

AERODYNAMIC DESIGN OF TRANSONIC AIRFOILS

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Abstract. *Nowadays, during the aerodynamic transonic wings design, the aeronautical community has focused in optimization technique. Other design techniques, as direct design and inverse design, are in disuse or are being considered obsolete due to some factors. This occurs because the optimization technique has some advantages like high multidisciplinary and aerodynamic improvement. Moreover optimization technique finds the best geometry minimizing or maximizing the objective function using the restrictions as limits, independent of aerodynamicist experience. The drawback of this design technique is the high computational cost in comparison with direct and inverse design.*

The inverse design, technique that determines the airfoil which produces a target pressure distribution in a specific flow, presents better solutions than optimization method considering shorter simulation time. However the inverse design presents the drawback that requires the prescribed target pressure distribution that will produce the best aerodynamic performance and that isn't a trivial task even for experienced aerodynamicists.

There are simplified models that help the aerodynamicist in the target pressure distributions prescription, and because of their empiric formulation, it was evatuated their applicability using more fidelity solutions as MSES, a CFD tools. In this work it will be modified and evaluated this modified methodology to determine the control point locations and pressure coefficients levels from supercritical airfoil target pressure distribution. It will be evaluated also the methodology that model the velocity distributon between the control point from target pressure distributio, comparing its results with CFD results.

Moreover, there are design guidelines that yielded the best compromises in drag characteristics over a range of conditions. One of these guidelines propose to design supercritical airfoil at off-design conditions with pressure distribution that corresponds to a sonic-plateau. It was evaluated the applicability of this guideline using CFD tools.

Keywords*: parametric pressure distribution, aerodynamic design, inverse design, direct design*

1. INTRODUCTION

The airfoil design can be divided into three classes: direct design, inverse design and optimization technique.

In the direct design the aerodynamicist select an airfoil and modify its geometry at a Mach number to obtain a specific lift coefficient (*CL*), drag coefficient (*CD*) and moment coefficient (*CM*). This technique presents fast response time as pros and difficulties to deal with multidisciplinary restrictions as cons.

The inverse design determines the shape of the airfoil that is responsible for a target pressure distribution established by the aerodynamicist, at a specific flow condition. This target pressure distribution is established by the aerodynamicist to have good aerodynamic performance and the airfoil must satisfy structural and manufacture restrictions. Despite the fact that experienced aerodynamicist can identify desirable flow characteristics, for example, shock wave reduction and/or detached flow elimination, it is not a trivial task to establish a pressure distribution that provides this pros while maintain other aerodynamic requirements, like *CL*, *CD* and *CM*. The pro of this technique is the low computational cost in comparison with optimization technique.

The optimization technique tries to design an aerodynamic airfoil by minimizing or maximizing a given objective function, which is usually a function of aerodynamic coefficients such as *CL*, *CD* or *CM*, besides geometric parameters such as maximum relative thickness *(t/c)max* and leading edge radius, *rle*. The optimization technique presents the pro of finding the optimum airfoil to a known objective function despite the high computational cost required.

In the current scenario, the aeronautical community has been focusing in optimization technique. Other design techniques are in disuse or are considered obsolete, like the direct and inverse design. It occurs due to some good advantages that optimization technique provides like: considerable gain of aerodynamic and multidisciplinary efficiency; and stating properly the problem in terms of objective algorithm and restrictions, the optimization technique finds the optimum geometry regardless of aerodynamicist experience.

Author's experience is that each one of those techniques have its applicability in complex and high efficiency aerodynamic designs, as mentioned at Resende (2004).

In the pre-project, during the cruise wing design, it's recommended the use of optimization technique. In this phase, the wing lofting correspond to an artistic estimative from project group. Therefore the potential for aerodynamic improvement is high due to freedom to propose new wing loftings without activities schedule and development cost losses. Moreover, there are many multidisciplinary restrictions to be established.

During the project development there are several trade-offs between many disciplines that must be studied. It can be mentioned some trade-offs studies like: inboard panel thickness increase for main landing gear installation that reduces the structural weight and increase fuel volume; outboard panel thickness increase facilitates flaps/ailerons installation; but both modifications mentioned increase drag. Other very common trade-off is the leading edge droop to increase maximum lift coefficient (*CLmax*) in low speed versus loss of transonic performance. Due to the high number of trade-offs to be studied, the necessity for fast response time to do not compromise the activities schedule and the computational resource limitations make it more difficult, in the sense that, for each one of these numbered studies, a new wing optimization be done. Because of this, in these studies, it is recommended the use of direct/inverse design.

As mentioned before, the direct/inverse design demands a relative experience of the aerodynamicist. It is important comprehend how the bi dimensional geometry relates with pressure distribution. In this part can be cited the relation between the *(t/c)max* versus flow velocity levels and leading edge droop versus forward load.

Also it is important identify efficient pressure distribution behavior and how this behavior affect the airfoil performance. In this part can be mentioned the relation between the suction peak in leading edge versus *CLmax*, adverse gradient versus viscous drag (*CDviscous*), wave shock intensity versus wave drag (*CDwave*) and flow velocity levels in airfoil upper side versus drag creep.

Even with CFD tools development, to identify these relations and generate design recommendations are not trivial tasks. However, there are simplified analytical equations that can indicate the main compromises between these relations. Kim and Rho (1997) developed several simplified equations to represent the pressure distribution over a airfoil in function of its geometry and flow characteristics. Moreover, the authors relate the maximum relative thickness, *(t/c)max*, with pressure distribution in upper and lower surfaces.

In Campbell (1992) the author estimates semi-empirically control points for airfoil pressure distribution in function of design conditions as Mach, *CL*, *CM* and *CDwave* and geometry *(t/c)max*. Some of these equations were studied in Magalhães et al (2008).

Some authors like Harris (1990) used the experimental studies to propose guidelines in airfoil pressure distribution to maximize the transonic performance.

The objective of this work is to study the simplified equations for airfoil pressure distribution determination to help the aerodynamicists during direct/inverse design, using CFD tools to validate them and for evaluation. It will be studied the equations for determination of control point locations and pressure distribution, (*Cp*), levels, and it will be evaluated a modification in methodology. It will be evaluated the formulas for velocity distribution between these control point. Moreover, it will be evaluated the possible advantage to design an airfoil in off-design conditions, mentioned as a guideline in Harris (1990).

The CFD tools used for validation were XFOIL and MSES. XFOIL is a CFD program developed by Mark Drela for airfoil design and subsonic analysis, as mentioned in Drela and Youngren (2001). MSES is a CFD software that includes the capabilities to analyze, modify, and optimize single- and multi-element airfoils for a wide range of Mach and Reynolds number. The numerical formulation of MSES consists of a finite-volume discretization of steady Euler equations on an intrinsic streamline grid, coupled with boundary layer, as mentioned in Drela (1994).

To perform the simulations it was used several airfoils with different thickness. The baseline airfoil that was previously optimized at Mach 0.75 and *CL* 0.60, was modified using XFOIL to generate other airfoils with different thickness and constant camber. The airfoils generated present thickness from 9% to 16%.

The simulations performed by MSES used the followed parameters: Reynolds number of 9 millions and forced boundary layer transition at 10% of chord at upper and lower surface. The other parameters were used the values for standard model.

2. SIMPLIFIED MODELS VALIDATION

2.1. Control point locations and levels

The pressure distribution over an airfoil is divided into seven points in the upper surface and seven points in lower surface, as detailed in Campbell (1992) and sketched in Fig.1. The formulas to obtain the control points are calculated by flow, pressure distribution and geometric parameters: free stream Mach number, *CL*, *CM*, *CDwave* and *(t/c)max*.

In Magalhães et al (2008) the authors detailed the methodology and applied it in several supercritical airfoils to validate it. Supercritical airfoils, with thickness from 9% to 16%, were simulated in MSES, at specific conditions, and the pressure distributions from MSES were compared to the control point calculated by Campbell's methodology. The best result occurred in a airfoil with 13% of thickness at *CL* 0.60 and Mach 0.75. As mentioned in Magalhães et al (2008) conclusion, the Campbell's methodology is inconsistent because it needs the *CL* and *CM* to calculate the control point, but when the control point are linked and the integral is made, the resulted *CL* and *CM* are very different from the initial *CL* and *CM*. The figures in Magalhães et al (2008) indicate that the main errors occurred in control point number 2 and 6 located in the lower surface.

To reduce the error it was proposed to modify the *CP* value of these two control points, 2 and 6 in the lower surface, maintaining the *x/c* positions constant, until the *CL* and *CM* from Campbell's control points reach the previous *CL* and *CM*. The result of this modification is shown in Fig.2.

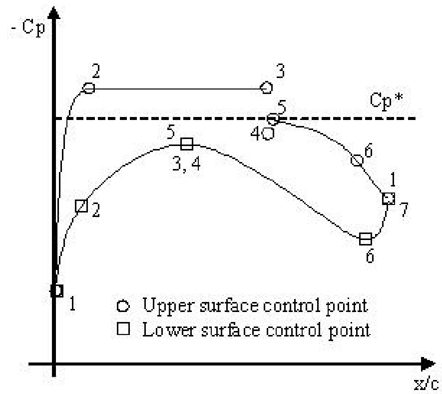

Figure 1. Control point from supercritical airfoil pressure distribution.

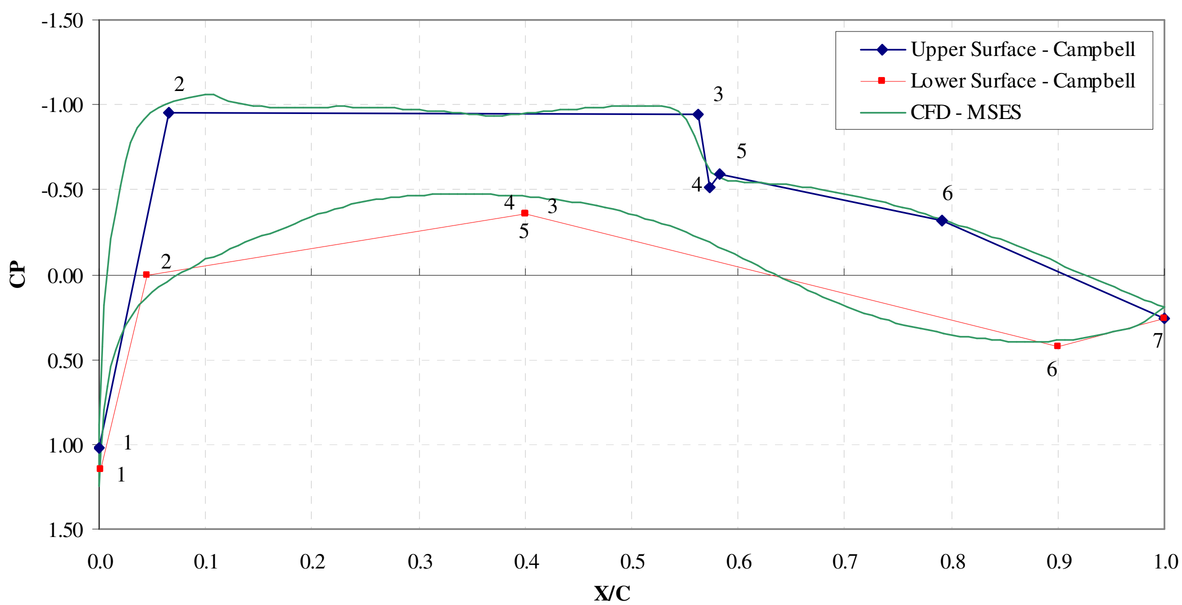

Figure 2. Campbell (1992) methodology with modified control point 2 and 6 from lower surface compared with MSES results. Airfoil with (t/c)máx = 13%, at Mach = 0.75 and CL = 0.60

With this *Cp* modification in control point 2 and 6, Campbell formulation turned consistent. Moreover, the modified control point are more close to MSES results.

2.2. Velocity distribution between control point

The velocity distribution behavior between the control points in the upper and lower surfaces were described in Kim and Rho (1997) and were divided in seven parts: stagnation flow region, rooftop region, shock relation, constant pressure region behind shock, pressure recovery region on upper surface, pressure recovery region on lower surface and rear loading region, as shown in Fig.3. The straight lines connecting the control points were used for simplicity.

The stagnation flow region in the upper surface starts in control point 4 and ends in control point 3, and in the lower surface starts in control point 4 and ends in the control point 5, detailed in Fig.4. The stagnation flow shape is defined by approximating the potential flow velocity distribution around elliptic cylinders (for small *x/c* and small incidence angle), detailed by Eq. (1), Eq. (2) and Eq. (3).

$$\frac{u}{u_\infty} = \frac{(1+da_1)\sqrt{da_3 \frac{x}{c}} \pm \left[da_2 + da_3 da_4 \frac{x}{c}\right]\sqrt{1-da_3 \frac{x}{c}}}{\sqrt{da_3 \frac{x}{c} + \frac{1}{4} da_1^2}} \tag{1}$$

$$(r_{LE}) = (0.5/da_3) da_1^2 (1-M_\infty^2) \tag{2}$$

$$\alpha = da_2 \sqrt{1-M_\infty^2} \tag{3}$$

Where u is the velocity in the upper and lower surface, u_∞ is the flow velocity without perturbation, r_{LE} is the leading edge radius, α is the angle of attack in radians, M_∞ is the Mach of the flow and x/c is the position in function of chord. The signal + represent the upper surface, the - signal represent the lower surface, and da1 ~ da4 represent variables that must be adjusted.

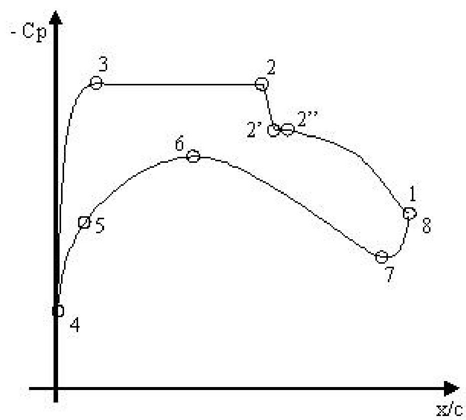

Figure 3. Schematic representation of target pressure distribution.

The rooftop region consists in the upper surface, from control point 3 to control point 2, and in the lower surface, from control point 5 to 6. The equation is detailed by Eq. (4).

$$M = db_1 + db_2 \left[(x/c) - (x/c)_{b_0}\right]^{1+db_3} \tag{4}$$

Where index b_0 refers to the starting point of the region and the parameters db_0 ~ db_3 are the design variables that must be adjusted.

The shock relation starts in control point 2 and ends in control point 2', shown in Fig. 3, is less than the Rankine-Hugoniot pressure jump. A modified Hankine-Hugoniot relation of the following form was used for approximation of pressure jump, detailed in Eq. (5).

$$\frac{p_{2'}}{p_2} = 1 + \frac{2\gamma}{\gamma+1} 0.65 (M_2^2 - 1) \tag{5}$$

The p is pressure and M_2 is Mach number in control point 2, detailed in Fig. 3.

The constant pressure region behind the shock wave is necessary to stabilize the boundary layer. This region starts in control point 2' and ends in control point 2'', in Fig. 3.

Pressure recovery region on upper surface starts in control point 2'' and ends in control point 1, detailed in Fig. 3. The formulation to model the pressure recovery is represented by an approximation of Stratford pressure distribution.

Pressure recovery region on lower surface starts in control points 6 and ends in control point 7, shown in Fig. 4. This region is modeled as a fourth-other polynomial, detailed in Eq. (6).

$$M = M_6 + dc_1\left(\frac{x}{c} - \frac{x_6}{c}\right) + dc_2\left(\frac{x}{c} - \frac{x_6}{c}\right)^2 + dc_3\left(\frac{x}{c} - \frac{x_6}{c}\right)^3 + dc_4\left(\frac{x}{c} - \frac{x_6}{c}\right)^4 \qquad (6)$$

Where the $dc_1 \sim dc_4$ are design variables that must be adjusted.

The rear loading region, limited by control points 7 to 8 in Fig. 3, is defined by a simple polynomial detailed in Eq. (7).

$$M = M_7 + dd_1\left(\frac{x}{c} - \frac{x_7}{c}\right) + dd_2\left(\frac{x}{c} - \frac{x_7}{c}\right)^2 \qquad (7)$$

Where $dd_1 \sim dd_2$ are design variables that must be adjusted.

It was simulated several airfoils in MSES and the velocity distribution were plotted. For each airfoil, the control point were not calculated by Campbell methodology, used in section 2.1. This is done to separate the errors due to control point methodology from velocity distribution methodology errors. Because of this, all control point parameters, location and level, were established by the designer. With the control point known the velocity distribution can be calculated.

In stagnation flow region, it is known the location and velocity level of control point 3, 4 and 5. With four equations, eq. (1) for upper surface, eq. (1) for lower surface, eq. (2) and eq. (3), and with fours variables da1 ~ da4 it is possible to find the results for the design variables.

Design variables from eq. (4), (6) and (7) were calculated by least-squares fitting technique.

Figure 4 shows velocity distribution simulated at MSES and the one calculated by methodology in a supercritical airfoil with 13% of *(t/c)max* at *CL* 0.60 and Mach 0.75.

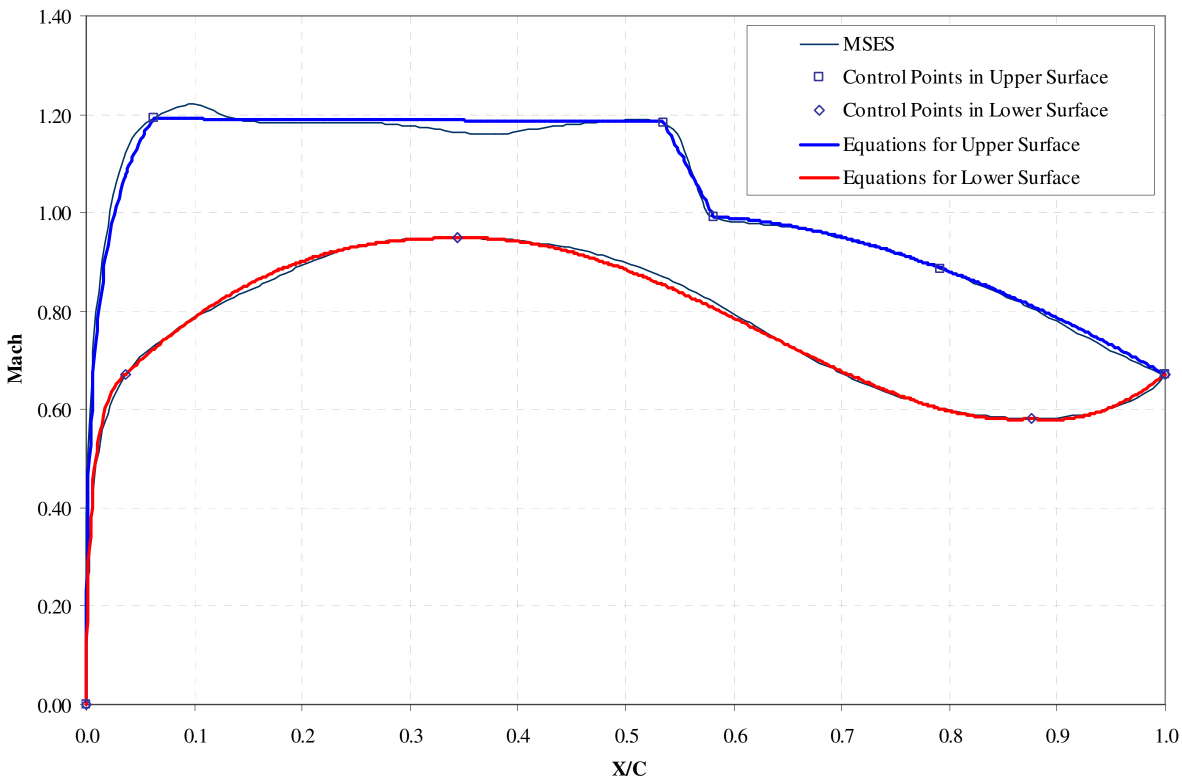

Figure 4. MSES results compared with Kim and Rho (1997) equations in an airfoil with *(t/c)máx* = 13%, at *CL* 0.60 and Mach 0.75.

As can be observed in Fig. 4 the results from the equations are very similar to CFD results.

Modifications in the upper surface and lower surface were done to verify if the equations could reproduce properly the velocity distribution and it was observed that the equations have similar behavior in comparison with CFD results, like in Fig. 4.

3. SUPERCRITICAL AIRFOIL DESIGN GUIDELINE

During experimental development of supercritical airfoils, some design guidelines recognized that yielded the best compromises in drag characteristics over a range of test conditions, as mentioned in Harris (1990). One of these guidelines proposes that the supercritical airfoil presents in a specific off-design condition a sonic-plateau pressure distribution over the upper surface. This off-design condition corresponds to a Mach number just below the design condition Mach and the off-design *CL* is lower than the design condition *CL* also. The off-design condition is sketched in Fig. 5 and the design condition is shown in Fig. 6.

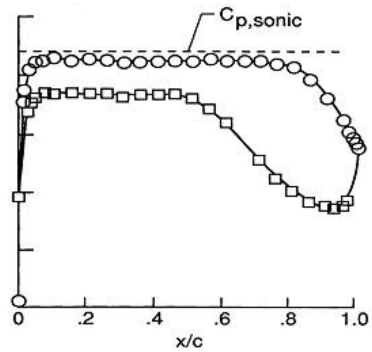

Figure 5. Sonic-plateau pressure distribution at off-design condition with Mach and *CL* lower than design condition for supercritical airfoils.

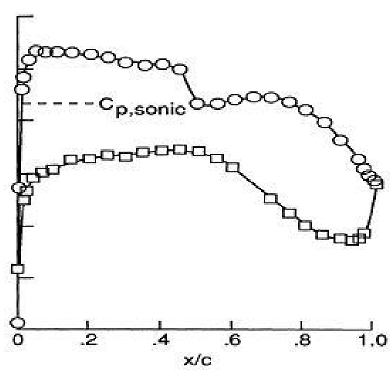

Figure 6. Pressure distribution at design condition for supercritical airfoil.

To evaluate the applicability of this guideline it was used a supercritical airfoil that was previously optimized at a design condition of Mach 0.75 and *CL* 0.60. At this condition, this supercritical airfoil presents 5.7 drag counts of *CDwave*. The off-design guideline will be evaluated and the parameter to confirm if the new airfoil is better than the baseline one will be the *CDwave*.

The off-design condition where the baseline airfoil pressure distribution presents similar behavior from Fig. 5 is at Mach 0.73 and *CL* 0.35.

At this off-design condition, Mach 0.73 and CL 0.35, it was made three modifications in the baseline airfoil pressure distribution detailed in Fig. 7. The first modification is plotted in red line and represents the sonic-plateau. The second modification represented in light blue line was done due to author experience in other projects. The fourth and last modification done in the baseline airfoil at Mach 0.73 is plotted in green color and was done to verify the new airfoil performance behavior due to peak increase.

With these three new pressure distribution plotted, it was used MDEP, the inverse design tool from MSES, to generate the new geometry that is responsible for each pressure distribution at this off-design condition. With the three new geometry defined, each one were simulated in MSES at design condition, Mach 0.75 and *CL* 0.60, to evaluate its pressure distribution and wave drag. The pressure distribution at the design condition, Mach 0.75 and *CL* 0.60, for each airfoil is plotted in Fig. 8. In Tab. 1 is listed the *CDwave* of each airfoil simulated to compare them with the baseline result.

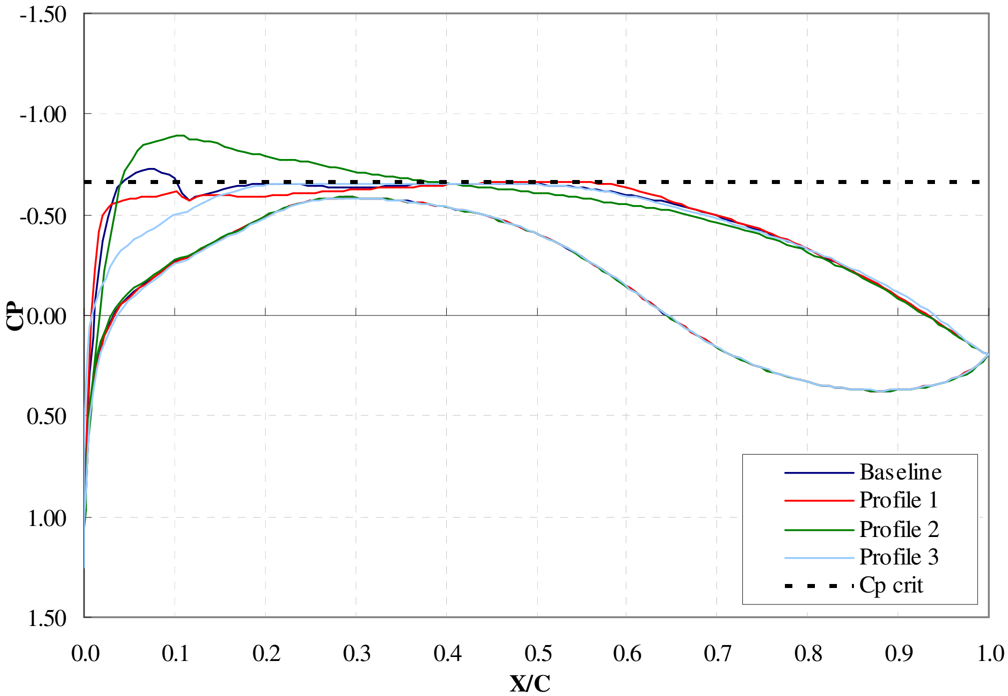

Figure 7. *Cp* versus *X/C* at Mach 0.73 for baseline airfoil and modified ones.

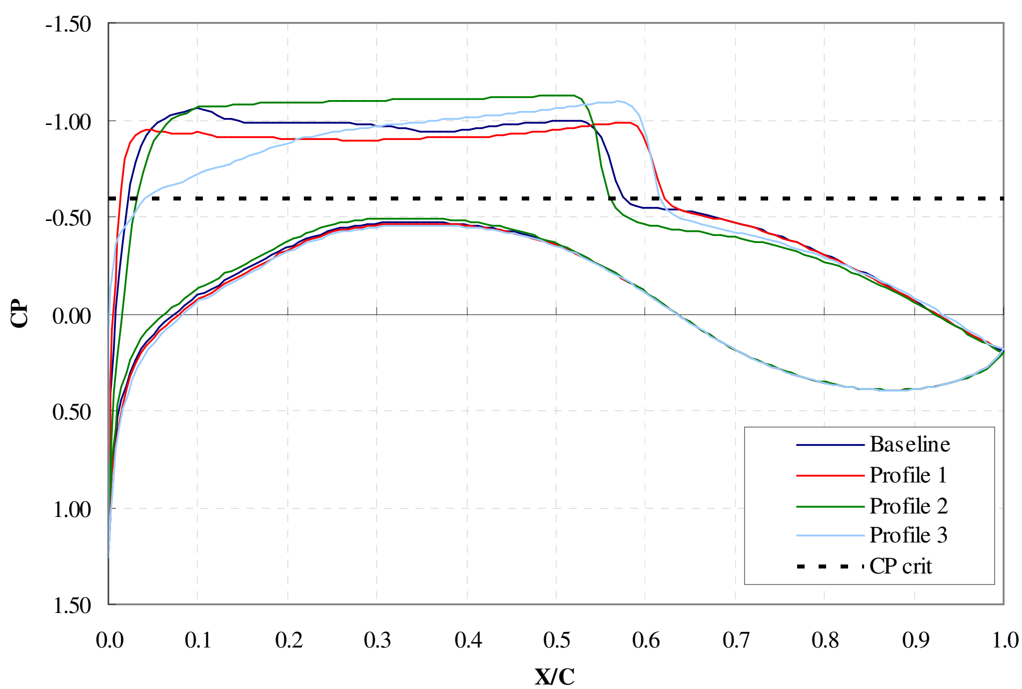

Figure 8. *Cp* versus *X/C* at Mach 0.75 and *CL* 0.60 for the four airfoils

Table 1. Wave drag results for the four airfoils at Mach 0.75 and *CL* 0.60.

Airfoil	CDw [drag counts]
Baseline	5.7
Airfoil 1	5.9
Airfoil 2	36.1
Airfoil 3	21.2

As observed in Tab. (1) the baseline airfoil presented the lower *CDwave* in comparison with the other airfoils. With this data, the guideline was evaluated and it didn't achieve lower *CDwave* than the baseline one.

4. CONCLUSIONS

It was validated semi-empiric formulation comparing them with CFD results, using MSES as tool. Besides, design guideline was applied in an airfoil that was previously optimized for a design condition and was evaluated its applicability.

The semi-empiric formulation to locate the control point from target pressure distribution, described in Campbell (1992), is inconsistent because it needs as input the *CL* and *CM* to locate the control points but the methodology results don't give the same *CL* and *CM*. The solution is to modify the *Cp* from control points 2 and 6 in the lower surface, detailed in Fig. 1, until reach the specific *CL* and *CM*, turning the methodology consistent and the control point more close to MSES results.

The formulations for velocity distribution between control points, detailed in Kim and Rho (1997), are very accurate and are very similar to CFD results.

The off-design guideline, described in Harris (1990), was evaluated in a supercritical airfoil that was previously optimized in a specific design condition and the guideline didn't achieve success of reducing the *CDwave*.

This study is still under development to understand the complete methodology applicability and study parametrically the inverse design.

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6. RESPONSIBILITY NOTICE

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