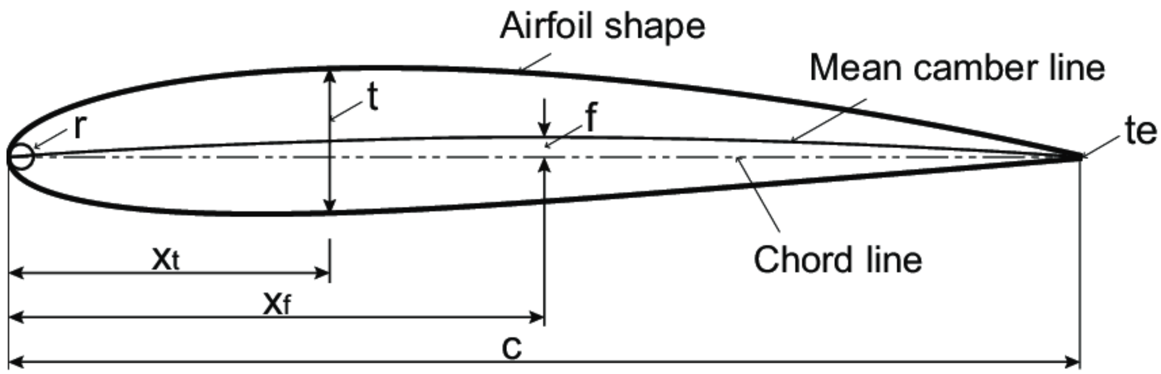
Assignment 2 Airfoil analysis and preliminary design

Academic year: 2023-2024



Requirements

- Personal or group task: personal
- Indicative minimum / maximum number of pages: 10 / 15 (excluding Title page)
- Font size: 10 – 12

Introduction

The aerodynamic characteristics and performance of any aircraft part exposed to the external flow is significantly influenced by the state of the boundary layer (BL), that can either be laminar or turbulent. To determine in how far the BL transition on a 2D airfoil is affected by its shape and the atmospheric turbulence scale, this assignment addresses the airfoil design in relation to the transition point location.

The so-called e^n-method[1] is a very good method for the estimation of BL-transition as it is fast and rather accurate. Many airfoil design and analysis programs employ this e^n-method. In this assignment you will use the program *XFOIL* to determine the effect of shape changes and the user selected amplification factor n on performance of a 2-dimensional airfoil.

1) J.L. van Ingen, "Theoretical and experimental investigations of incompressible laminar boundary layers with and without suction", Doctoral Thesis, Technische Hogeschool Delft, 1965 or H. Schliching, "Boudary Theory".

Note:
When discussing XFOIL results in your report, please do not use screen prints as they are hardly readable. Instead export the data and produce high quality plots in Matlab, Excel or any other convenient program. Note that XFOIL also allows you to quickly generate high quality postscript files (which can easily be converted to pdf-format).

Tasks

Part 1 Airfoil analysis

1. Before you start your airfoil calculations shortly explain in a separate section of the report the following concepts:

    a) The Prandtl boundary layer equations. **Write them down by hand**[2] (and add as figure(s)) and **explain what the various terms mean**.
    b) Explain the link between the so called ***first compatibility equation*** and the form of the **velocity distribution** (add hand drawings).

2. If you did not already do so download the XFOIL program and the manual from the internet (for example from: https://web.mit.edu/drela/Public/web/xfoil/) and install it in a local directory.
3. Before you start, make yourself familiar with the theoretical background of this solver

4. Determine your **personal 4-series airfoil** shape that you are going to investigate in this assignment. The procedure for this can be found in Appendix A. Please note that failure to work with the right airfoil shape will result in zero credits for Assignment #2

5. Show this calculation and its results and **clearly state what airfoil shape you use** (provide a figure with the airfoil shape as well)

6. For your **personal airfoil**, calculate the lift and drag polar between -2 and 8 deg angle of attack using the (non-default) value of the **amplification factor** $\boldsymbol{n} = \mathbf{12}$ and produce clear plots of the results (no screen prints). Use a chord-based Reynolds number of $\boldsymbol{Re} = \mathbf{0.7} \times \mathbf{10^6}$ and leave the Mach number at $\boldsymbol{M} = \mathbf{0}$ (this indicates that no compressibility correction is applied)

7. Check the change in the chordwise location of the **transition point** with increasing angle of attack and produce a clear plot showing the effect.
8. Shortly discuss what happens to the drag coefficient as you force the transition to be in a specific location. The transition point location can be changed through selecting the menu: `vpar → xtr`. Add a plot of the drag coefficient versus the point of forces transition.

[2] In case hand drawn figures are required in the assignments report make sure that they are clear. Use colors as needed. Add a date and signature to proof that the drawing was made by you.

9. Produce a plot that shows the **frictional coefficient**, C_f, along the upper side of the airfoil for $\alpha = 0$ and $\alpha = 4$.
10. **Annotate this plot** and indicate what flow is found in the various areas. Add a short descriptive text in which this C_f plot is discussed.

Part 2 Airfoil improved design

The airfoil that you have been using in Part 1 may not be optimum w.r.t. its performance (glide ratio C_l/C_d) at cruise lift coefficients. Therefore, in this part of the assignment you will be using the **inverse airfoil design** option in XFOIL to manually design an airfoil that has an increased amount of laminar flow at the wing upper side, thus **improving the lift/drag ratio**.

To design the improved airfoil, perform the following tasks:

1. Calculate the C_l/C_d ratio of the airfoil used in Part 1 at a fixed cruise lift coefficient[3] of $\boldsymbol{C_l = 0.4}$.
2. Explain how the shape of the airfoil may be changed to improve the lift to drag ratio of this airfoil through shape adaptation. **Add hand drawings to support your reasoning**.
3. Use the **inverse airfoil design** routines in XFOIL to manually design an airfoil with an increased amount of laminar flow at $C_l = 0.4$ and present its higher C_l/C_d value. Make sure that the original airfoil and the optimized airfoil have the **same relative thickness**, t/c! You can do this by **scaling the airfoil thickn**ess using a specific routine in XFOIL `gdes → scal → 0` (to allow only y-scaling) `→ 1` (maintains the x-scaling) → `<value>` (this will be the scaling factor for the y-coordinate).
4. Plot and shortly discuss the typical pressure distribution differences between the original and the modified one.
5. Show and discuss the effect that the airfoil shape adaptation has on the position of the transition point.

Part 3 The laminar separation bubble

At **low Reynolds nu**mbers the airfoil may develop a so-called **laminar separation bubble**. This has consequences for the drag and the stall behaviour of the airfoil. To analyze the effect of a bubble on airfoil performance perform the following tasks:

1. Shortly explain why the **laminar separation bubble** often leads to an **increase in the drag coefficient**. Use a **hand drawing** to show the flow pattern (streamlines) in and around the bubble and indicate (in a qualitative way) how both the pressure drag and skin friction drag change in the bubble region compared to a case where no bubble is present. Remember to add a date and signature to the drawing.
2. Take the original airfoil of Part 1 and select a low Reynolds number (typically between 2×10^5 and 5×10^5) at which this airfoil produces a **clearly recognizable laminar separation bubble.** Use a n-factor of $\boldsymbol{n = 12}$.
3. The **length of the bubble** will become larger when the **Reynolds number** is lowered. Explain shortly why this happens.

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[3] This can be done by selecting `CL` in the OPER menu.

4. Using the same airfoil, **remove the laminar separation bubble** to reduce the drag coefficient at a fixed lift coefficient (you may choose one). For this, carefully select a position to fix transition on the upper side of the airfoil. Present the drag coefficients found in a table.
5. What is the **distance between the point of natural transition and the fixed position** that you selected? Discuss why a **small or larger distance** will not work to lower the drag.
6. Explain what is meant with the so-called "critical roughness height". Add a hand drawing to clarify.

Appendix A. Airfoil selection

The form (NACA designation) that you analyze is based on your **student number**. To find the airfoil shape that you need to investigate within this assignment do the following:

1. Take the last 3 digits of your student number, for example: 989, and take the sum, which we will denote N. In this example: N=9+8+9=26. This number determines the relative thickness of the airfoil you are going to investigate.
    a. In case your number is 5 or lower, use N=10
    b. In case your number is between 5 and 10, add a zero in front (so 9 becomes 09).
2. The first digit of the NACA designation is always 2
3. The second digit of the NACA designation is the second digit of your student number.

Some examples of this calculation are provided in the table underneath (check).

Student number	N	NACA Airfoil
4223977	23	2223
4218124	7	2207
4303784	19	2319
4712900	09	2709
4599667	19	2519
4391594	18	2318
4724143	8	2708
4221000	10	2210
4078005	10	2010

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