HOMEWORK 1

Write a computer code to analyze NACA 4-digit airfoils with trailing edge flaps using the Discrete Vortex Method (DVM). The inputs of the computer program should be the airfoil designation, the angle of attack of analysis and the number of panels (N) along the camber line. In addition, the flap hinge location (x_h) and deflection angle (η) are required for flap simulation. This will be done simply by modifying the coordinates of the airfoil's camber line along the flap by rotation about its hinge.

With the input data described, the program should compute the lift coefficient (C_l) and the coefficient of moment about the leading edge (Cm_{le}). In addition, the flap's lift (Cl_f) and pitching moment coefficient about the hinge (Cm_f) should be computed in flap simulations. Once your code runs properly (**test the example in M2_4 pp. 12** to be sure of that!), carry out the following analyses.

1. Verification assessment

In order to verify the DVM's solution, choose a **NACA 2408 airfoil** with no flap deflection and angle of attack α=4º and calculate reference exact values using the Thin-Airfoil Theory (TAT). Then, assess grid convergence by performing simulations with an increasing number of panels along the chord, e.g. N = 4, 8, 16, 32, etc., up to a **reasonably** maximum number of panels (e.g. N<200). Plot the computed C_l and Cm_{le} coefficients as a function of the number of panels and compare these values with the reference TAT solution. Also, calculate the percentage of error. In the report, you must **present only two plots**: one with C_l convergence and error (in the same plot, using different vertical axes) and one with Cm_{le} convergence and error.

2. Validation assessment

Using the same NACA airfoil with no flap deflection and the number of panels based on the grid convergence analysis above, calculate the lift slope ($Cl_{,\alpha}$), the zero-lift angle of attack (α_{l0}) and the coefficient of moment about the aerodynamic center (Cm_0). Compare your numerical results with experimental (high-Reynolds number) data and calculate the error.

For comparison purposes, you can use data from the book *Theory of wing sections* of Abbot and Doenhoff (1959). Please, **present these results in a single table**.

Calculate the flap efficiency for flap-chord ratios 0.15, 0.2, 0.25 and 0.3 and compare your results with the experimental values presented in pp. 190 from Abbot and Doenhoff's (also reproduced in M2_3 pp. 37). In addition, check the suitability of the flap effectiveness correction (see M2_3 pp. 38) for a few data points and extract some conclusions from the results obtained. **These results must be presented in a single plot.**

Finally, using the same flap-chord ratios of above, calculate de variation of the flap's lift and hinge moment coefficients with the flap deflection[1]. Also obtain the force and pitching moment (per unit span) acting on the hinge. Consider a symmetrical airfoil at zero angle of attack flying at sea level with speed 100 m/s and flap deflections up to 20 deg. **These results must be presented in two plots**, one with the variation of coefficients and another with forces and moments.

3. Discussion

Taking as a reference a NACA 4-digit camber line, examine the behaviour of α_{l0} and Cm_0 with the maximum camber (*f*) and position (*p*). You must prepare **only one plot for each variable** (i.e. α_{l0} and Cm_0). The plots must show curves for the variables to be analyzed as a function of *p* for different values of *f*. Use reasonable values for the maximum camber and position, e.g. f = 0, 0.02, 0.04, 0.06 and p = 0.1, 0.2, 0.4, 0.6. Draw some conclusions about the observed behaviours.

Submission

This homework must be performed in **groups having a minimum of 4 students and a maximum of 5**. The computer code must be included in an appendix, and the entire document (i.e. points 1, 2 and 3 above and appendix) cannot exceed **12 pages**. Report submission will be online (ATENEA) and the deadline is **Dec 20, 2021**. Late submissions will be penalized (-10% of the full grade).

[1] It is customary to define the flap coefficients using its surface area S_f and chord c_f. The flap area is $S_f = b_f * c_f$, where b_f is the flap span (b_f=1 in 2D calculations).