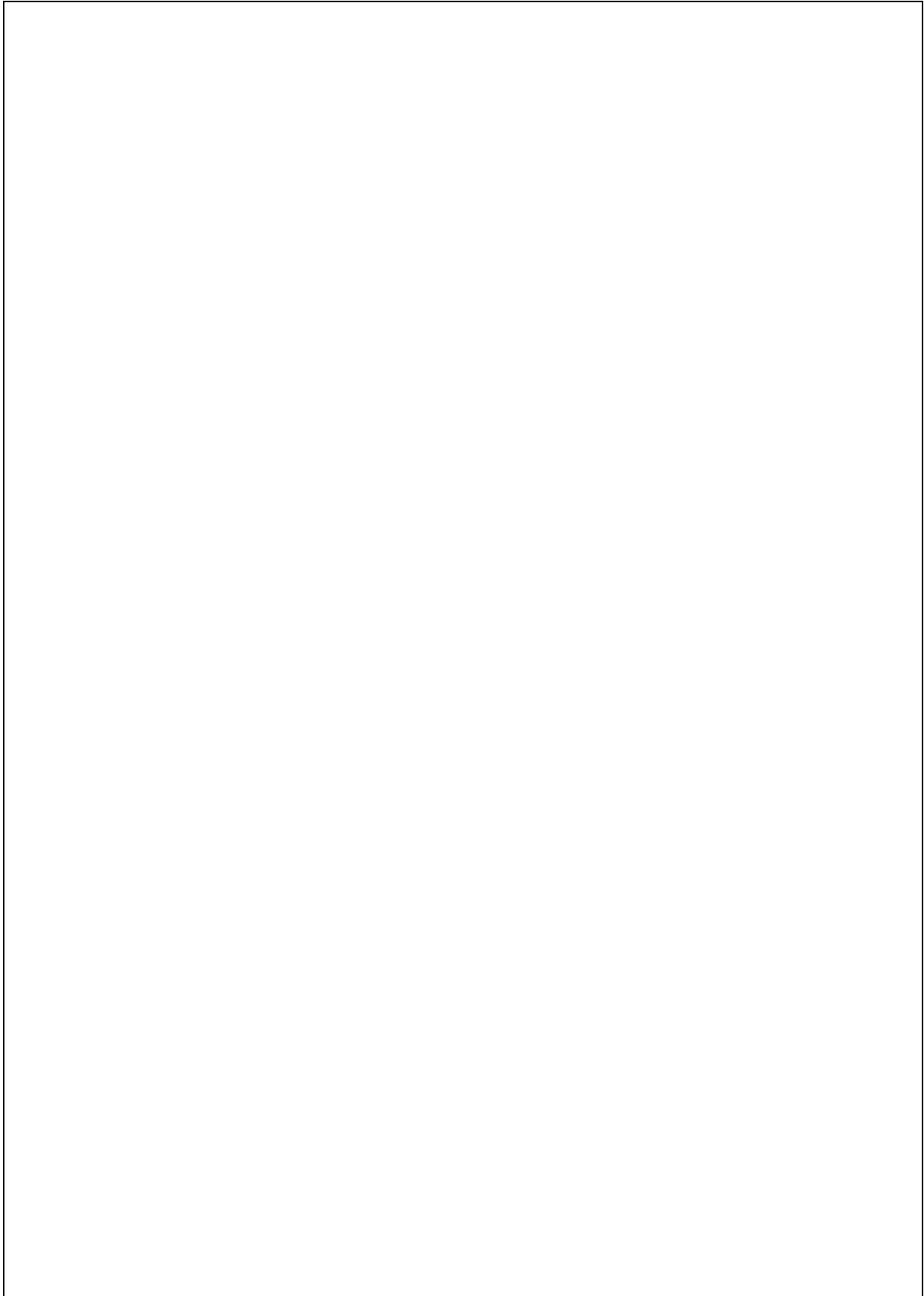
## Task 2 – Comparison of results

1. Select an airfoil according to the criteria described at the header and gather experimental data along with inviscid XFOIL predictions of its  $C_p$  distribution at zero angle of attack,  $C_l$ , and  $C_m$  versus angle of attack. With the data collected, do the following:
  - a. Input a plot comparing the pressure differential ( $\Delta C_p$ ) along the airfoil chord line obtained with your code, with xfoil, and with experiments at  $0^\circ$  angle of attack

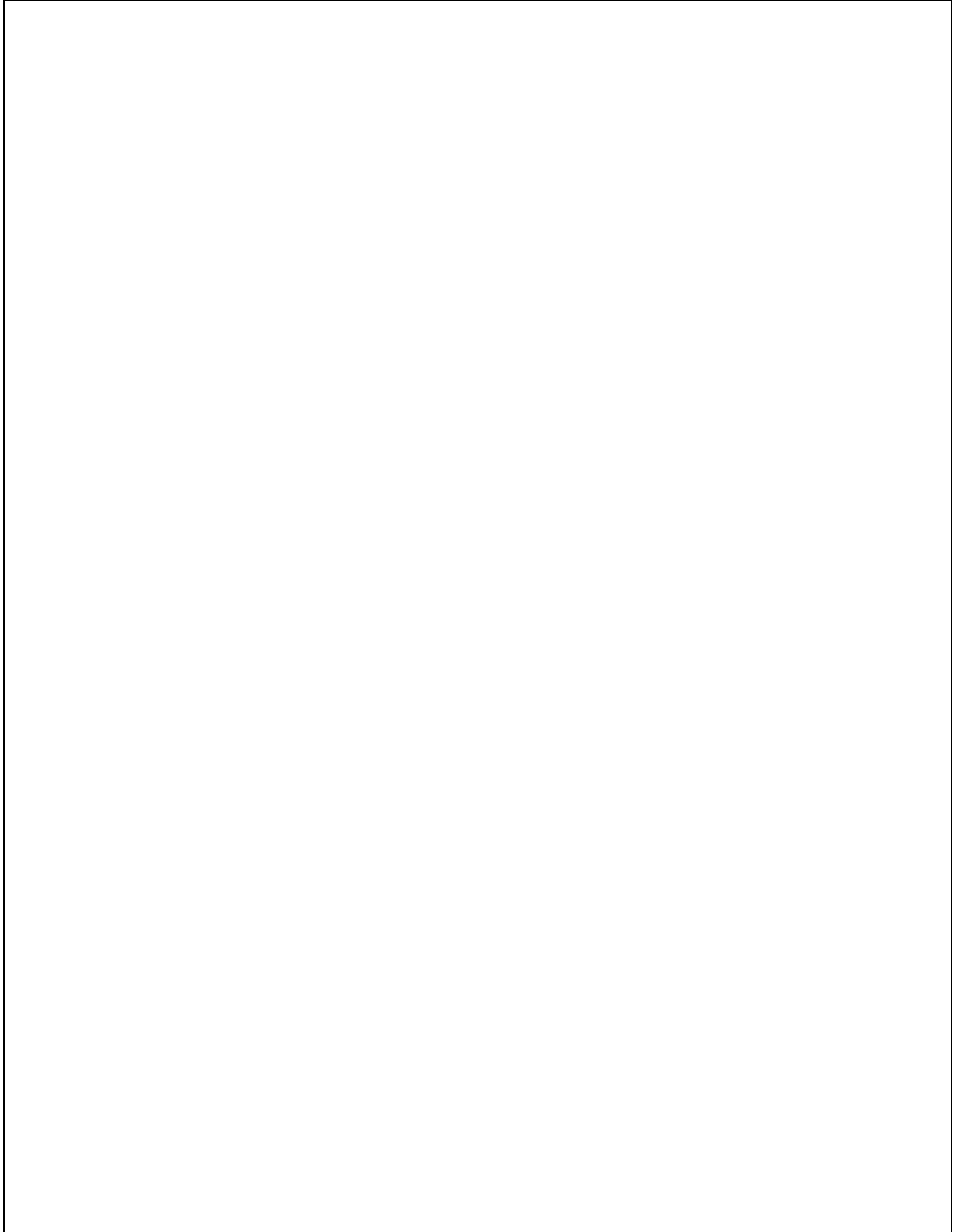




- b. Input a figure comparing  $C_l$  vs.  $\alpha$ , and  $C_m$  vs.  $\alpha$  obtained with your code, with xfoil, and with experimental data (see "Theory of Wing Sections ")



2. With your baseline airfoil defined, apply the following modifications and plot the outcomes on a graph of  $C_l$  vs.  $\alpha$ , and  $C_m$  vs.  $\alpha$ :
  - a. Modify the airfoil maximum camber
  - b. Modify the location of maximum camber.



- c. Discuss the effects of altering the maximum camber and location of maximum camber.

## Task 3 – Discussion of results

1. Briefly discuss the following findings from the previous results. Is the thin airfoil theory adequate to describe:
  - a. The maximum  $C_l$  of an airfoil?
  - b. The lift slope ( $C_l$  vs.  $\alpha$ ) of an airfoil?
  - c. The  $C_m$  of an airfoil?
  - d. The  $\alpha$  of zero-lift of an airfoil?

2. With handwritten drawings and remarks, describe the differences expected between a thin and thick airfoil:
  - a. How does the pressure distribution (or delta pressure distribution,  $\Delta C_p$ ) differs from a thin to a thick airfoil?

b. What is the influence of thickness on the lift gradient of an airfoil?

## Task 4 – Code

Please upload the code script that you have used in here. Add comments describing the functionalities and parts of the code. You can alternatively add a link to the code repository in surfdive or onedrive.