COMPARISON BETWEEN MODERN PROCEDURES FOR AERODYNAMIC CALCULATION OF SUBSONIC AIRFOILS FOR APPLICATION IN LIGHT AIRCRAFT DESIGNS

Luiz Augusto Tavares de Vargas
Federal University of Minas Gerais – Centre for Aeronautical Studies
luizatv@superig.com.br

Paulo Henriques Iscold Andrade de Oliveira
Federal University of Minas Gerais – Centre for Aeronautical Studies
iscold@ufmg.br

Ricardo Luiz Utsch de Freitas Pinto
Federal University of Minas Gerais – Centre for Aeronautical Studies
utsch@demec.ufmg.br

Marcos Vinícius Bortoulus
Federal University of Minas Gerais – Centre for Aeronautical Studies
borta@demec.ufmg.br

Marcos da Silva e Souza
Federal University of Minas Gerais – Centre for Aeronautical Studies
marcus3d@uol.com.br

***Abstract**: With the noticeable growth of computer calculation capacity, several procedures for calculation of aerodynamic characteristics of subsonic profiles have been developed in the past couple of decades. Taking into account the need for robust and fast procedures, such developments are usually based on solutions parallel to potential and viscous flow. The use of these procedures as a source of information for the development of light aircraft projects is still questionable, mainly due to lack of careful comparison to experimental results. This paper presents and comments the main models found in literature, developed by several authors, among them Moran, Eppler, and Drela. The results obtained through these procedures for laminar and turbulent profiles in common conditions for light aircraft design are compared to experimental data obtained in wind tunnel, also reported in literature. The conclusions obtained through this comparison point towards the procedure which offers the most adequate and trustworthy result in regards to needs present in the design of light aircraft.*

***Keywords:** aerodynamics, airfoil, aircraft design*

1. Introduction

It is well known that the performance and flight characteristics of an aircraft are considerably affected by their aerodynamic design (Anderson 1991), which can currently be done with the aid of various tools. Among these, wind tunnel experiments and computerized simulations can be mentioned.

Tests in wind tunnels, though more trustworthy in their results, are still procedures that take time and present high costs. Computerized numerical methods, on the other hand, allow for faster analysis with lower costs; however, the trustworthiness of their results is still in question.

Therefore, the present work aims to analyze results obtained with several computerized procedures for the analysis of bi-dimensional flow on aerodynamic profiles and compare these to experimental results available in literature. The originality proposed consists in executing such comparison in regards to the needs present in the design of aircrafts, in other words, the results of interest are not details concerning of the flow on aerodynamic profiles, but the result of forces and moment which act on these airfoils.

2. Computerized procedures for analysis of aerodynamic profiles

Taking into consideration that the object of this work concerns activities in aircraft design, an analysis procedure is considered efficient when, the results are not only coherent with experimental measurements, but they are also robust and fast, and applicable to a large variety of cases.

According to Drela and Giles (1987) procedures for calculation of existing airfoils can be divided into:

- Algorhythms which resolve the complete flow (Navier-Stokes equations)
- Algorhythms which solves, separately, the viscous and inviscid regions (based on the boundary-layer theory).

In their paper, Drela and Giles also state that methods based on Navier-Stokes equations are slower and do not show advantages in regards to precision of results when compared to methods that resolve, separately, viscous and inviscid regions. Therefore, it has become consensus that methods which resolve viscous and inviscid regions separately are more adequate as tools for analysis of airfoils.

It is also consensus that the use of Boundary-Element Method (Panel Method) for the solution of inviscid regions of flow provides precise and trustworthy results. However, for solution of viscous regions and their interaction with non-viscous areas, there are still questions concerning the best method to be used.

According to Moran (1984), the solution of the limit layer of an aerodynamic profile can be done by two formulations:

- Differential formulation: in which simplified Navier-Stokes equations are resolved through numeric techniques, as finite differences, resulting in a detailed profile of the Boundary-Layer.
- Integral formulation: in which simplified Navier-Stokes equations are resolved with the aid of pre-established Boundary-Layer profiles obtained, usually, through experimental data.

Moran states that the integral methods have simpler solutions, are faster and have good precision, making them the first choice for analysis of the flow around the airfoils. Only in tridimensional flow or in cases where integral methods are not appropriate, the differential Boundary-Layer should be used.

According to Dini, Coiro and Tangler (1992) the existing relations between solutions of viscous and inviscid regions of the flow can be divided into:

- Weak interaction: also known as direct method, where the invisid region of the flow is calculated and then the viscous region is calculated separately, using the result of the inviscid region as boundary condition.
- Strong interaction or fully coupled: where formulations for inviscid and viscous regions are simultaneously solved, where one result interferes in the result of the other, and this interaction occurs until there is convergence for a result that satisfies both formulations (viscous and inviscid).

In this context, literature shows that most current work in development of computerized procedures for analysis of aerodynamic profile aim towards the development of new techniques for treatment of interaction between viscous and inviscid regions of the flow.

3. Inviscid region: Panel Method

As mentioned earlier, the most common technique for calculation of the inviscid region (also known as potential region) of airfoils is the Panel Method.

It is a boundary-element method, which had its development triggered by the work of Hess and Smith (1966) which in these days has a series of variation seeking robustness and speed in solutions.

The formulation of the panel method consists in the resolution of Laplace's equation (Eq.(1)) through the superposition of simpler solutions of elementary flows distributed throughout the body. This characteristic makes the method fast, because it is not necessary the discretization of all flow domain.

$$\frac{\partial^2 \phi}{\partial x^2} + \frac{\partial^2 \phi}{\partial y^2} = 0 \tag{1}$$

Where ϕ denotes the velocity potential. The elementary flows used in the formulation of the panel method are sources, vortices or doublets, with single strengths which, when added, attend to the pre-established boundary conditions of impermeability (no normal velocity) and Kutta's condition (velocity in the trailing edge equals in upper and lower surfaces). The final problem to be solved is a linear equation system according to Eq. (2).

$$[A] \times [q] = [b] \tag{2}$$

Where $[A]$ denotes the matrix of coefficients of influence; $[q]$ the intensities of elementary flows and $[b]$ the boundary condition. Notice that since only the body surface is discretizated, the number of variables is reduced.

4. Viscous Area: Integral Methods

The simplified Navier-Stokes equations which are used in the solution of the boundary layer in the integral form are known as Von Kármán equations and Energy Equation, according to Equations (3) and (4) respectively.

$$\frac{d\delta_2}{d\xi} + \frac{\delta_2}{V_e}\left(2 + H_{12}\right)\frac{dV_e}{d\xi} = \frac{c_f}{2} \tag{3}$$

$$\frac{d\delta_3}{d\xi} + 3\frac{\delta_3}{V_e}\frac{dV_e}{d\xi} = 2c_d \tag{4}$$

Where: ξ denotes the coordinate system tangent of the body surface, V_e the velocity in the boundary layer edge, δ_2 the momentum thickness, δ_3 the energy thickness, H_{12} the shape factor defined as $H_{12} = \frac{\delta_1}{\delta_2}$, c_f the skin-friction coefficient, c_d the dissipation coefficient.

In order to solve Equations (3) and (4) numerical techniques are used together with experimental relations for the shape factors, skin-friction and dissipation coefficients, all dependent on flow characteristics.

Many mathematical models and experimental relations were developed in the past few decades seeking the solution of the Von Kármán and Energy Equations (Moran 1980). Each of these models is applicable to one particular kind of flow and to this day much research has tried to develop new models for specific situations (Dini, Coiro and Bertolucci 1995). It is important to keep in mind that the flow in aerodynamic profiles develops from the leading edge to the trailing edge in three distinct regions, being those: i) laminar region ; ii) transition region e iii) turbulent region. Therefore, for each of these regions there are distinct mathematical or experimental models for boundary-layer. Particularly the transition region is not completely modeled, being considered only with the objective to alter the mathematical model according to the flow characteristics (laminar or turbulent). For the calculation of airfoils, the integral mathematical models most used in literature are:

i) Proposed by Eppler and Somers (1980);
ii) Proposed by Drela and Giles (1987);
iii) A combination proposed by Moran (1980) which uses the laminar boundary layer formulation of Thwaites (Thwaites 1960), the turbulent boundary layer relations of Head (Cebeci and Brandshaw 1977) and the transition criterion proposed by Michel (Michel 1951).

Throughout of the flow on airfoil, other phenomenon which considerably affect the development of limit layer may occur, for instance, the separation of the flow before the trailing edge, or the separation of the flow followed by it's reattachment, phenomenon known as laminar separation bubble. The occurrence of these phenomenon may be considered the cause of the non-linearity of the profile lift curve, and it's preview is important, especially when the goal is the design of aircrafts.

According to Eppler and Moran, their models are valid only for the flow attached while according to Drela and Giles, their relations are valid also for the cases in which there is separation of the flow or laminar separation bubbles, which makes their code more robust.

The weak interaction codes cannot work with separated flow and force the transition in case of laminar separation and interrupt the calculation of the turbulent boundary layer when turbulent separation is foreseen. These deficiencies cause two effects in the calculation of airfoil characteristics: i) the calculation of airfoil drag is incomplete and ii) the non-linearity of the lift curve is not possible to preview. Aiming to curb the second deficiency, Eppler suggests an additional model able to estimate the effect of turbulent separation in the profile lift curve.

5. Softwares for calculation of aerodynamic profiles:

Aiming towards the analysis of the main formulations for analysis of aerodynamic profiles, the obtained results with all three mathematical models mentioned earlier in this paper (Moran, Eppler and Somers, Drela and Giles) are confronted with experimental results available in literature. Commercially available software will be used, chosen according to the following criteria:

- Proper software documentation with description of used models
- Available for free
- Version compatible with IBM-PC system
- Faithful representation of the originally proposed method

The chosen softwares contemplate the models and methods according to Table 1.

Table 1. Software used and their models and methods

Software	Panel Method	Laminar Model	Transition criterion	Turbulence model	Interaction	Separated flow
Eppler e Somers	Linear vortex	Eppler e Somers	Eppler e Somers	Eppler e Somers	Weak	Valid*
Moran	Linear vortex	Thwaites	Michel	Head	Weak	Not valid
Drela e Giles	Linear vortex	Drela e Giles	Drela e Giles	Drela e Giles	Strong	Valid

* with additional model

In all cases, the available corrections for compressibility and surface finish were not used.

6. Results

In order to determine the best model for calculation of airfoils in regards to aircraft design, some aerodynamic characteristics of some typical airfoils used in light aircraft design were determined, through each of the chosen softwares. These results were then confronted with experimental results available in literature and each software was scored according to the accuracy of their results in comparison to experimental results. This score was defined as shown in equation (5).

$$S = \left(\frac{\sum_{i=1}^{n} (Y_i - \bar{Y}_i)^2}{n} \right)^{-1} \qquad (5)$$

Where S denotes the score, Y denotes the experimental value of the coefficient in question and $\bar{Y}$ denotes the value of the coefficient calculated numerically.

The airfoils used in the tests are subsonic and with an ample use of the project in light aircraft in general aviation, including symmetric, non-symmetric, turbulent and laminar airfoils. Therefore, the analyses airfoils were:

- NACA 0012,
- NACA 4415,
- NACA 23015,
- NACA 63_2415,
- NLF(1)-0215F.

For all numeric tests and for obtention of experimental information it was determined that Reynolds number equaling 3 million, null flap deflection and smooth surface finish. The geometry of the profiles in each software was generated from the same coordinates and using the same number of panels. Figures 1 to 5 show the aerodynamic polars of the analyzed airfoils. In each polar curve the obtained results with each algorithm are shown, as well as the experimental value available in literature. Based on the polar curves presented in figures 1 to 5, the secondary aerodynamic characteristics were determined, but what is of fundamental importance for aircraft design are: inclination of the lift slope and the zero-lift angle. The values obtained through numerical analysis and through experimental polars are presented in Table 2 and 3.

Table 2. Lift slope (1/rad)

		Moran		Drela e Giles		Eppler e Somers	
	Literature	Value	Error	Value	Error	Value	Error
NACA 0012*	6.42	6.89	0.47	6.35	-0.06	6.86	0.45
NACA 4415*	5.73	7.05	1.32	6.41	0.68	7.02	1.29
NACA 23015*	6.07	7.05	0.98	6.36	0.29	7.04	0.96
NACA 63 2 415*	6.88	6.94	0.07	6.67	-0.20	6.91	0.04
NLF(1)-0215F**	6.45	6.93	0.48	6.73	0.28	6.80	0.35

*Abbot (1959)
** Somers (1959)

Table 3. Zero lift angle (degrees)

		Moran		Drela e Giles		Eppler e Somers	
	Literature	Value	Error	Value	Error	Value	Error
NACA 0012*	0.00	0.00	0.00	0.00	0.00	0.00	0.00
NACA 4415*	-4.00	-4.30	-0.30	-4.36	-0.36	-4.50	-0.50
NACA 23015*	-1.08	-1.18	-0.11	-1.21	-0.14	-1.67	-0.60
NACA 63 2 415*	-2.83	-3.25	-0.42	-2.99	-0.16	-3.13	-0.29
NLF(1)-0215F**	-5.74	-6.35	-0.61	-5.61	0.13	-6.33	-0.59

*Abbot (1959)
** Somers (1959)

Each software's score, obtained as described earlier, for the lift, drag and moment polar is presented (normalized value) in Table 4 and represented as graphs in figures 6 to 8.

Table 4. Normalized Score for each software

	CL			CD			CM		
	Moran	Drela e Giles	Epler e Somers	Moran	Drela e Giles	Epler e Somers	Moran	Drela e Giles	Epler e Somers
naca 0012*	0.110	0.423	1.000	1.000	0.286	0.290	1.000	0.324	0.477
naca 4415*	0.195	1.000	0.634	1.000	0.302	0.207	0.911	1.000	0.358
naca 23015*	0.202	1.000	0.788	1.000	0.377	0.511	1.000	0.332	0.101
naca 63 2 415*	0.062	1.000	0.587	0.299	0.223	1.000	1.000	0.790	0.447
NLF(1)-0215F**	0.077	1.000	0.164	1.000	0.566	0.364	0.004	1.000	0.003

*Abbot (1959)
** Somers (1959)

7. Discussion of Results

In Table 2, regarding the lift slope, a slight advantage can be noted for the Drela and Giles model; while Table 3, referring to the determination of null lift angle, shows equilibrium between softwares with slight disadvantage for the Eppler and Somers model. This leads to the conclusion that the potential formulation of the three algorithms has similar precision levels in linear region of the lift curve, which was expected since they use the same potential formulation of for panel methods. Table 3 clearly shows a great disadvantage for the lift coefficient obtained by the Moran algorithm. The linear region results were satisfactory, so the poor score can be attributed to the non-linear region. This effect was expected for this model, because it is not able of analyzing areas where there is separation of the flow. In contrast, the Drela and Giles method is the one with the best results in the non-linear region of the lift curve, which is probably due to the strong interaction between viscous and inviscid models.

The determination of moment coefficient curves in the linear area is reasonable for all models, but in the non linear area the precision of models reduces considerably in relation to the separation of the flow, because any small imprecision in pressure distribution on the airfoil causes great variations in moment. A noteworthy situation is the curve of moment coefficient obtained by the Drela and Giles method on the profile NLF(1)-0215F.

In regards to determination of drag coefficient, is notable the advantage of the Moran model.

8. Conclusions

Trough the comparisons between the many available numeric models in literature for the calculation of airfoils, is possible to note that potential formulation of panel methods with linear vortices presents good results in the linear part of the lift curve, both in weak and strong interaction models.

In the non-linear region, the model with strong interaction (Drela and Giles) does not require an additional method for analyze separated flow, as do in the weak interaction models.

The model proposed by Moran (Thwaites, Head, and Michel) presents good performance for drag calculation of the chosen airfoils.

Therefore there is not a single model which can be named as the best for analyze an airfoil, because each model's trustworthiness varied according to: i) angle of attack, ii) particular aerodynamic characteristics of each airfoil and iii) coefficients of interest.

During aircraft design tasks, the airfoil analyze by numerical models should be done with caution and awareness. Must be noted that, normally, the results could be questionable, and special strategies must be used in order to assure the foreseen airfoil performance.

Further tests are necessary in order to determine the best method for each family of airfoils with similar aerodynamic characteristics. Other tests with different Reynolds numbers, flap deflections and surface finish are also interesting in order to amplify the knowledge about these numerical models in order to be used in the complete aircraft design.

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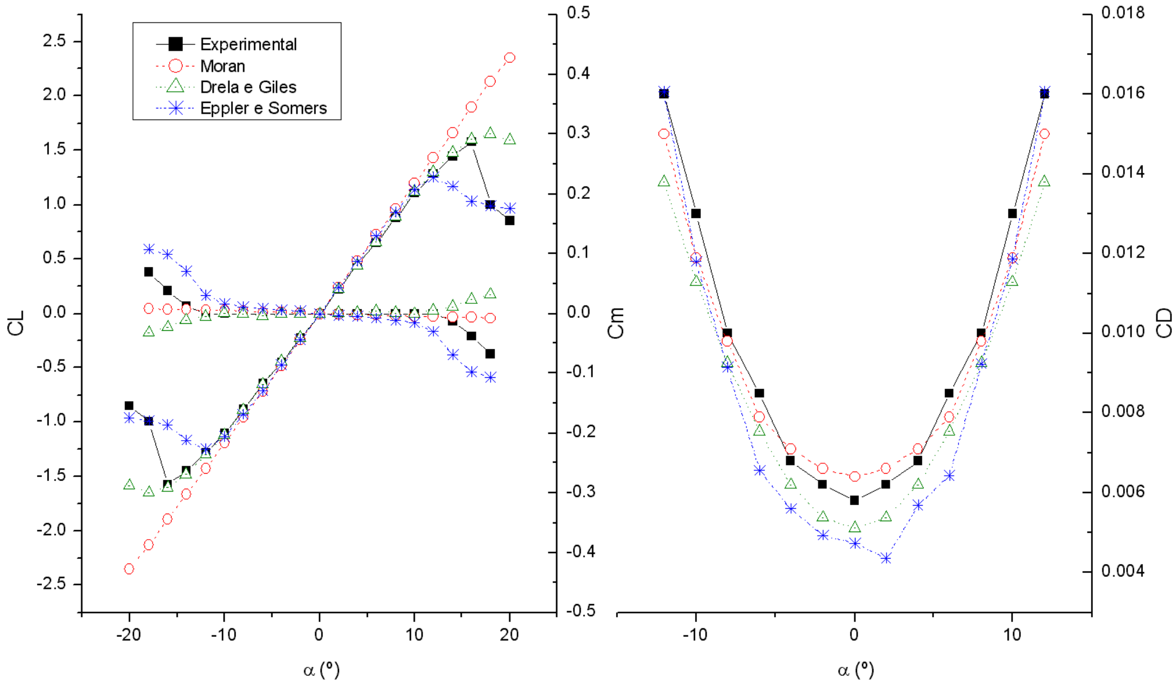

Figure 1. NACA 0012, Re $3x10^6$.

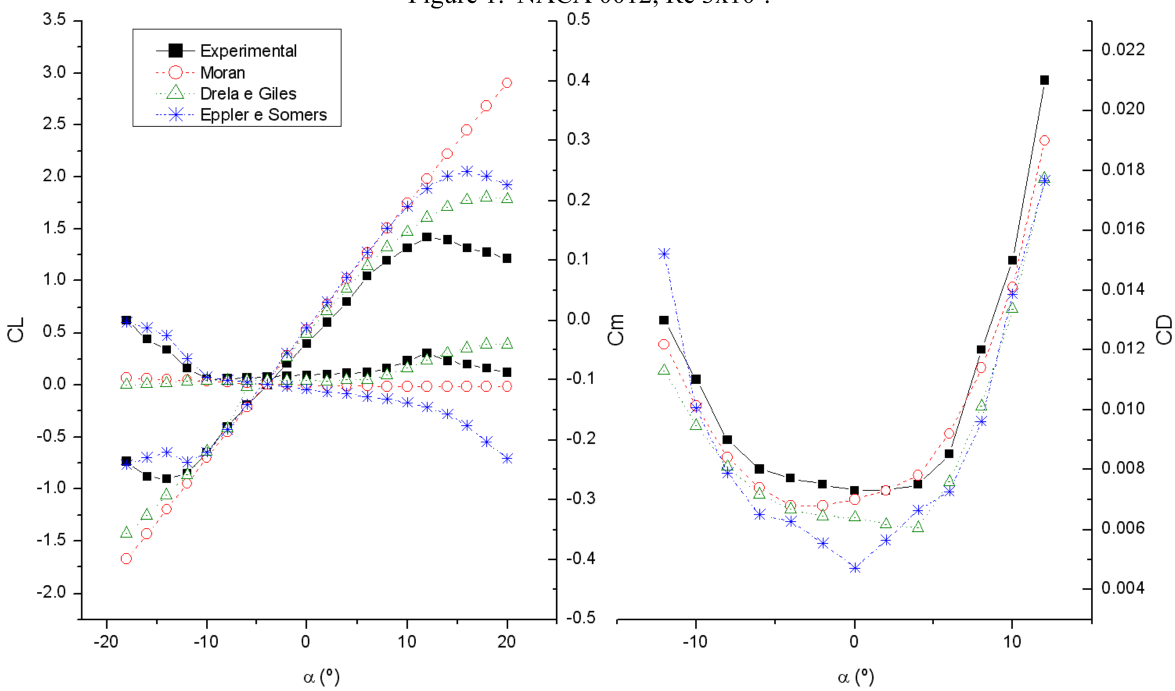

Figure 2. NACA 4415, Re $3x10^6$.

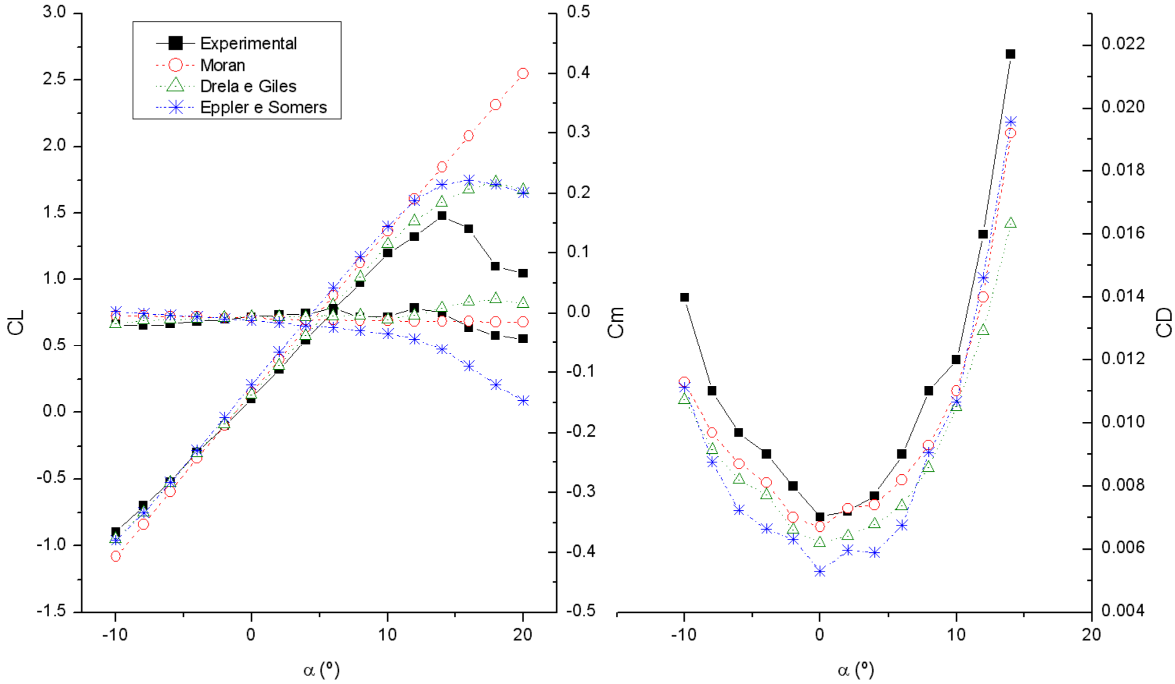

Figure 3. NACA 23015, Re $3x10^6$.

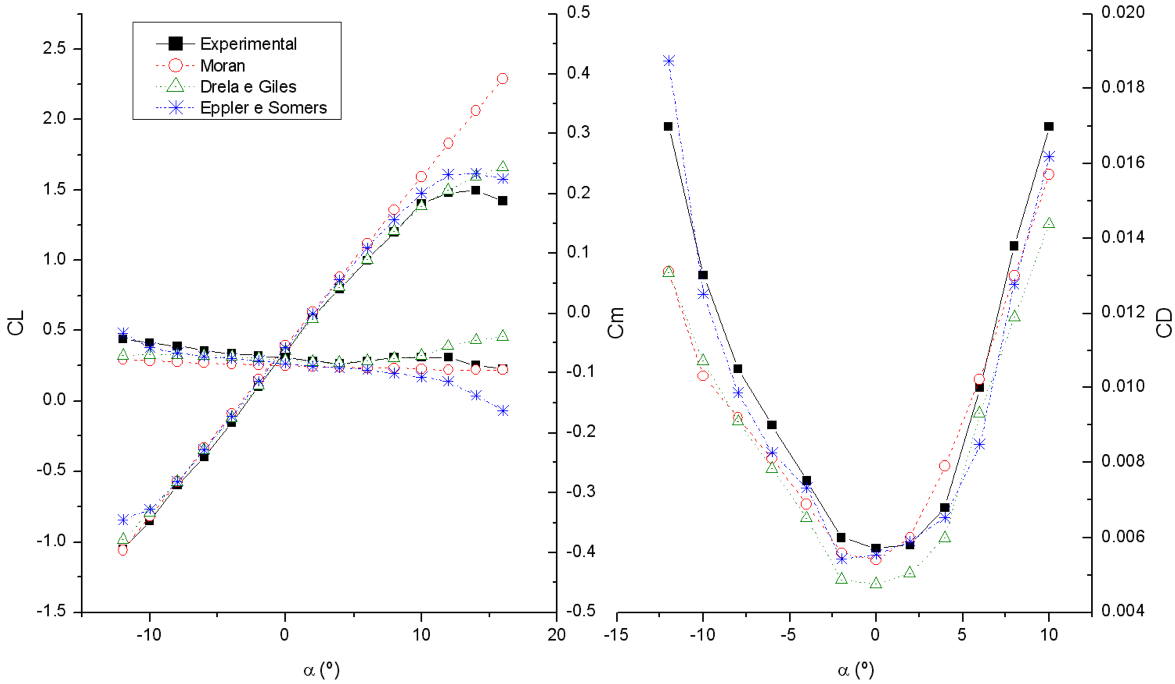

Figure 4. NACA 63_2415, Re $3x10^6$.

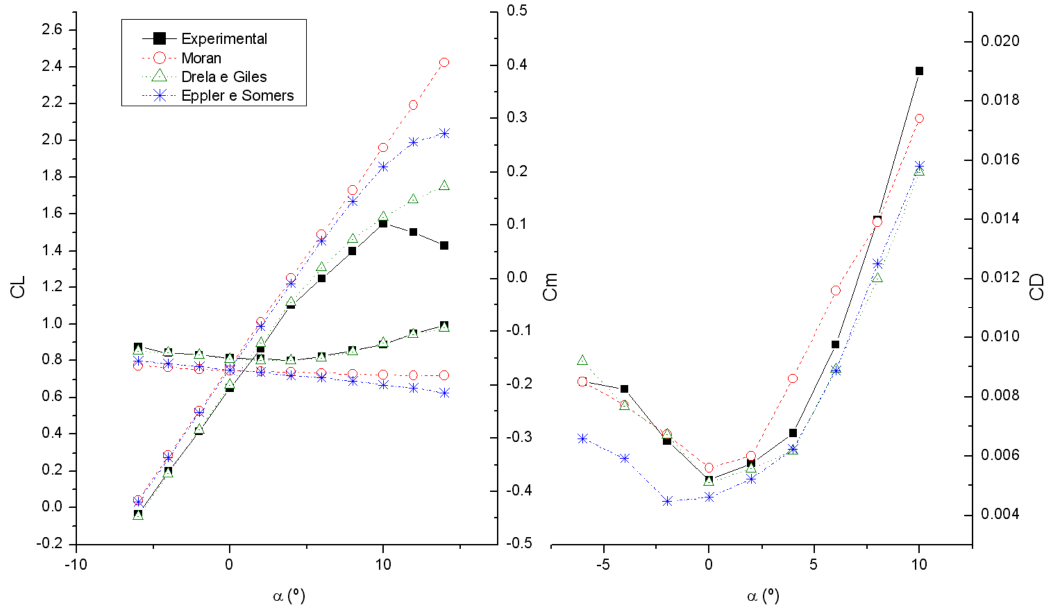

Figure 5. NLF(1)-0215F, Re 3x10^6.

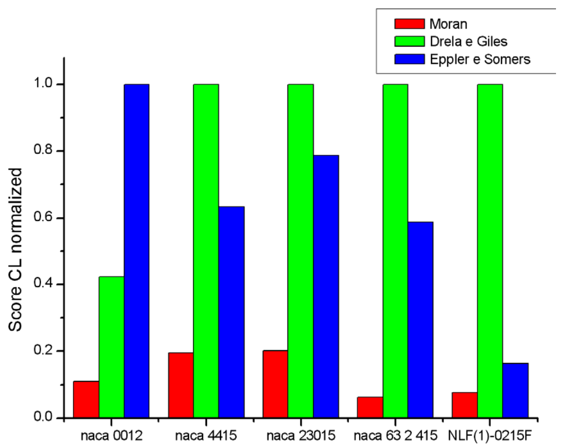

Figure 6. Normalized score for the lift coefficient

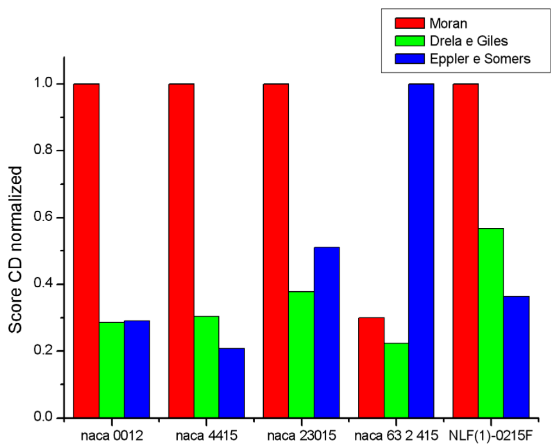

Figure 7. Normalized score for the drag coefficient

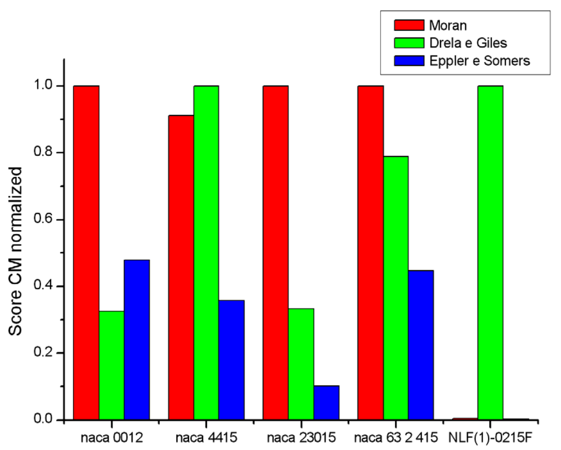

Figure 8. Normalized score for the moment coefficient