Aerodynamics Laboratory Report

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Part 1: Hess-Smith method

1. Implementation and validation of the code for Hess-Smith method

The validation of the Hess-Smith implementation on MATLAB® is made using Xfoil. The two share a similar panel method, but with some differences:
→**pressure coefficient (CP)**: in Hess-Smith it is computed in the control points (in the centre of each panel), whereas Xfoil computes it in the nodes of the airfoil (which are the extremities of the panels);
→**viscous mode**: in the MATLAB® implemented code there is no viscous mode, and therefore it allows only the investigation of attached flow (an underestimation of CP is thus expected).
Using a value of $\alpha = 1^o$ (as the profile is symmetric and would not produce aerodynamic forces without an angle of attack), the values of CP and CL obtained using the above mentioned methods appear to be slightly different. However, the CP diagrams appear to be almost completely overlapping (Figure 1a), but for the trailing edge bit. This is due to the fact that the trailing edge is sharp, and therefore the method is not to be considered accurate there. Defining an error function for the CP

$$err_{CP} = |CP_{HS} - CP_{XF}| \tag{1.1}$$

and introducing a polynomial interpolation of the CP values obtained from the Hess-Smith method in order to have the same number of values for Hess-Smith and Xfoil, it is possible to see that the relative error is sufficiently small (Figure 1b), thus negligible, clearly but for the above mentioned problem with the trailing edge.

Moreover, the obtained values of CL ($CL_{HS} = 0.119651$, $CL_{XF} = 0.117100$) and CM ($CM_{HS} = 0.001727$, $CM_{XF} = -0.000900$) also differ of a negligible error (the moment coefficient opposite sign is only due to a different definition of the reference frames in the two methods, *[Drela (1989)]*).

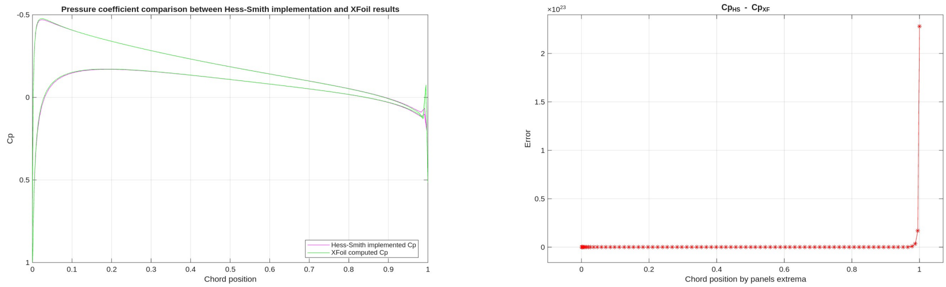
The MATLAB® implementation is thus to be considered validated.

2. Comparison between the profiles NACA0008 and NACA1410

Let us consider two different profiles. The NACA0008 is a symmetric thin profile whereas the NACA1410 is a non-symmetric but still thin profile. Therefore, when comparing them, some differences are to be expected.

It is interesting to analyse when laminar-to-turbulence transition occurs, when separation occurs, and how these results depend on the level of turbulent intensity, for the two selected airfoils. For this task, Xfoil is used.

As a starting point, it is important to highlight some critical characteristics of the



(a) *Pressure coefficient* (b) *Error*
Figure 1: *C_p comparison between Hess-Smith and Xfoil*

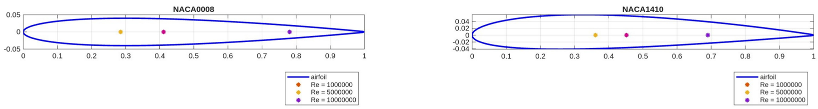


Figure 2: *Chord position of transition points on the suction side of the airfoils*

selected instrument, that must be taken into account when performing this type of analysis. When working in viscous mode, an important parameter is represented by the iterations limit (here set to 250), which determines wether Xfoil's iterations will actually converge to a solution or not. In addition to that, in order to investigate transition and separation it would be significant to choose high values for α, but to preserve the accuracy of the method this cannot be considered. *[Drela (2001)]*

A representative set of Reynolds numbers is chosen instead, so as to obtain significant results, while remaining within the approximation capabilities of the method. In this case, three values are considered to be sufficient to represent three different levels of turbulent intensity (low intensity 10^6, intermediate intensity $5 \cdot 10^6$ and high intensity 10^7).

Regarding transition, it is possible to extract the chord position of the transition points directly from Xfoil (Figure 2). It is observed that for the second profile transition delayed with respect to the first one, which was expected because of the non symmetry of NACA1410. Furthrmore, both cases clearly show that the higher the Reynolds number, the farther away from the leading edge the transition happens.

Whereas for separation the skin-friction coefficient is a powerful indicator that can be investigated (Figure 3). Separation, represented by an adverse pressure gradient, is a primary driver of transition, and is identified by a change in sign of C_f. *[Anderson (2024)]* When a laminar boundary layer separates from an airfoil, it forms a detached shear layer. Because the flow is no longer restrained by the friction of the solid surface, it undergoes transition to turbulence much faster than an attached layer would, which eventually causes the flow to reattach. Lowering the Reynolds number typically makes profiles more susceptible to separation.

In this case, separation is not observed for the chosen Reynolds values, on both profiles, as the friction coefficient does not become negative. It is clear however that in all the considered cases transition occurs, indicated by the presence of the so-called transition bubbles. Therefore, we deduce that transition occurs before the flow has the chance to separate.

Part 2: Weissinger method

3. Implementation and validation of the code for Weissinger Method

For the validation of the MATLAB® implemented code, the software XFLR5 is used. The latter implements a general vortex lattice method, for which the former represents a

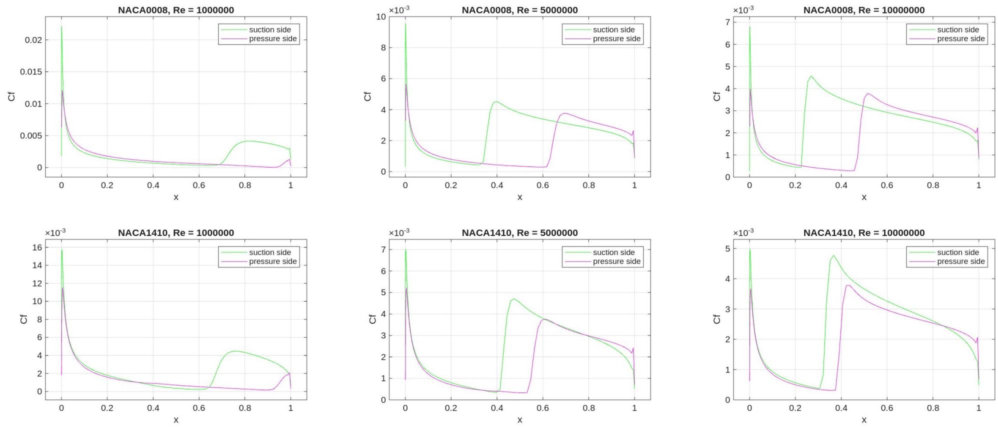


Figure 3: Friction coefficient diagrams

	C_L	$C_{D,ind}$
Weissinger	0.1630	0.0011
XFLR5	0.16221	0.00107

Table 1: Code validation results for $\alpha = 2^o$

special case. For simplicity, the validation is made using a single, rectangular wing with no sweep nor dihedral angles. Once defined the same geometry both for the Weissinger implementation and XFLR5, the numerical results that are obtained are reasonably similar (Table 1), therefore the MATLAB® implementation can be considered validated.

4. Comparison between Cessna 172 Skyhawk and Tecnam P2002-JF considering wing and tail plane

Let us consider two different aicrafts: Cessna 172 Skyhawk *[Cel (2019)]* and Tecnam P2002-JF *[Costruzioni Aeronautiche TECNAM S.p.A. (2004)] [European Union Aviation Safety Agency (2021)]*. It is interesting to study the combined aerodynamic behaviour of their wings and tailplanes, comparing the two results. Together with distinct geometric characteristics, the two aircraft comprise a major difference in the wing configuration: the first one has a high-wing, whereas the second one has a low wing, resulting in the former to have the tail below the wing plane while the latter has the tail located only marginally above. This will have an impact on the behaviour of their lifting surfaces, as the two tails will experience a differently diverged flow due to the interaction with the corresponding wing. It is important to underline that the chosen method is taking into account only the wing and tail interaction; therefore any other contribution (such as the interaction with the fuselage) is not considered for the purpose of this study.

The study is carried out using the above mentioned MATLAB® implemented and validated method. It is interesting to analyse the distributions of the lift coefficient along the span (Figure 4), and also compare the two polar curves (Figure 5a). Finally, a comment on the induced angles on the tailplanes is also significant (Figure 5b).

The computed values of C_l for the two aircraft clearly show some similarities and some differences. Both have a negative lift generated on the tail, which compensates

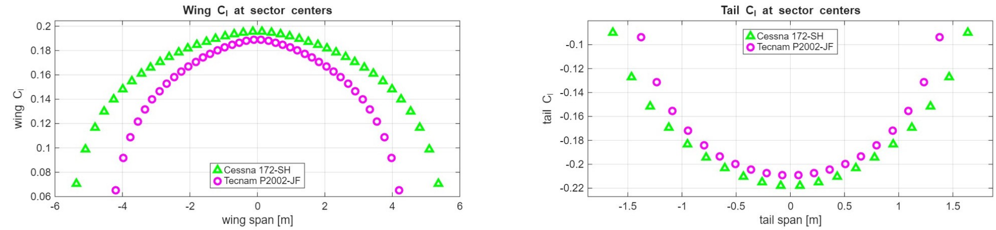
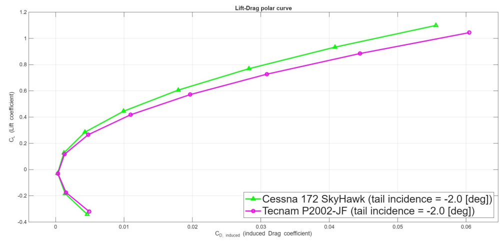
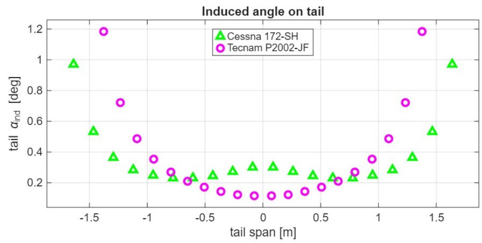


Figure 4: *Computed C_l distribution along wing and tail spans*



(a) *Comparison of the two polars*



(b) *Computed induced angles on the tails*

Figure 5

for the high lift on the wing. However, the absolute values of C_l are slightly higher for the Cessna with respect to the Tecnam. This can be justified by the different geometric characteristics of the two wings: the former has indeed higher aspect ratio and wing span than the latter, allowing for more lift to be generated. The plot of the polar curve also confirms this conclusion: the two curves appear to be very similar, with the one referred to the Cessna being slightly above the one referred to the Tecnam.

A final remark is to be made regarding the induced angles on the tails, whose distributions appear to be different. This is due to the distinct wing configurations of the airplanes. The high wing on the Cessna does not influence too much the flow that then goes on the tail, which is located below the wing plane, resulting in a "w-shaped" distribution of the induced angles. This is not observed for the Tecnam: being the tail almost on the same level as the wing, it feels the influence of the wing on the flow, resulting in smaller induced angles in a more "u-shaped" distribution across the tail span.

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