Correction of Wind Tunnel Results for the Airfoils of ITA's Unmanned Aerial Vehicle

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Abstract. *Unmanned Aerial Vehicles (UAV) have became more important in the last years, due to the low prices of electronic systems required for the guidance and control of an aircraft. This kind of vehicle is capable of performing a large number of missions with lower costs compared to manned aircraft. The UAV studied in this work must inspect the elements of an energy transmission line, that is, the towers, the cables and the nearby vegetation. The speed of the aircraft should be relatively low (close to 80 km/h) and, thus, it is necessary to obtain an aerodynamics database for low Reynolds numbers. At the tests of airfoils performed at ITA, corrections were needed in order to eliminate wall effects. To perform these corrections, a computer code, based on a panel method formulation, was used. The applicability of this code for the Reynolds numbers typical of unmanned aerial vehicles is discussed using the results obtained with the wind tunnel.*

Keywords: *aerodynamic profiles, block rate, wind tunnel corrections.*

1. INTRODUCTION

Many authors treated the subject of wind tunnel corrections, such as Theodorsen (1931), who studied corrections for wind tunnel tests with open walls, and Batchelor (1944), who developed a method to treat wind tunnels with octagonal section. The most used method is the one proposed by Pope (1984).

However, there are few references that present corrections for wind-tunnel tests of airfoils of any thickness and camber in a rectangular wind-tunnel section where the flow can be considered bi-dimensional. Lock (1929) and Glauert (1993) studied the interference of the walls on the flow over symmetrical airfoils at zero angle of attack. Glauert (1993) also studied bi-dimensional flows over thin cambered airfoils inside a test section. Goldstein analyzed the interference of the wind tunnel walls for a cambered airfoil of finite thickness in an incompressible fluid flow. In another work, Goldstein and Young (1942) presented compressibility corrections. Allen and Vincenti (1944) presented a method of correction for an airfoil in a flow with compressibility effects. However, this method is valid for configurations with small thicknesses and cambers, and the disturbances are small if compared to the undisturbed flow velocity.

For the wind tunnel tests of ITA's Unmanned Aerial Vehicle, a high blockage ratio became necessary to test the wing airfoil at the cruise Reynolds number. There was also a need to correct the results for high cambers, since the airfoil was tested with great flap deflections. Therefore, the existent methods were not sufficient, and there was a need to develop a more general calculation based on a panel method.

The fundamentals of the method of correction are presented in this paper. Further reference should be made to Gomes (2005). Also, a presentation is made of the calculations performed to correct the lift and pitching moment coefficient of the wing airfoil, showing how the described procedure was applied to eliminate blockage ratio effects for the performed wind tunnel tests.

2. NUMERICAL IMPLEMENTATION

2.1. Mathematical model

The problem treated in this work is the calculation of the pressure and velocity fields of a steady flow over a bidimensional airfoil with wall restrictions. Figure 1 shows the geometry of the problem, with a representation of an airfoil with chord *C* and angle of attack α in a wind tunnel. The walls have length *L* and are separated by a distance *h*. The undisturbed flow velocity is V_{inf}.

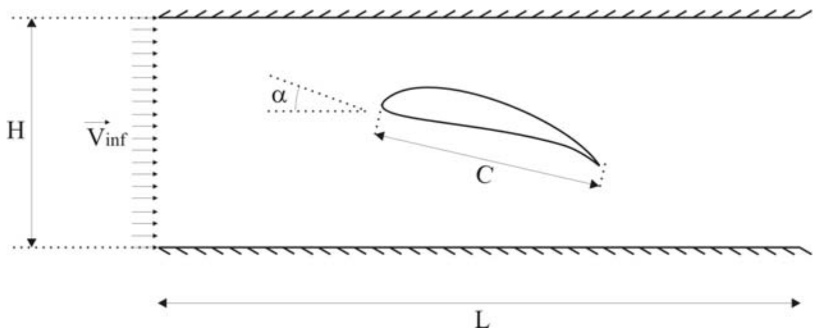

Figure 1. Representation of the aerodynamic problem.

The desired result for the calculations is the pressure distribution over the profile surface, which allows the calculation of the lift and the pitching moment with the appropriate integrations. This problem can be solved using the model of incompressible irrotational flow. With this model, it is possible to define a velocity potential *Φ*, being the velocity given by the potential gradient.

$$\vec{V} = \nabla \phi \qquad (1)$$

For incompressible flows, the continuity equation for the velocity potential is Laplace's equation, as shown below.

$$\nabla^2 \phi = 0 \qquad (2)$$

The boundary conditions are of zero normal velocity at the surfaces, satisfied at both the airfoil and the tunnel walls. Also, the velocity must be equal to $\boldsymbol{V_{inf}}$ at large distances upstream or downstream the airfoil. The boundary conditions at the surface for the velocity potential become

$$\frac{\partial \phi}{\partial n} = 0\,. \qquad (3)$$

In the equation above, *n* is the unit vector normal to the surface.

The solution of Eq. (2) subject to the boundary conditions of Eq. (3) leads to the velocity field, which is related to the velocity potential by Eq. (1). The determination of the pressure field is possible with Bernoulli's equation in the form for incompressible irrotational flows.

$$\frac{V^2}{2} + \frac{p}{\rho} = cte \qquad (4)$$

The constant in Eq. (4) can be calculated with the conditions (velocity, pressure, density) at the infinity.

To solve numerically the problem posed by the equations above, a panel method was used, as proposed by Hess and Smith (1967). In this model, the surface is divided in panels, and an appropriate distribution of singular solutions (sources, doublets and vortices) along this panels allows the determination of the potential in the field. The division of the surface in panels with a fixed distribution of singularities (constant along the panel, for example), and the definition of a control point where the boundary conditions must be satisfied, usually at the center of each panel, makes the solution possible by means of a linear system, with the unknowns associated to the parameters of the singularities distribution over each panel. The solution of this system gives the distribution of singularities at the boundary of the problem, which allows the calculation of the potential at any point of the field.

The following section shows how the discretization was made for the present problem.

2.2. Discretization of the Airfoil and of the Tunnel

In order to represent the geometry of the problem, it is necessary to define the position of the axis of rotation of the airfoil. This position should be equal to that of the wind tunnel model for an accurate representation of the blockage variation with angle of attack changes.

Two discretizations must be done: one at the airfoil surface, and another at the tunnel walls. The discretization of the airfoil was done with small panels at the leading and trailing edges, and greater panels at the middle of the profile. This was done to allow better calculations at the leading and trailing edges, with a greater number of panels.

The wind tunnel walls were divided between three regions, as shown below. A central region, Region 2, was defined as correspondent to the airfoil position. The two other regions are labeled 1 and 3.

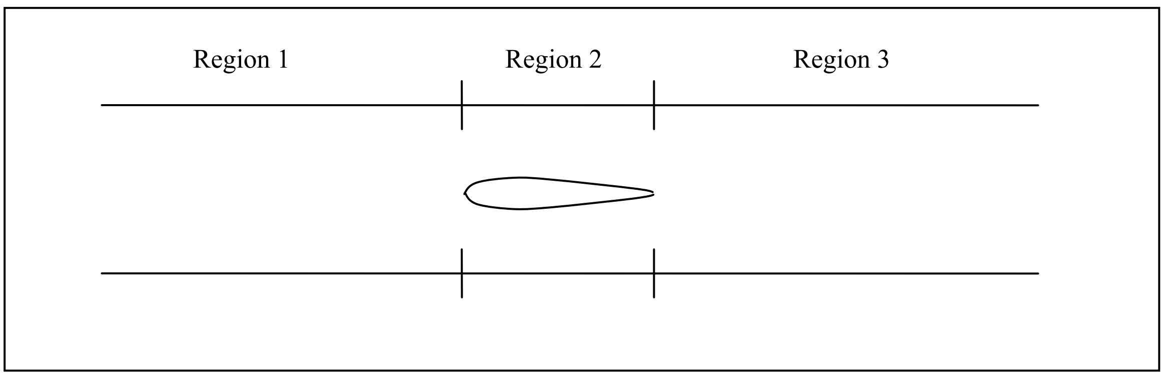

Figure 2. Division of the wind tunnel walls between regions.

This division was done in order to allow a finer discretization at the central region, which is the one with greater velocity and pressure gradients, since it is closer to the airfoil.

Region 2 of the walls was divided into equal sized panels. The discretization of regions 1 and 3 was done with a sinusoidal variation of the panel size, beginning with the panel size adopted for Region 2, and with greater sizes as the distance to the airfoil is increased.

At both the airfoil and the wall panels, control points were defined at the centers, where the condition of zero normal velocity must be satisfied.

In order to describe accurately the flow at the trailing edge region, the Kutta condition must be respected. This was done by the introduction of a new control point, as close as possible to the trailing edge of the airfoil, where an additional condition was imposed, of zero velocity normal to the camber line.

Being thus defined the division of the surfaces between panels, the singularities distributions must be chosen in order to determine a linear system for the solution of the problem. A constant distribution of sources was specified for each of the airfoil panels, being the constant an unknown to be calculated. A third-degree polynomial distribution of vortices was specified for the whole airfoil, with zero vortex density at the two trailing edge panels, and a maximum of vortex density at the leading edge. This leads to only one unknown related to the vortex distribution, which is the value of this maximum.

The wind tunnel walls were treated as flat plates without thickness. Thus, a normal doublet distribution was done at the wall panels. The doublet distribution was specified as constant at each panel.

With these singularity distributions, it is possible to form a linear system, with n unknowns and n equations related to the n control points, including the one relative to the Kutta condition.

Gomes (2005) presents a detailed study of the convergence of the method as the number of panels at the tunnel and at the walls is increased. The analysis showed that 256 panels for the airfoil and 128 panels for each of the walls provided good results, and a further increase of the number of panels does not change significantly the values of the lift and moment coefficients.

2.3. Method for wind-tunnel correction

The presented mathematical model is a valid approximation for angles of attack where the boundary layer does not change much the pressure distribution. Therefore, the proposed correction is appropriate for the linear range of the lift curve.

The method of correction is represented for the two equations presented below.

$$k = \frac{(C_{L\alpha})_{without\ tunnel}}{(C_{L\alpha})_{with\ tunnel}} \tag{5}$$

$$(C_L)_{corrected} = k \cdot (C_L)_{measured} \tag{6}$$

The procedure consists in perform the panel method calculation for the desired airfoil without the tunnel walls to obtain its lift curve for this configuration. After that, the same calculation is performed considering the airfoil and the tunnel walls. The factor *k* is the ratio between the slopes of the two curves. This factor is multiplied by the measured lift coefficients in order to determine which would be the values of C_L without the influence of the tunnel walls.

A similar procedure can be done to correct the pitching moment coefficient.

2.4. Validation of the method

The procedure for the validation of the method is based on the comparison of the results corrected with Eqs. (5) and (6) with the results corrected by the method presented by Allen and Vicenti (1944).

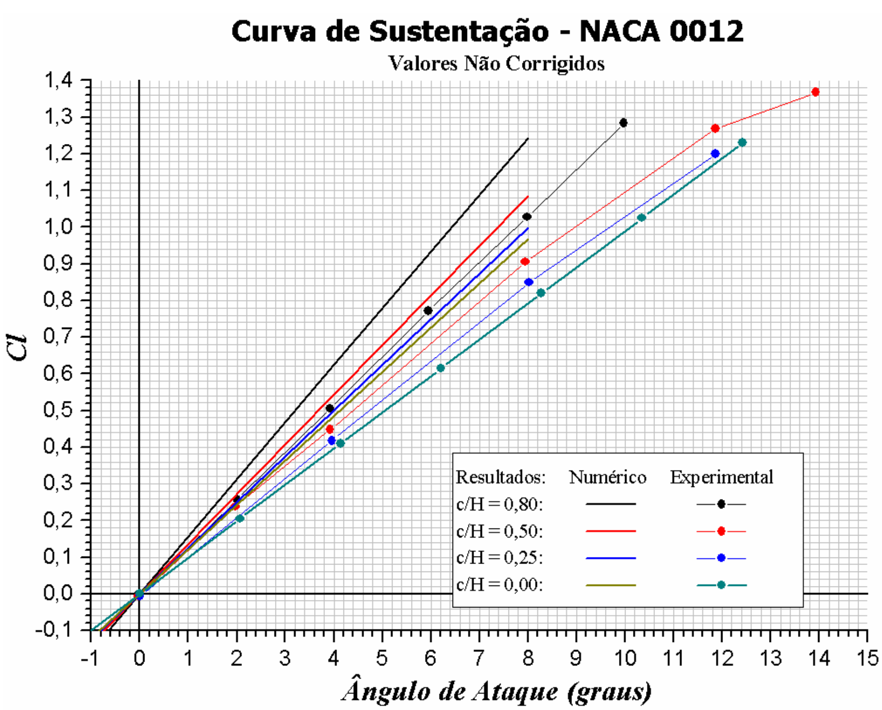

Figure 3. Lift curves obtained with the present method (numerical) and experimental curves without wind-tunnel corrections.

Figure 3 shows the results for a NACA 0012 airfoil obtained with the present method and the measured values for the lift coefficient presented by Allen and Vicenti (1944) for some blockage rations c/H. The value of zero c/H ratio means free flow, that is, without restriction by the wind tunnel walls.

Some differences can be seen between the numerical and experimental results for the lift slope. The numerical results show a greater C_L than the experimental value, for any of the blockages considered. This is due to boundary layer effects, as shown by Schlichting (1979). The presence of the boundary layer, which is not considered by the present method, changes the shape of the effective aerodynamic body, and causes a decrease of the effective camber, which causes a loss of lift.

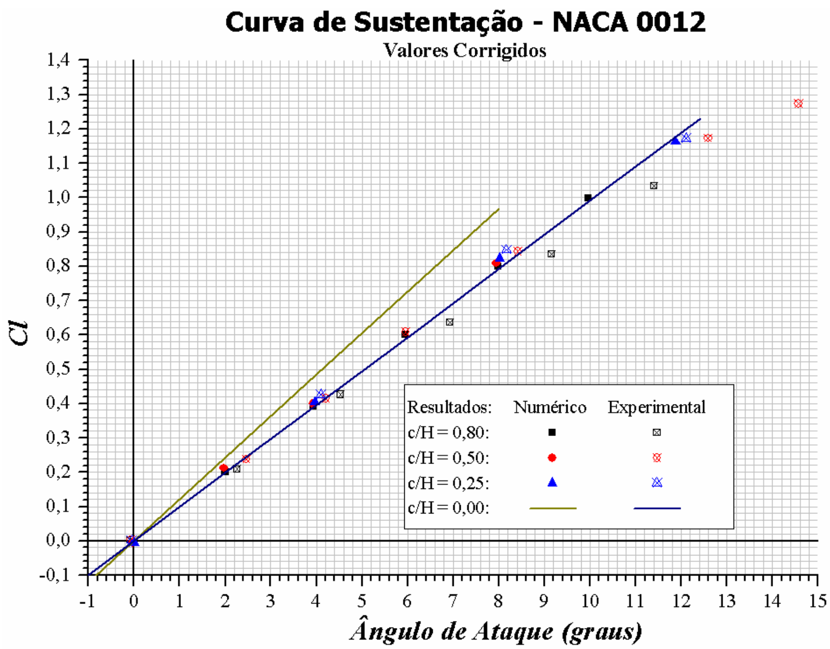

Figure 4. Lift curves corrected with the present method (full symbols) and curves corrected with the method of Allen and Vicenti (1944) (open symbols with cross)

Figure 4 shows the results corrected by the present method (full symbols) and the results corrected by the method presented by Allen and Vicenti (1944) (open symbols with a cross). The numerical line corresponds to the curve obtained by the present method for the flow without the tunnel walls, and the experimental line corresponds to the results for c/H = 0.

The results obtained with the present method are very close to the experimental values of the reference. The difference between the lift slopes for the two methods are of 1.17%, 2.48% e 0.09%, for ratios c/H equal to 0.80, 0.50 and 0.25, respectively.

3. CORRECTION OF WIND-TUNNEL RESULTS FOR ITA'S UAV

The method presented here was used to correct the experimental data obtained for the wing airfoil of ITA's unmanned aerial vehicle. The studied airfoil was Selig-Donovan SD7062. The model had a chord of 361mm, and was tested in a square section with side equal to 460mm, which gives a c/H ratio of approximately 0.785.

This airfoil was tested with plain flap deflections varying from 0° to 26°, with changes of 2° between the deflections. To correct the results for the lift and the pitching moment coefficient, a different calculation was made for each flap deflection, since the way each configuration is influenced by the wind tunnel walls is not the same. The deflection of the flap was done in the method only by a rotation of the flap region around the hinge. The airfoil continues to be a single element, with a break in the flap region, as shown in the figure below.

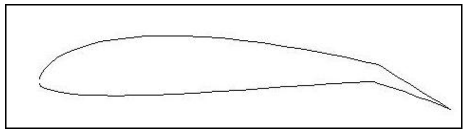

Figure 5. Geometry used to simulate an airfoil with flap

The following figure shows the calculations for the lift coefficient for the airfoil without flap deflection.

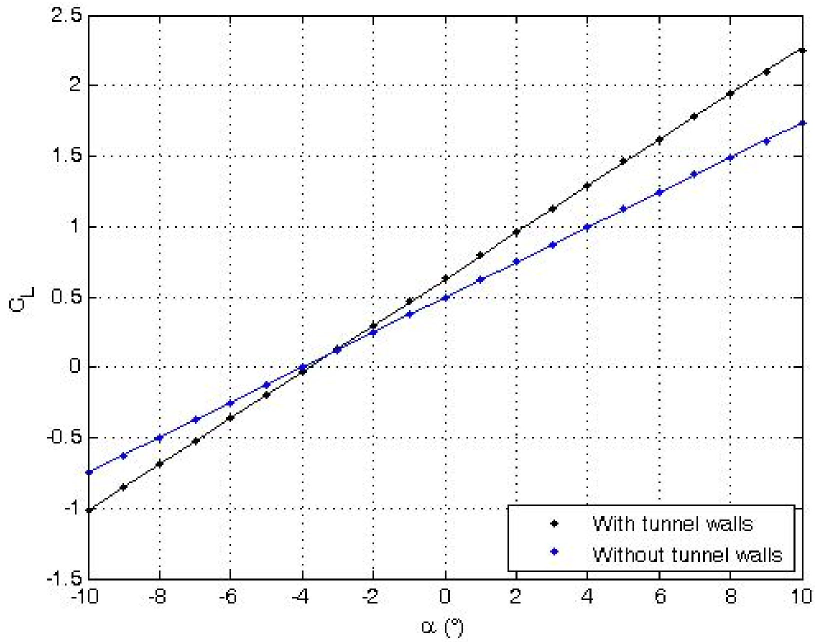

Figure 6. Calculated lift coefficients for the SD7062 airfoil without flap deflection

The presented results show an increase of the lift coefficient if the wall interference is considered. The same effect was verified in the validation of the method. The factor *k*, as shown in Eq. (5), for this flap deflection was 0.756.

The results presented in Figure 6 show that there is not a significant change in the angle of attack for zero lift due to the blockage effect. Hence, only the lift slope should be corrected for this deflection.

This result cannot be used for all flap deflections. The calculated values for the lift coefficient are shown in the following figure for a flap deflection of 26°.

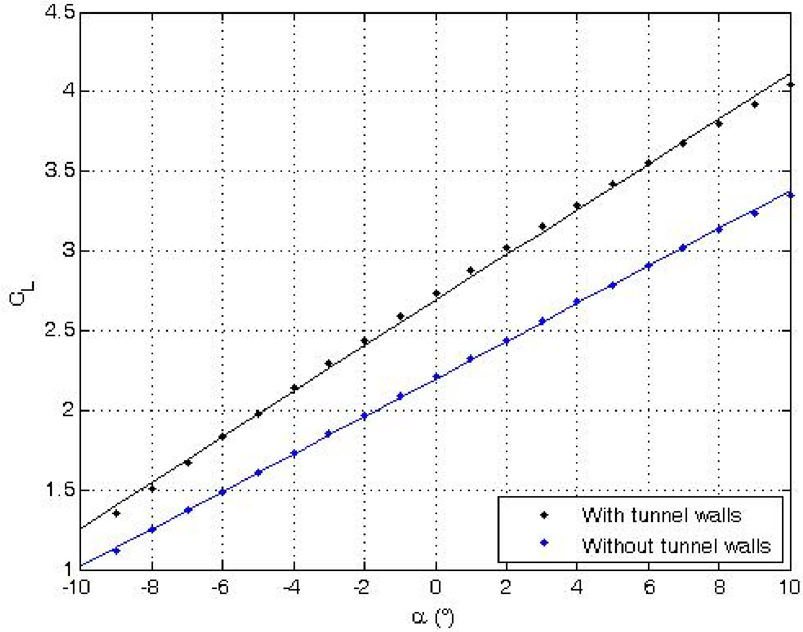

Figure 7. Calculated lift coefficients for the SD7062 airfoil with flap deflection of 26°.

For this flap deflection, the calculated value of k was 0.832. Therefore, a different correction must be performed for each flap deflection, since the variation of the lift slope is different for each flap deflection. This happens because a change in the deflection causes a different augmentation of the blockage ratio as the angle of attack is increased. Consequently, a different value of k is expected, and the corrections must be performed independently for each deflection.

Although it is not shown in Figure 7, the angle of attack for zero lift has a significant change for that flap deflection. To show this effect, the figure below shows the results for a flap deflection of 8°.

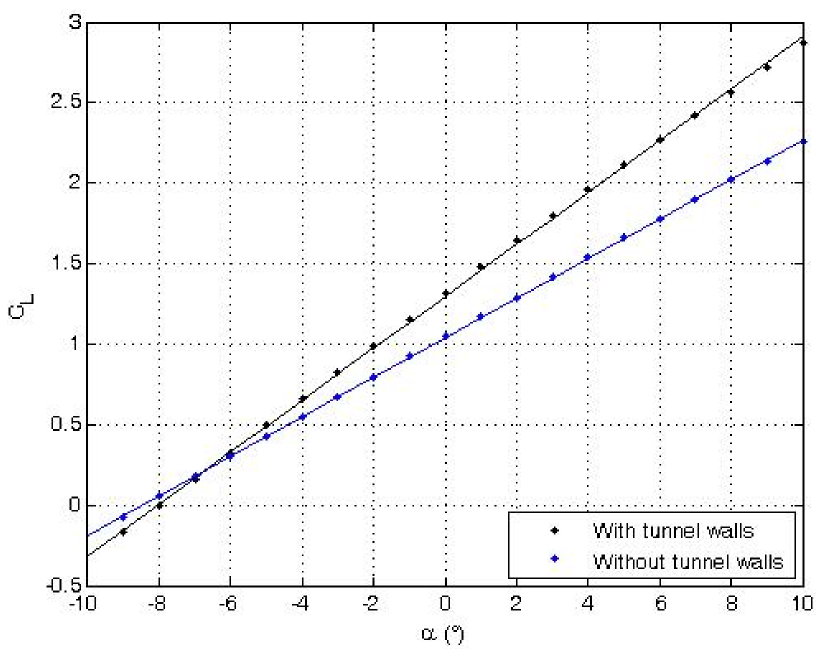

Figure 8. Calculated lift coefficients for the SD7062 airfoil with flap deflection of 8°.

For the results presented in Figure 8, a change of 0.5° was calculated for the angle of attack of zero lift. This change becomes greater for higher flap deflections. Consequently, for an accurate correction, this variation must be considered. Corrections for symmetrical airfoils or for airfoils with small cambers can disregard the variation of α_0, but this cannot be done for airfoils with a high camber or a high flap deflection.

Corrections were also performed for the moment coefficient. The results for the airfoil without flap deflection are shown below.

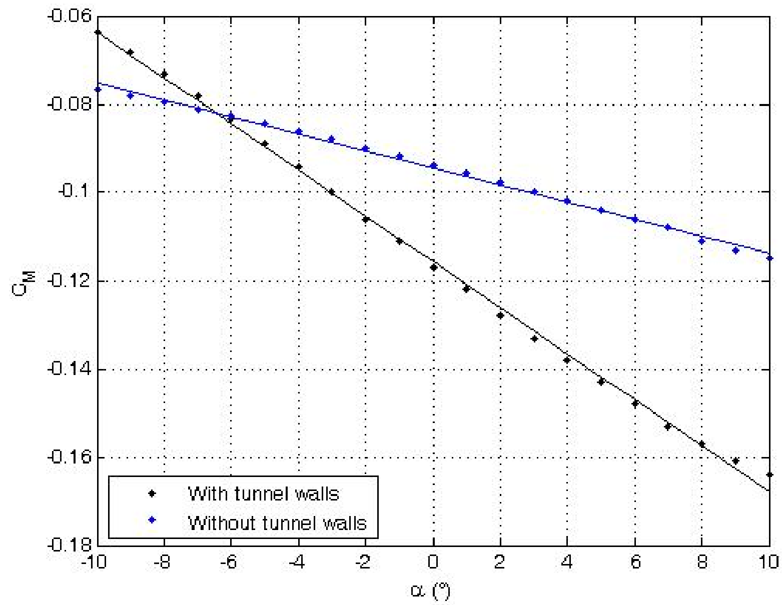

Figure 9. Calculated pitching moment coefficients for the SD7062 airfoil without flap deflection

The results presented in Figure 9 are fitted with a linear function. It is possible to see that the linear fit is a good approximation for both calculations, with and without the tunnel walls. However, for higher flap deflections a polynomial fit was necessary to provide a good representation, as shown below for a flap deflection of 26°.

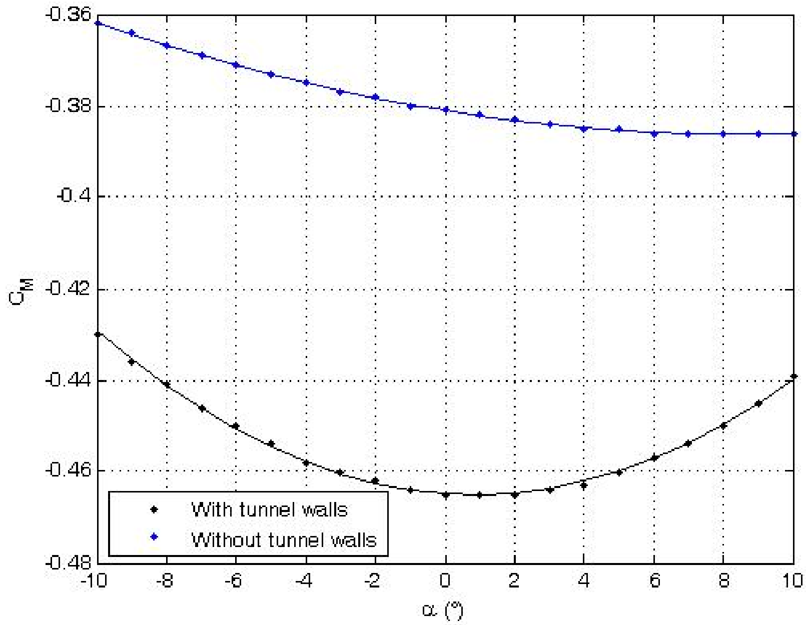

Figure 10. Calculated pitching moment coefficients for the SD7062 airfoil with flap deflection of 26°.

The results presented above were fitted with a second-degree polynomial. This kind of function provided accurate representations for all the tested flap deflections.

Since the moment coefficient ceased to be linear with the angle of attack, the suggested method for C_L correction was not appropriate to correct the pitching moment coefficient. A correction for each angle of attack was performed, with the calculation of a ΔC_M for each angle of attack. This value was used to correct the measured value of C_M for the corresponding angle.

4. FINAL REMARKS

The validation performed with the present method showed that this is an accurate technique for the range of angles of attack without boundary layer separation. However, nothing can be said about the corrections for higher values of α, and, specially, for the maximum lift coefficient. A further development for the present method would be the analysis of boundary layer effects at the airfoil surface, and this must be done in order to allow a correction for angles of attack outside the linear range.

The results showed that special attention must be paid for the correction of the pitching moment coefficient, since for high blockages it can become non-linear even for small angles of attack. For airfoils with small cambers, though, it is possible to use a linear fit for the moment coefficient and correct the results as was done for the lift.

Another correction that can be performed with a similar method is for the hinge moment coefficient for the flap. This will be developed in a future work, since the hinge moment determination is of great importance for the design of UAVs.

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