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.

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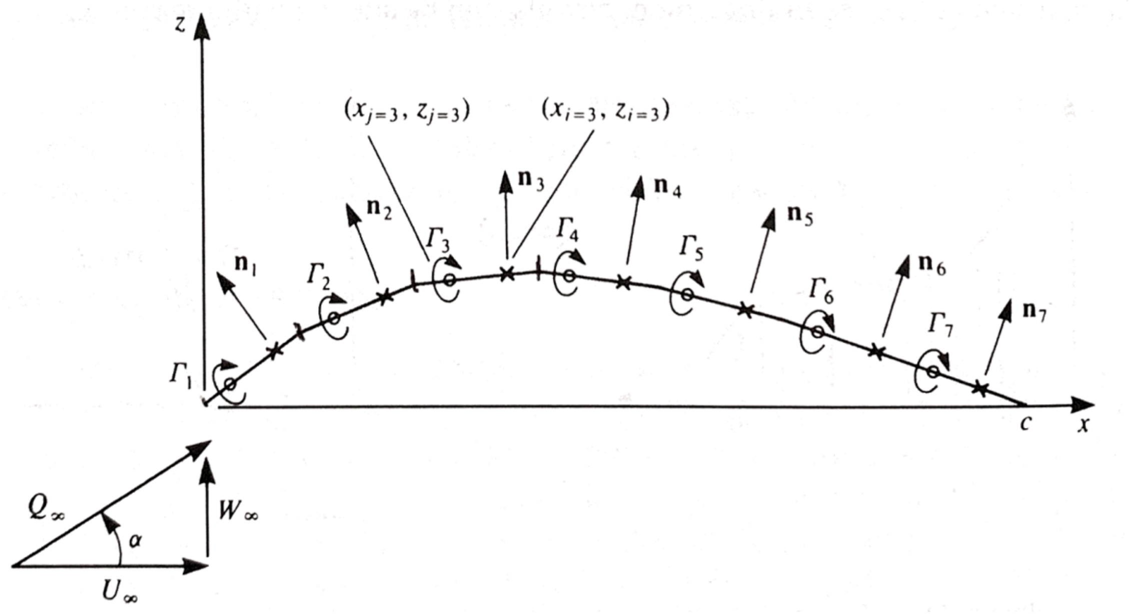
Date:	27-11-24



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 (ΔC_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:

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

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

Kilian's Student Number has been used (5310644)

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.

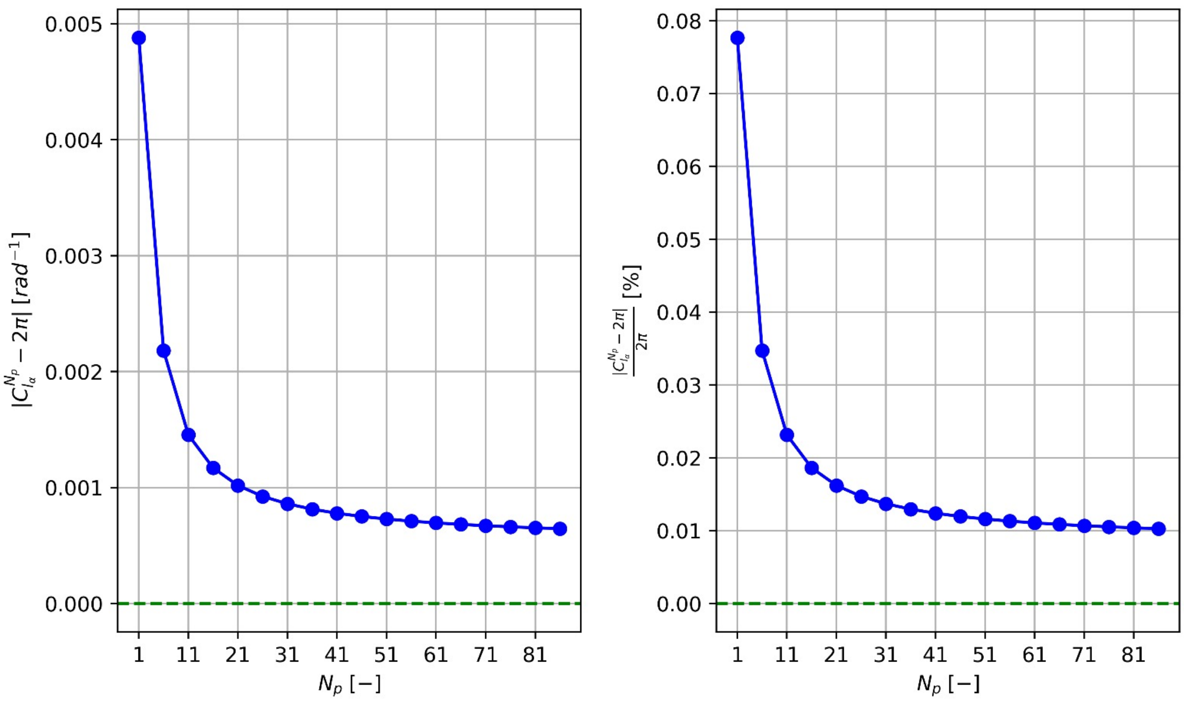
2. Describe the theory that is being implemented along with the equations being solved. First, please input handwritten figures with the respective equations and comments of the following:
    a. Describe the induced velocity at a point in space (x, z) due to the presence of a point vortex of strength $\Gamma_i$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 Γ_i at the trailing edge.

d. Describe the equations that provide the pressure coefficient $\left(\Delta C_p\right)$, the lift $(C_l)$and pitching moment coefficients $\left(C_{m_{\frac{1}{4}c}}\right)$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:
    a. Do so for different number of panels around the airfoil.
    b. In the box below, plot the obtained derivative as a function of number of elements including the expected theoretical derivative from thin-airfoil theory.

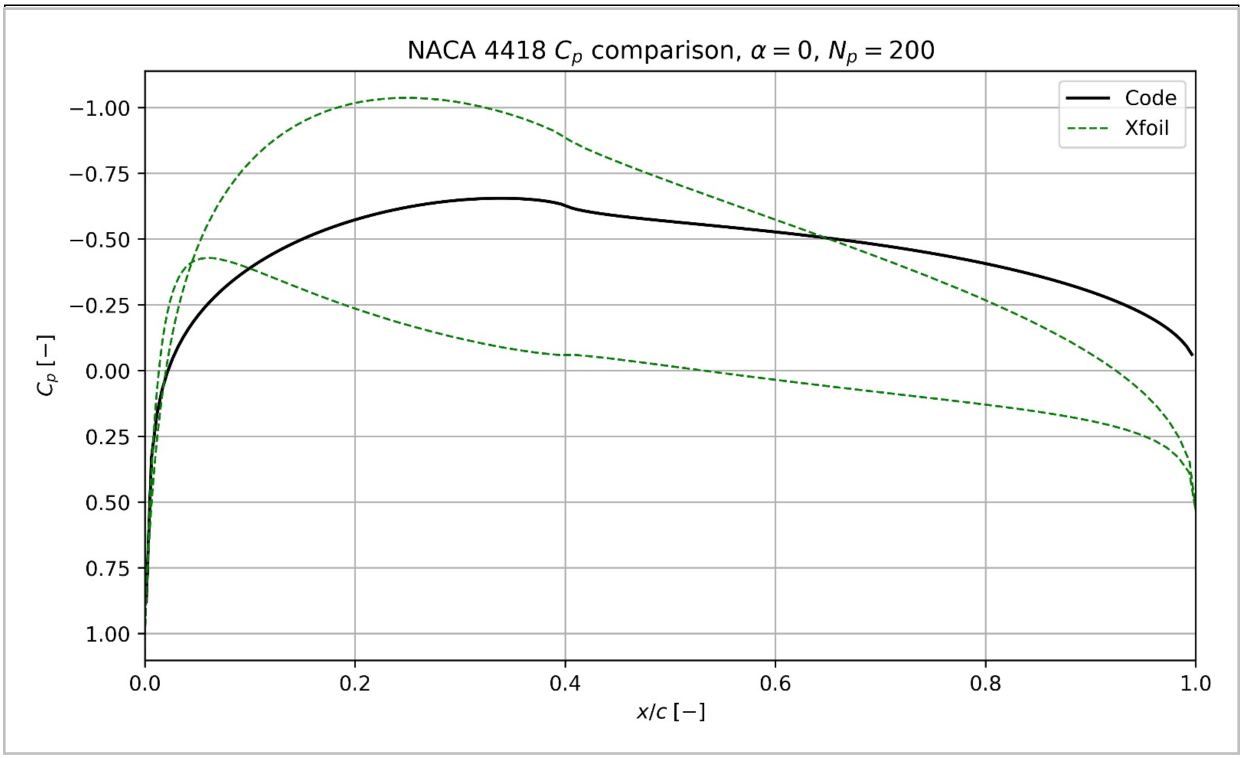


    c. Add a short description on the convergence rate and minimum number of panels required (max. 100 words).

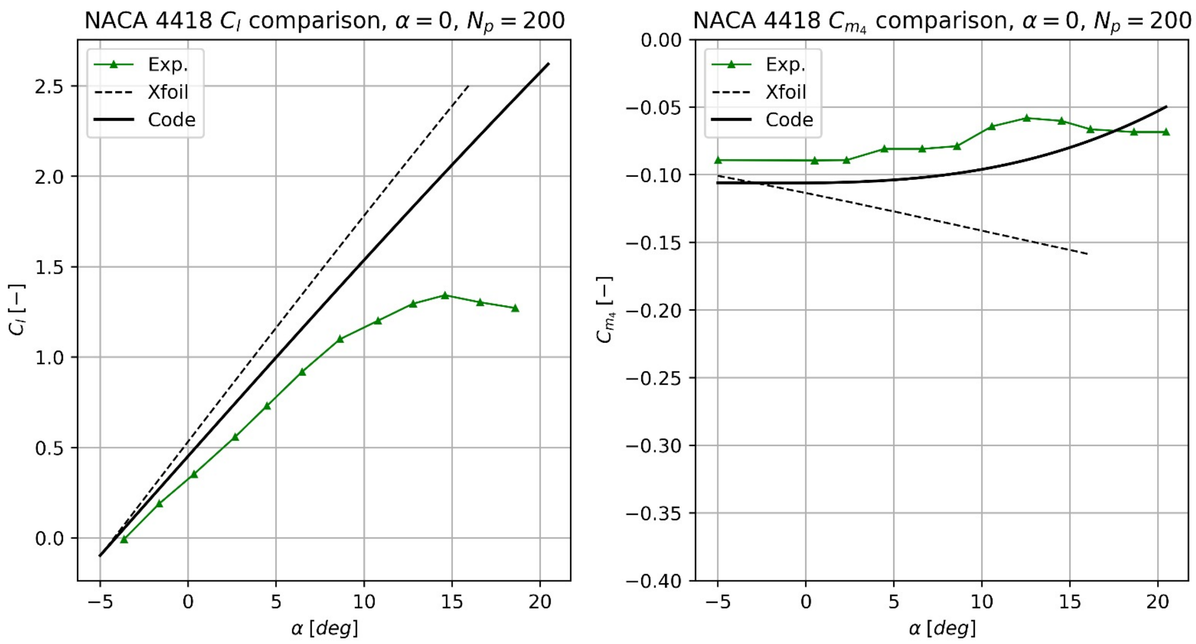
The figure above shows the lift gradient as a function of the number of panels compared to the theoretical prediction due to the thin-airfoil theory (TAT): 2pi. As can be observed, the relative error decreases exponentially (retrieved from plotting the error in log-y plot). This means the panel method converges. Furthermore, it can be noted that the error does not converge to exact zero as expected from TAT. The error seems to converge towards a value of approx. 0.001. This could be explained due to the fact that the NACA 4418 airfoil has a relatively high t/c ratio and thus doesn't agree with the thin airfoil theory. However, plotting the relative difference between numbers of panels and assuming that convergence is obtained if the relative difference between steps is 1.5%, 71 panels are shown to be enough to pass the convergence criterion.

Task 2 – Comparison of results

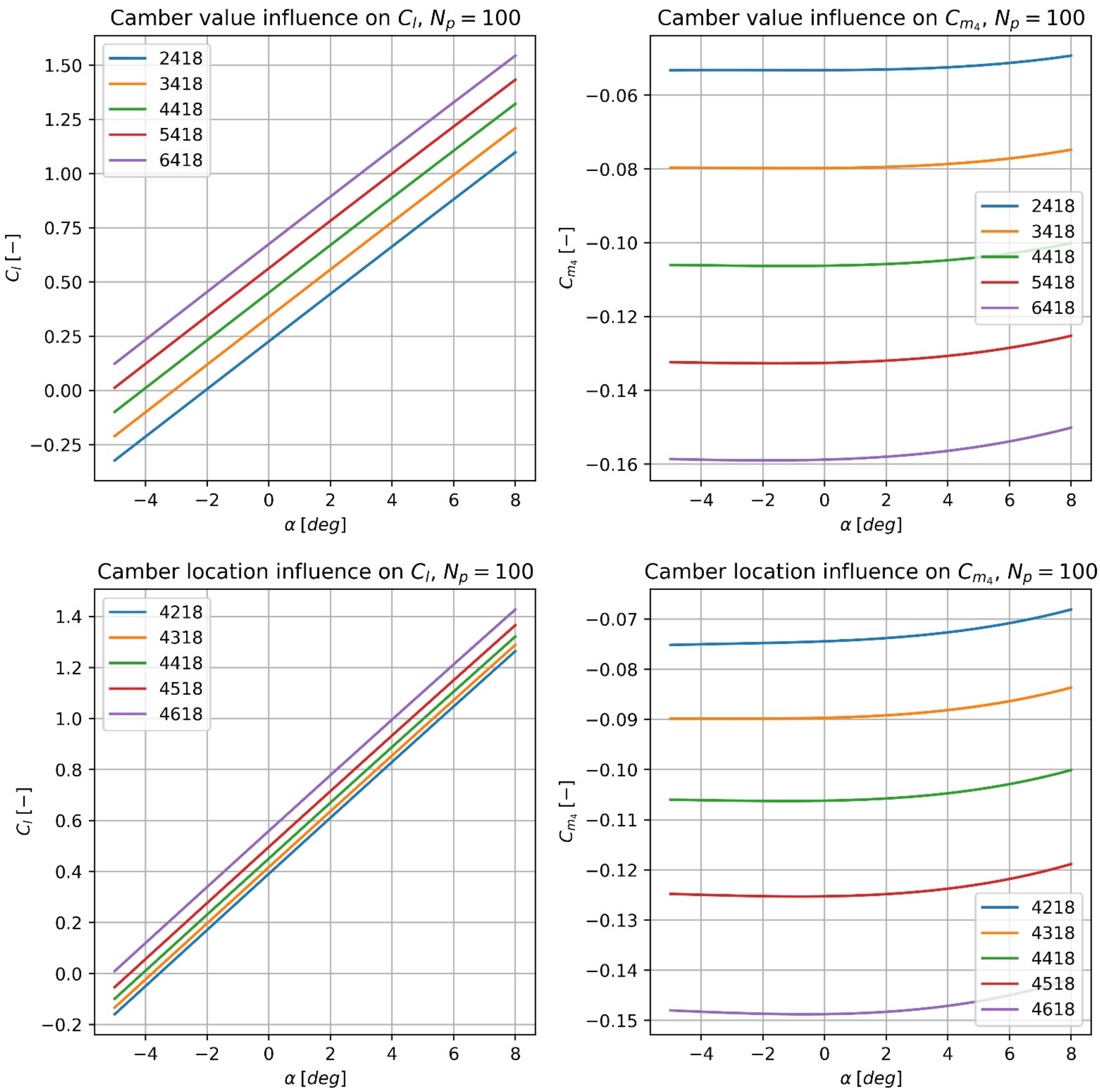
1. Select an airfoil according to the criteria described at the header and gather experimental data along with inviscid XFOIL predictions of its C_p distribution at zero angle of attack, Cl, and Cm versus angle of attack. With the data collected, do the following:
    a. Input a plot comparing the pressure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



b. Input a figure comparing C_l vs. α, and C_m vs. α 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. α, and C_m vs. α:
    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.

As can be noted from the graphs above increasing camber leads to an upwards shift in lift gradient. For the same angle of attack an airfoil with a higher camber has a higher Cl and thus can generate more lift.

Changing the location of maximum camber has the same effect, moving the maximum camber chordwise will shift the lift gradiant upwards.

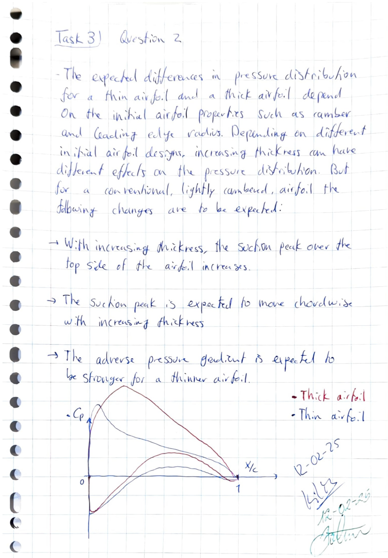
Both increasing and moving the camber chordwise have a similar effect of shifting the moment coefficent curve downwards. The moment coefficient about the quarter chord becomes increasingly negative as the maximum camber increases, and a more rearward location of the maximum camber further accentuates this negative moment.

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. α) of an airfoil?
    c. The C_m of an airfoil?
    d. The α of zero-lift of an airfoil?

Based on the findings presented above, the thin airfoil theory is not adequate for describing the maximum lift coefficient of the airfoil. It fails to capture the stall and according to thin airfoil theory, the lift keeps increasing with the angle of attack.
Comparing the lift slope (Cla) of the experimental results, XFOIL, and thin airfoil theory, it seems that only relatively small differences are present between them, likely due to the absence of the boundary layer or viscous forces. But, although with these very rough simplifications made it seems that thin airfoil theory can predict Cla relatively well.
The moment coefficient is also predicted (relatively) well compared to the experimental data, with the only a small shift downwards present. Interestingly, XFOIL seems to have failed to capture the correct Cm behaviour completely.
Finally, the zero-lift angle of attack seems to be predicted very well, with only a very small difference present between the thin airfoil approximation and the experimental data.

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, ΔC_p) differs from a thin to a thick airfoil?


Task 3) Question 2

- The expected differences in pressure distribution for a thin airfoil and a thick airfoil depend on the initial airfoil properties such as camber and leading edge radius. Depending on different initial airfoil designs, increasing thickness can have different effects on the pressure distribution. But for a conventional, lightly cambered, airfoil the following changes are to be expected:

→ With increasing thickness, the suction peak over the top side of the airfoil increases.

→ The suction peak is expected to move chordwise with increasing thickness

→ The adverse pressure gradient is expected to be stronger for a thinner airfoil.

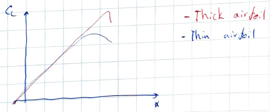
b. What is the influence of thickness on the lift gradient of an airfoil?

- Task 3| Question 3

- The effect of thickness on the lift gradient is also nuanced and depend on the initial airfoil shape. But generally, for lightly cambered airfoils, with relatively low initial thickness, increasing the thickness has the following effects.

→ The increase in thickness causes a delay in stall.

→ For a thicker airfoil the stall is more abrupt



- For different initial airfoil shapes the effect of increasing thickness might be different

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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 surfdrive or onedrive.