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Aspects of the Application of a Three Dimensional Panel Method Tool in the Design of a Light Aircraft

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ABSTRACT

This paper presents the computer implementation process of the Panel Method for aerodynamic analysis of three dimensional bodies with or without lift. The computer implementation consists in the creation of pre and post processing environments and a solver for the solution of the problem in external flow potential over three dimensional bodies with and without lift. For cases in which there's no lift, a constant distribution of sources over the body (fusaleges, farings, etc.) is used, their intensities calculated in order to guarantee impermeability of the body. For cases in which there is lift, a constant distribution of sources and doublets are used on the body (wings, tails, etc.). The intensity of the sources is solved in the same manner as cases with no lift. Intensity of doublets is solved in order to satisfy the equality condition regarding pressure on the under and upper side of the trailing edge (Kutta Condition) of the body without lift. An analysis of the aircraft CEA-308 is shown in order to illustrate the use of the developed program in light aircraft projects.

INTRODUCTION

The computer tools for aerodynamics have progressively increased their capacity and complexity. Currently, it is possible to study cases where the viscosity effects are relevant or where the dynamic effects have intense actions. However, for classic aerodynamics of subsonic aircrafts, the potential studies are still quite useful (Busch, 1991; Garrison, 1998; Garrison, 1996; Lednicer, 1997; Lednicer, 1999), especially regarding concepts which pertain to flow over the aircraft in question.

The Center for Aeronautical Studies (CEA) of the Federal University of Minas Gerais has made use of light aircraft design and construction for the teaching of Aeronautical Engineering. In this process, the use and development of computer tools for analysis of its studies has been an important form of perfecting the education of the students who frequent this university. In particular, the development of a computer tool for aerodynamic analysis of complete aircrafts has been an important objective.

This paper presents the application of a computational tool (fully developed in CEA), capable of solving the problem of potential three dimensional flows over aircrafts using panel methods, in the aerodynamic design of a light aircraft. The considerations taken into account for calculations are presented as well as a series of analysis done for the CEA-308 aircraft project, the last aircraft that was fully designed and built by students of this center.

A BREFILING DESCRIPTION OF THE SOLUTION METHOD

For the solution of the potential flow problem a numerical procedure will be used, consisting of division of the body surface into a finite number of panels, distribution on each panel of a flow singularity, solution of the intensity of these singularities in order to obey the surrounding conditions and, then proceeding to the numerical integration.

Three Dimensional Bodies without lift

In this paper, (Hess, 1966) the division of the surface of a body without lift (fuselage, farings, etc.) is done through quadrilateral flat panels that possess the coordinates of their vertices on the real surface of the body. The organization of the panels that compose a body is free and the central point is taken as a control point. Over each panel a constant unitary distribution of sources is applied.

Three dimensional bodies with lift

The three dimensional bodies with lift should be treated in a similar fashion as are those without lift. Some differences should be noted:

Initially, the first difference between the division of bodies with and without lift is that, in the case of those with lift, one must also divide the wake, which starts at the body trailing edge.

The surface of bodies with lift is divided also by quadrilateral panels with vertices that coincide with the real body surface. However, special attention should be given to panel organization. In reality, the coordinates for all panel vertexes are organized and stored in a group of lines, namely N-lines. The N-lines are lines that describe the section of the body and wake set, containing points that represent the vertices of each panel. All the N-lines possess the same number of points and are also organized from right to left of the body. Therefore, if two consecutive points of N-lines are taken, one has the vertices that form a panel. The panels between two N-lines form a set named lifting strip.

The center of each panel is also considered a control point. Therefore, the panel center of a lifting strip will be, approximately, on the medium line between the two lines that compose each strip.

The panels that compose the body with lift receive, similarly to the panels of bodies without lift, a constant unitary distribution of sources. In order to comply with the condition of having constant pressure as the wake is crossed, the panels that compose the wake do not receive source distribution. Additionally, the body with lift must also receive vortex type singularities. In this work was chosen to use vortex distribution on panels of the body with lift and its wake. In order to simplify the issue, intensity of distribution of vorticity for each panel was considered to be constant, both in chord wise and in span wise directions.

In order to obtain a constant vorticity distribution on each quadrilateral panel, was choose to make use of a doublets distribution over the surface. This distribution has the advantage of being scalar, and generating potential flow no matter what the form.

A fact that must be mentioned is the impossibility of true application of the physical reality of the Kutta condition, which is finite velocity in the trailing edge, in numerical procedures. If it were possible to obtain an explicit analytical experience, the appropriate parameters could be adjusted to eliminate the singular terms of the expression for velocity in the trailing edge. However, in a numerical procedure, a condition of finite velocity without specification of its value does not determine a solution for the problem.

Therefore, what one does is specify another property in order for the flow to, indirectly, comply with Kutta's condition. Among the many properties that can be defined for this condition to be met, a few can be singled out (Hess,1972): a) A stream surface leaves the trailing edge with a known direction or at least an estimated one; b) On the limit of the trailing edge, the pressures on the upper and under surfaces tend towards a common limit; c) The density of sources on the trailing edge is zero.

Pondering the characteristics of each condition and following some tests was chose to use the condition (b) in order to guarantee compliance to Kutta's condition. Despite the non-linearity which must be treated (due to the fact that pressure values are involved), the reduction of dependence with geometrical precision and the discretization of the wake was an important aspect for its choice..

The issue to be dealt with is still the wake position in the flow near the body. In these calculations was used a fixed wake leaving the body trailing edge and extending five chord away from the body.

ANALYSIS OF THE AIRCRAFT CEA-308

The aircraft CEA-308 (Figure 1) is a Project developed by the Center for Aeronautical Studies of UFMG (CEA) which has, as its main objective, the maximization of maximum velocity in level flight (Oliveira, 1999). Maximum weight at take off is limited to 300kgf and its propulsion system must be a conventional engine driving a propeller, in order for the aircraft to be inserted in the FAI C1-a0 class of the Federation Aeronautique Internationale. With this objective and these restrictions, it is noted that the aerodynamics characteristics of this aircraft must be very well established in order to decrease its aerodynamic drag.

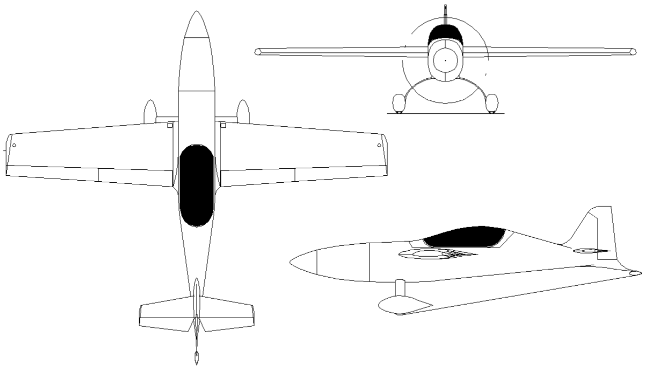

Figure 1 –CEA-308 Aircraft

When this aircraft was designed, the program presented in this paper was not operating yet, so the decisions made during the design were based on information obtained in literature or as a fruit of the experiences acquired during the design and construction of the other aircrafts developed at CEA. Currently, CEA-308 is in operation, therefore,, what is intended is a validation of the concepts used for the design of this aircraft through the qualitative analysis of results supplied by this program.

MODELING

The model built for the aircraft CEA-308 (Figure 2) is a simplification of the real geometry of the aircraft.

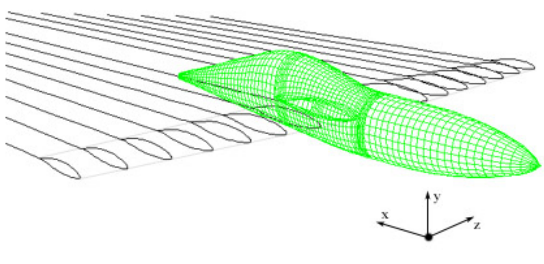

Figure 2. General view of a tipical model of the aircraft CEA-308

The main simplification is the exclusion of the vertical and horizontal tail. This simplification is justified by the fact that the results of this area are of little importance for the analysis that is intended. Furthermore, the results supplied by the program, for this area, are not very trustworthy since they do not calculate the true wake generated by the wing.

Although it is an aircraft moved by a propeller, the model does not take into consideration the influence of this component in flow. Doubtless, its influence is significant in this case; however, for a preliminary analysis, a model without the influence of the propeller is considered satisfactory.

Further details such as air escape for radiators, landing gear, and other excrescences were also excluded once they were considered, by the author, as having little relevance to the results.

Three different models were built, they are:

- One model with the intersection faring between the wing and the body (Figure 4), representing the real aircraft.
- One model without the intersection faring between the wing and the fuselage (Figure 4)
- One model (without the faring mentioned above) with the 200mm canopy pushed forward (Figure 5).

DISTRIBUTION OF PRESSURE ON THE AIRCRAFT

One may consider that the main information supplied by the program at hand is the distribution of pressure on each component of the aircraft. With the help of these results, the designer may be assisted in many areas of aircraft design, from modifications in the shape allowing for aerodynamic improvements to the calculation of aerodynamic pressure that occurs in the aircraft.

These pressure distributions may be presented in many forms, depending on what one wishes to observe. In this paper, these distributions are presented in two forms: i) three dimensional view of the aircraft covered with a spectrum of colors representing pressure levels and ii) pressure curves throughout predetermined paths on the aircraft.

<u>Pressure Spectrum on the aircraft</u>

The pressure spectrums are the most common way to initiate na analysis of results supplied by a program that utilizes the panel method. When considered quantitatively, the information is not very precise, on the other hand, qualitatively, this is the best way to visualize flow behavior. Figure 6 present examples of the pressure spectrum on the aircraft CEA-308 in three attack angles (0°, 3°, 6°)[1]

<u>Pressure curves on the aircraft</u>

The pressure curves on the aircraft are a much more precise form of information regarding the flow in question. With pressure curves graphics on can obtain a precise idea of pressure gradients throughout established directions on the aircraft. The graphics in Figure 7 present the pressure curves on the fuselage of the aircraft CEA-308.

The pressure curves on the wing, in the subsonic condition, do not supply information as conclusive as do the pressure curves on the fuselage. Pressure curves that extend in the direction of the chord are quite useful for the design of aerodynamic profiles; however, it is preferable to make such analysis with the use of bi-dimensional models that are simpler, but much more powerful than their three dimensional counterparts. The pressure curves that extend towards the wing span are not conclusive. In this case it's much more interesting to plot the lift distribution curves throughout the wing span. In order to do this, one must apply the integration of the distribution of pressure throughout the chord for each band of lift in the model, obtaining the local lift coefficient per unit of length in each band. Figure 3 presents the lift distribution on the wing of the aircraft CEA-308 for the three calculated attack angles.

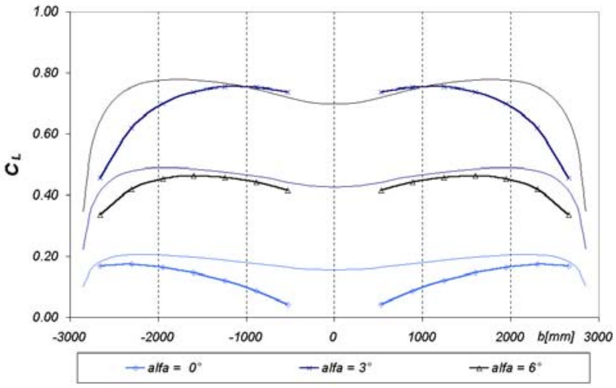

Figure 3. Lift distribution throughout the wingspan

Note that the thin lines represent the distribution of lift which was calculated by the Multhopp Method, without considering fuselage interference and influence.

The total lift coefficient values (only for the wing) are presented in Table 1.

Table 1 – Comparision between the lift coefficients of the wing calculated by the Multhopp Metod and the Panel Method

Attack Angle	Multhopp	Panel Method
0°	0.18222	0.10886
3°	0.45198	0.37592
6°	0.72174	0.60360

Note that the values obtained by the Multhopp method are always superior to those calculated by the program in question (Panel Method). This difference can be justified by the dismissal of fuselage interference when using the Multhopp Method.

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[1] These attack angle values were chosen since they are within the linear portion of the curve lift x attack angle of the aircraft.

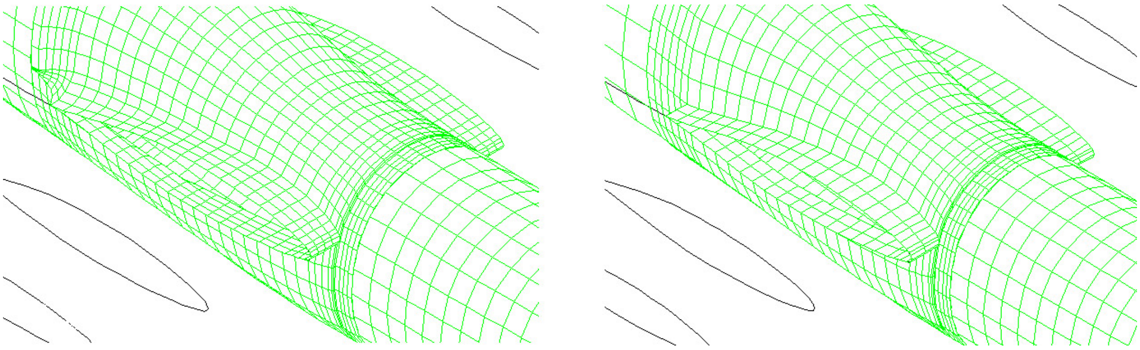

Figure 4. Difference between the model with and without the wing-body faring

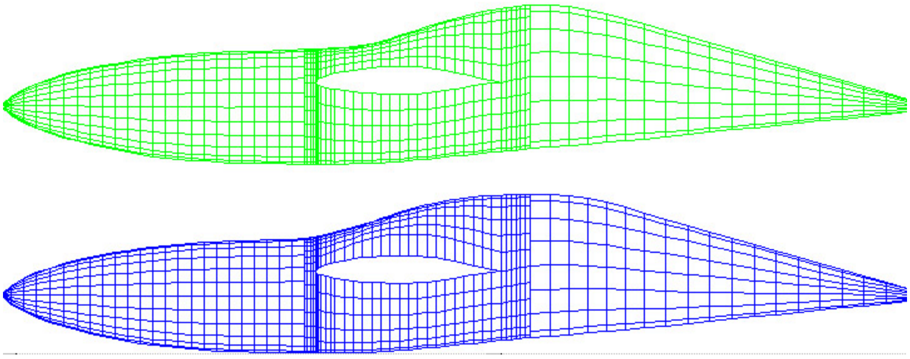

Figure 5. Difference between the normal model and the model with a forward canopy.

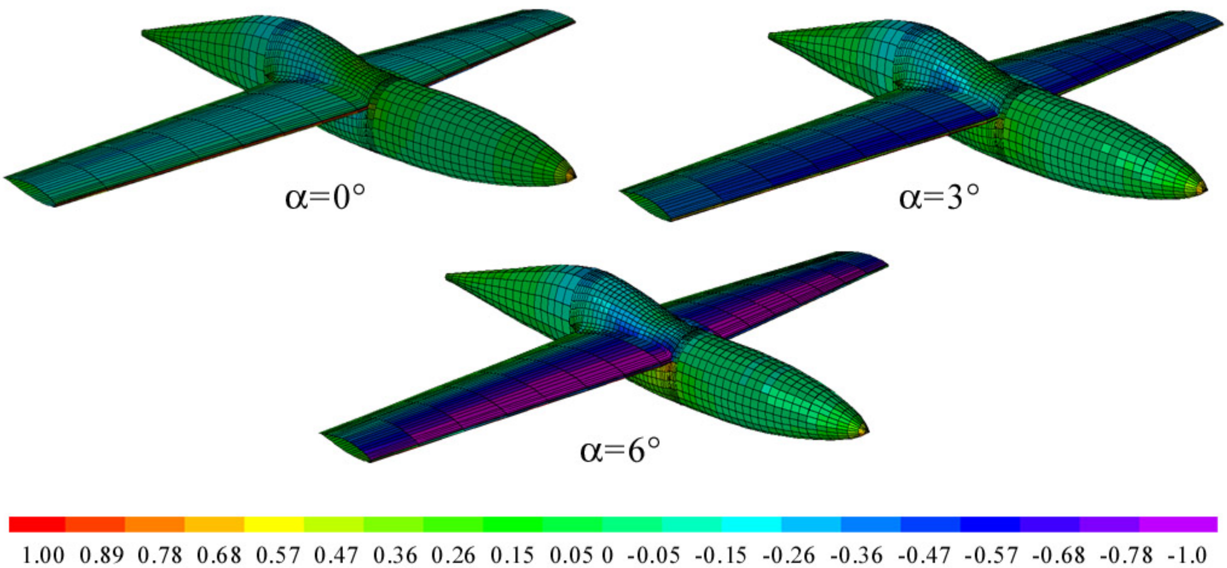

Figure 6. General inferior view of the spectrum of pressure distribution

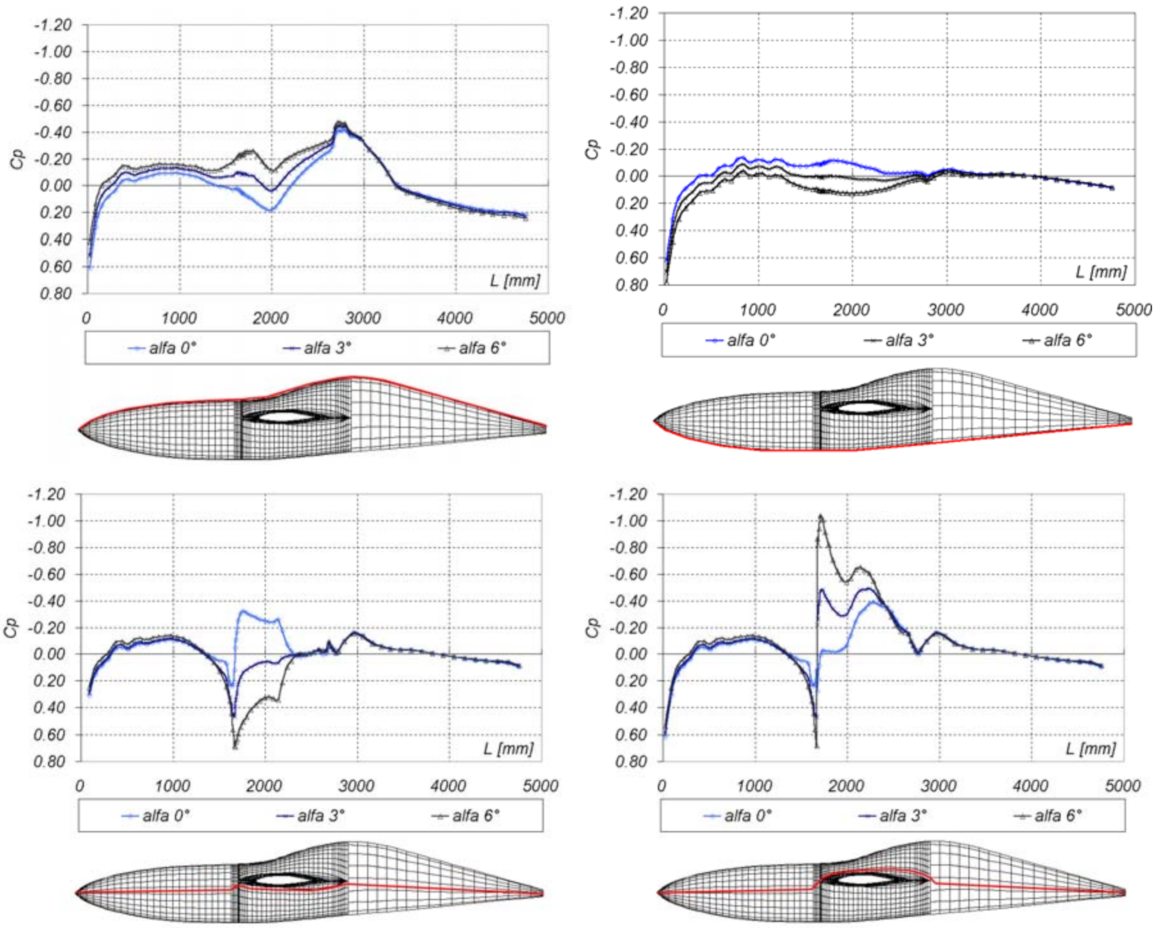

Figure 7. Pressure curves throughout the fuselage

AERODYNAMIC CHARACTERISTICS OF THE AIRCRAFT

The integration of pressure distribution on the aircraft provides values that can be quite useful in aerodynamic design of an aircraft. Taking into consideration that the solved problem is potential with lift modeling, the integral of pressure distribution on the aircraft will provide values that are coherent only for lift[1]. However, one must note that the aerodynamic moment in aircrafts is, mostly, due to pressure on the aircraft surface, with little influence from tangent forces (Anderson, 1991). Therefore, the moment values obtained through integration of pressure distribution can also be used efficiently. For the case in question, the aircraft CEA-308, some notable values, calculated through the results of the program discussed, are presented in Table 2, where they are compared with the respective values calculated through a semi-empiric method (Oliveira, 1999).

Note that the comparison between values obtained by each method shows some differences which may be considered significant. During the elaboration of this paper, it was noticed that the calculation of these amounts is quite influenced by the form in which the integration of pressure distribution takes place. A more precise analysis, both in the best form for pressure distribution integration, as in the reality of each result, must be done in order to obtain integral results which are closer to reality.

Table 2 – Comparison between aerodynamic characteristics of the aircraft CEA-308

Characteristic	Value calculated by semi-empiric methods		Value calculated by the panel method program,	
Lift coefficient	0°	0.1564	0°	0.1191
	3°	0.3952	3°	0.4067
	6°	0.6344	6°	0.6942
Null lift angle	-1.9		-1.2	
Moment coefficient in relation to ¼ mac	0°	-0.0479	0°	-0.0451
	3°	-0.0305	3°	-0.0470
	6°	-0.0131	6°	-0.0326
Pressure center position	0°	0.5562	0°	0.6288
	3°	0.3271	3°	0.3657
	6°	0.2706	6°	0.2970
Aerodynamic center position	0.177109		0.208286	

[1] If a special model weren't applied for bodies with lift, the pressure distribution integral would supply null results.

AIR EXIT AND ENTRANCE POSITION FOR THE RADIATORS

Although it doesn't model air entrance and exits for the radiators, the results obtained show that the chosen solution in the design appears to be quite satisfactory. Figure 8 shows that the air entrance area is the area of highest pressure in the entire fuselage, while the air exit are isn't the one with lowest pressure, but it has a pressure coefficient that is slightly negative. Therefore, the pressure difference between the exit and the entrance of air is positive, which guarantees air exchange in the refrigeration system.

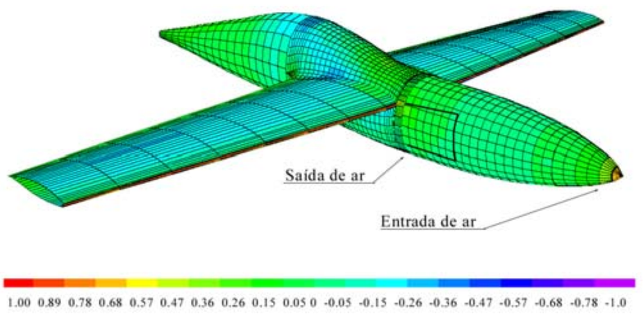

Figure 8. Pressures on the air entrance and exit áreas in the refrigeration system

Wing-fuselage intersection

Concurring with available literature (Hoerner, 1963), the wing-fuselage intersection must have a faring that begins after the point of maximum thickness in the wing roots and extends to the trailing edge with concordance rays becoming progressively bigger. After the trailing edge this faring must reduce it's radius of concordance until it extinguishes. In the CEA-308 design a faring of this type is used.

The main function of this faring reduce the intensity of the adverse pressure gradient that is formed in this area (due to reduction in wing and fuselage thickness, simultaneously), minimizing the flow instability in this area. .

As mentioned previously, two models were built, one with and one without this faringder to show it at work. The results obtained with the program in question for the models with and without faringhown in Figure 10.

The graphics in Figure 10 show that the pressure gradient in the faring (see arrow) is in fact smaller in the case with faringcan also observe that the flow in the area of the faring that is directly behind the wing trailing edge once again suffers a much more intense pressure gradient, which may provoke an unwanted instability in flow. An analysis of other shapes for the farings in this area may demonstrate more efficient solutions.

WING-CANOPY POSITION

The study in this paper includes the interference between the wing and the fuselage, especially regarding relative position between the wing and the canopy. For the design of this aircraft a concept presented by Arnold (1997) was used. According to him, the canopy must be positioned in order for the regions of the wing and the canopy itself, which have adverse velocity gradients are not superposed. This is done in order for there not to be velocities or adverse gradients of pressure higher than those that occur when there is only a wing or a fuselage flying alone. Therefore, with lower velocities, one must have a smaller drag of the aircraft, or slighter adverse gradients must have less instable flow.

For the case in study, a secondary configuration was modeled, where the canopy is 200mm ahead of its original position. Lateral views of the fuselage with pressure distribution is presented in Figure 11 for both cases.

Note that, in fact, when the canopy is pushed forward, for both attack angles considered (0° and 6°), the fuselage presents a low pressure area (blue area) that is much larger than the case in which the canopy is in normal position. Since low pressure areas correspond to high velocity, it is expected that the configuration with a forwarded canopy would generate more drag than the normal configuration.

POSITION OF THE PROBABLE AREA OF FLOW INSTABILITY ON THE FUSELAGE

It is known that the probable position of the point of flow instability on a fuselage is near the beginning of the adverse pressure gradient on this fuselage (usually after the point of maximum fuselage thickness) (Schlichting, 1955). Therefore, when one analyses the distributions throughout the length of the fuselage, one can obtain an estimative of the instability point of flow on the fuselage.

Taking into consideration that the point where adverse pressure gradient begins is after the maximum fuselage thickness and the area of flow transition, the areas of transition for each attack angle must be in accordance to Figure 12.

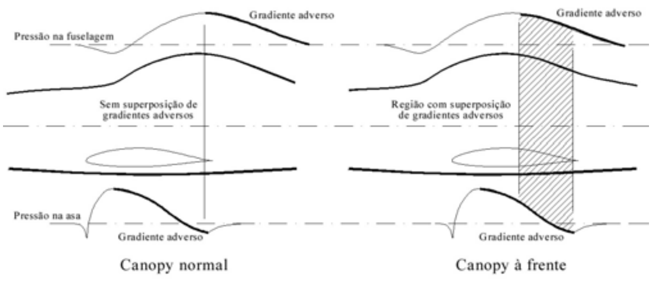

Figure 9. Arnold Proposal (1997) for reduction of drag of interference in wing and fuselage through choice of canopy position in relation to the wing.

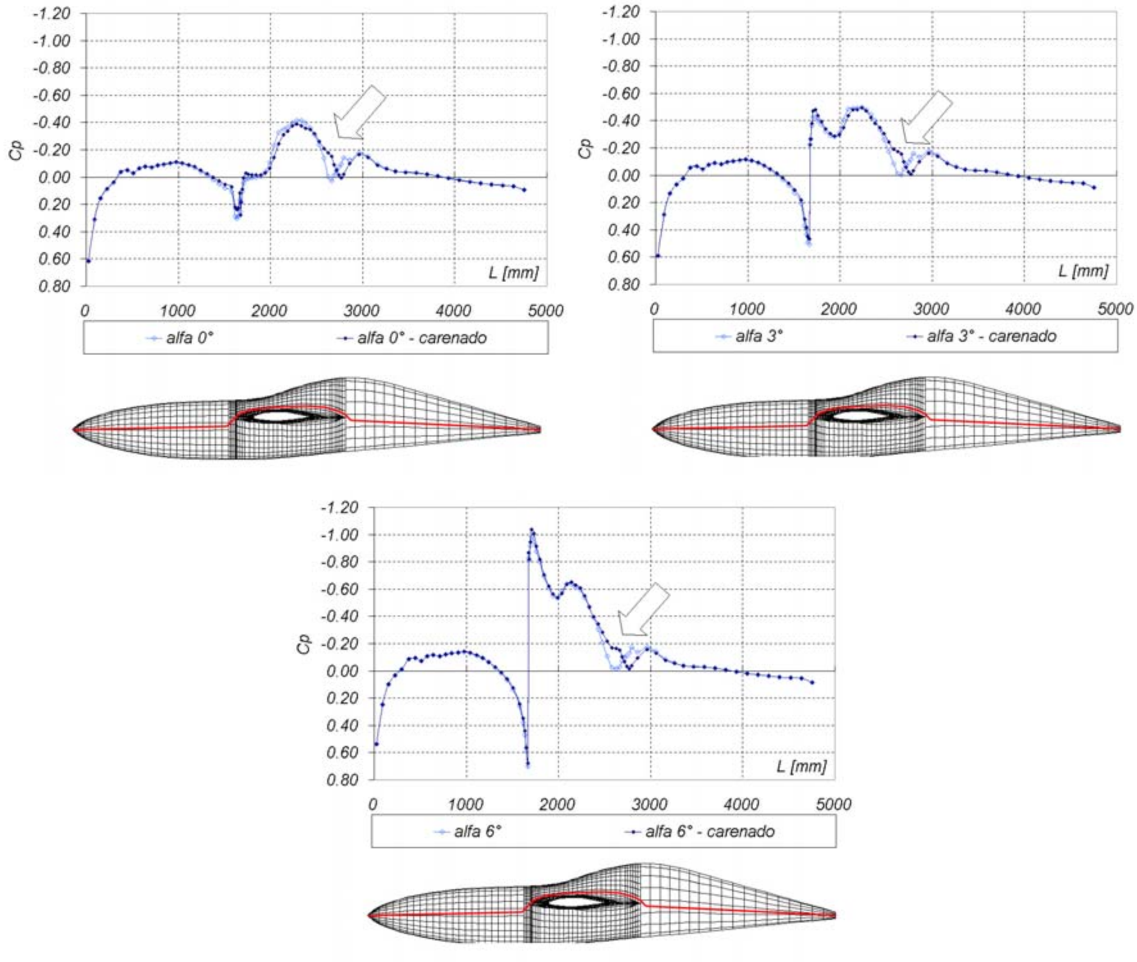

Figure 10. Comparison of pressure curves throughout the fuselage for cases with and without intersection faring between wing and fuselage

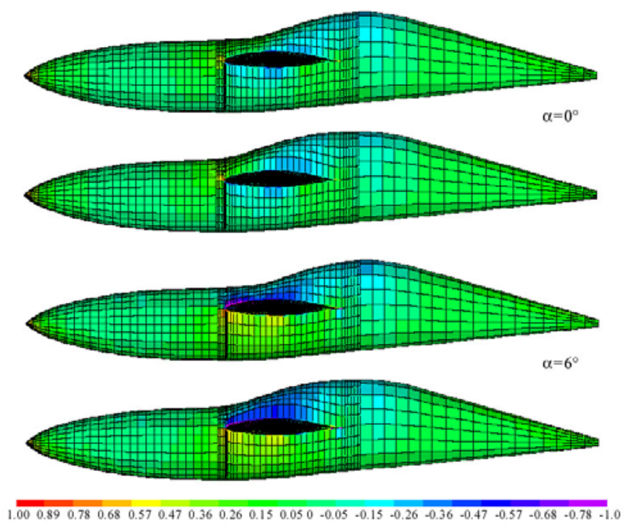

Figure 11. Lateral view of the aircraft CEA-308 with two canopy options

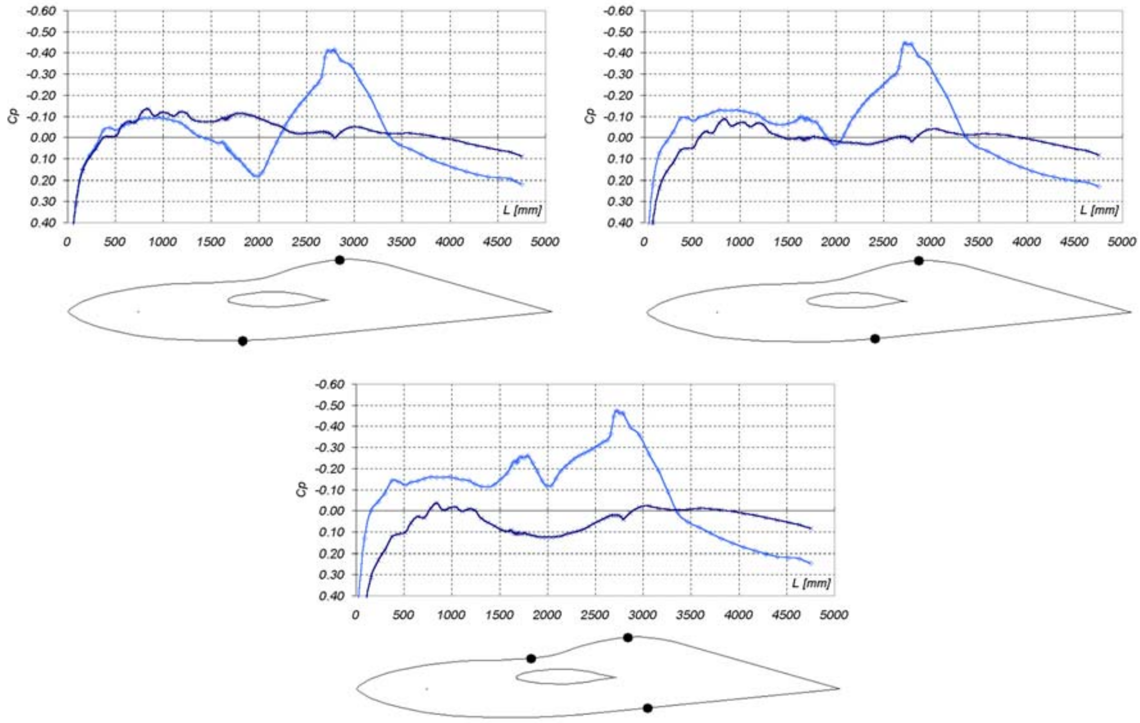

Figure 12. Probable points of instability α=0°, α =3° and α =6°

CONCLUSIONS

This paper presents an example of the application of a computer program of three-dimensional Panel Method for the aerodynamic design of light aircraft. This program uses the formulation presented by Hess (1966, 1972) which has supplied coherent and satisfactory results.

Comparison of the main aerodynamic characteristics calculated by the developed program and by a semi-empirical procedure presents some differences, which are attributed, basically to the difficulties in integration of pressure distribution over the aircraft surface.

The pressure curves on the fuselage show how the use of an wing-fuselage intersection farings is effective in the mitigation of the pressure gradient, which is usually adverse in this area.

Analysis of the wing-fuselage interference in regards to the canopy position confirms Arnold 's proposition (1997) for the correct relative position between wing and canopy.

In general, the analysis of the *CEA-308* aircraft showed that the use of the three-dimensional Panel Method as a designing tool is quite satisfactory. The details obtained in the results show that it is possible to analyze small alterations in the aircraft and the values obtained were, in terms of engineering, quite reasonable. Further improvements in regards to aircraft aerodynamic characteristics can still be reached with improvement of integration procedures for pressure distribution.

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