DESIGN AND OPTIMIZATION OF GLIDER COMPONENTS

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Summary

Modern techniques in aerodynamic surface construction allow long runs of Natural Laminar Flow (NLF) along airfoils and fuselages provided that their shape has been properly designed. A numerical optimization iterative procedure for drag reduction by shape modification of mono- and multi-component airfoils and three-dimensional fuselages, has been developed. To this aim we propose a geometric parameterization of a general 3D body. Reduction of drag through an extension of laminar flow runs, for airfoils and 3D glider fuselages, is shown. A modular numerical code developed to perform shape optimization for drag reduction has proven to be efficient and reliable.

Introduction

Present airplane construction techniques result in the production of smooth and accurate aerodynamic surfaces, allowing for long runs of natural laminar boundary layer flow (NLF), with a resultant drag reduction.

Attention has been focused on airplane lifting surfaces (Reference 1). Many airfoils have been designed in the past with large extension of laminar flow. In gliding it is also important to have high-lift for climbing with as low drag as possible. Variable geometry sailplanes apply this idea incorporating multi-component airfoils.

Furthermore, fuselage shaping is important for sailplane and hydrodynamic bodies. Fuselage skin-friction drag can reach about 70% of the total profile drag when wing and tail surfaces work in laminar flow (Reference 2). Table 1 shows drag coefficients for an axialsymmetric body with fineness ratio of *fr* = 5 are listed for various transition locations.

TABLE 1

Drag coefficients for an axialsymmetric body.

xtr (transition location)	C_D*
0.50	0.035
0.40	0.042
0.20	0.052
0.10	0.055

*reference area for C_D calculation is maximum frontal area

The present study investigates the possibility of obtaining:

1) low-drag airfoils
2) high-lift and low-drag multi-component airfoils
3) fuselage shapes with a large extension of laminar flow, not only for axialsymmetric bodies, but also for general three-dimensional configurations.

The main goal of this work is to refine and validate computational methods for the design of airfoils and general 3D fuselages.

A computational iterative optimization procedure to design airfoils, axialsymmetric bodies and general 3D glider fuselages has been developed. The design code is based on the numerical optimization technique and is made up of several replaceable modules, each of which addresses and solves a part of the complex problem.

Previous Studies

Most two-dimensional airfoils designed for gliders have been designed using the very well known Eppler's code (Reference 3) or codes based on one-point inverse design.

However, Eppler's code generated airfoils suffer from the fact that they are generated through conformal mapping and thus they are always characterized by a sort of a 'common base shape'. On the other hand, inverse design codes, like Marsden's (Reference 4), need the inviscid velocity distribution to be specified. Selig (Reference 5) has proposed a code for viscous airfoil design based on Newton iteration where the designer needs to specify just the desired aerodynamic global coefficients; it is, however, based on Eppler's conformal mapping.

Multi-component airfoils have been mostly designed by trial and error procedures, moving the flap around and finding the best ratio L/D (Reference 6). In this paper we propose a method to design airfoils starting from a given geometry through numerical optimization. Constraints, both geometrical and aerodynamic, such as minimum thickness or maximum moment coefficient can be imposed. Objective coefficients, such as Cd_{min} or Cl can be imposed on more than one point of the desired polar.

A comprehensive review of previous research related to axialsymmetric bodies has been made by Dodbele et al. (Reference 7). However, little experimental data are available for flows characterized by high Reynolds number (based on body length), i.e., for Reynolds number range of 30 to 70 X 10^6. It seems that the transition from laminar to turbulent flow occurs beyond the point of maximum thickness. This would indicate that the pressure gradient on the forebody of the configuration is the predominant factor in designing the body shape.

In this work we propose a method (Reference 8) to design the optimal shape of a general three-dimensional body, modifying directly the original geometry and investigating the effect of the local curvature variation on pressure distribution, transition and total drag.

Design optimization procedure

The numerical optimization method includes three elements: a constrained minimization program, an aerodynamic code (a solver to evaluate at each iteration the objective function that we want to minimize) and a parametric modification technique applied to the geometry. The computational design procedure proposed in this paper is described in the flow-chart presented in Figure 1.

As an optimizer, the constrained-minimization method proposed by Vanderplaats (Reference 9) has been used in the present investigation.

The objective function to be minimized is taken to be a function of a certain number of parameters

$$F_{OBJ} = F(x_1, x_2, \ldots\ldots\ldots, x_N)$$

These N parameters must satisfy Nc constraints conditions:

$$G_j(x_1, x_2, \ldots\ldots\ldots\ldots, x_N) \le 0 \qquad j = 1, \ldots\ldots\ldots Nc$$

and must be included in prescribed limits:

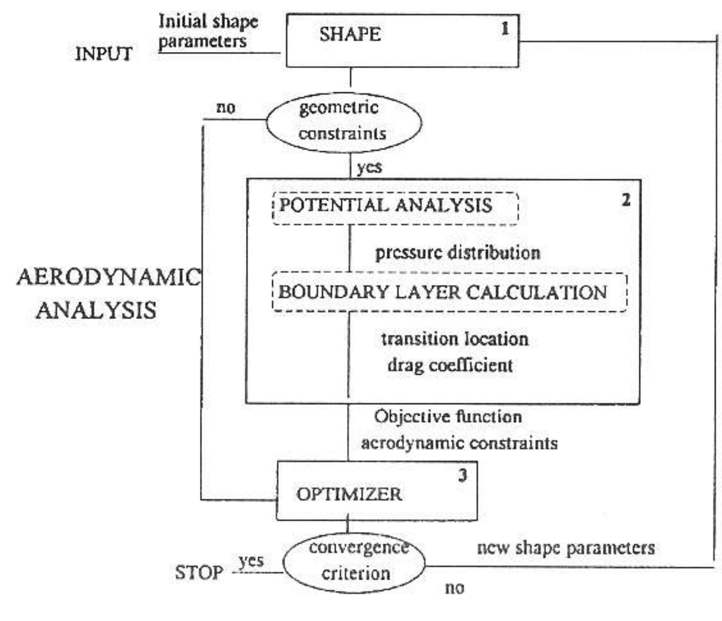

FIGURE 1. Flowchart of design iterative procedure.

$$x_i^L < x_i < x_i^U \qquad i = 1, \ldots\ldots\ldots N$$

A good choice of the geometric parameters is imperative in order to obtain good optimization results. It is important to choose parameters to which the objective function is more sensitive and to choose the correct number of parameters for geometric representation because there must be sufficient numbers to allow significant shape modifications. However, an excess number could render obtaining the desired results infeasible.

It is important to note that it is of fundamental importance to introduce and control both geometric and aerodynamic constraints for the problem under consideration, to avoid unsatisfactory solutions. The selection of pertinent parameters requires good knowledge of the physical problem by the designer.

Aerodynamic Analysis

The aerodynamic analysis is performed using an inviscid potential flow solver coupled with a viscous solver which predicts the boundary layer development along the body.

Two-dimensional viscous calculation

To predict mono and multi-component airfoils viscous characteristics, a numerical code (Reference 10) able to predict aerodynamic coefficients up to stall has been employed. The code is based on panel technique for external flow prediction coupled to direct or inverse boundary layer formulation in semi-inverse fashion (Reference 11). Boundary layer calculations are based on an integral formulation of the equations and transition is predicted using the Drela version of the e^n method. When bubbles are present, they are solved through direct computation using the semi-empirical formulation proposed by Dini (Reference 12).

Three-dimensional inviscid calculation

Because one of the main goals of the present work is to design, through numerical optimization, general three-dimensional bodies, we developed a 3D numerical code based on surface singularities distribution (Reference 13), to predict pressure distribution around the body.

Boundary layer calculation for 3D fuselages

An integral axialsymmetric boundary layer method has been coded and used to evaluate the effect of viscosity after external inviscid pressure distribution computation. The energy equation has been used in conjunction with the Von Karman momentum equation and Drela 2D closure correlations (Reference 14) have been coded for both laminar and turbulent parts of the boundary layer.

Drag is evaluated with the Young formula (Reference 15) which is based on integral quantities of the boundary layer, evaluated at the body's trailing edge. For general three-dimensional bodies we have calculated the axialsymmetric boundary layer along streamlines contained in the fuselage symmetrical plane (see Figure 2). The calculations were obtained from the 3D velocity field, using the radius distribution of an "equivalent axialsymmetric body." This was obtained from the original body by assigning a circular shape (of the same area) to every section.

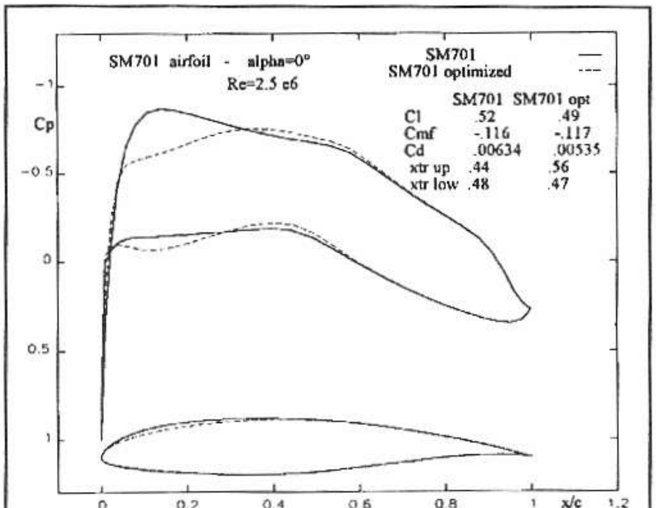

FIGURE 2. SM701 airfoil – pressure distribution at alpha = 0°, Re = 2.5e6 for original and optimized shape.

Transition

Because an accurate prediction of transition onset is a crucial point in designing low-drag airfoils and fuselages with large natural laminar flow area, we tested various methods and compared them to experimental results (Reference 16). As results of that investigation we found that e^n method, as proposed by Drela was able to predict transition even if, owing to the lack of experimental data, we could not test the validity of the method at high Reynolds numbers.

In general we can say that at those numbers there is a strong influence of external streamline curvature on transition onset. Furthermore, we are assuming that transition occurs always for spatial growth of T-S disturbances.

Also, the instabilities due to crossflow effect should be monitored for general three-dimensional configurations.

The prediction of transition onset is particularly difficult for bodies characterized by high fineness ratio because the pressure distribution is almost flat for a large part of the streamline, leading to an uncertainty in the predicted transition point.

Discussion of results

Mono-component airfoil design

After testing the code on the NLF(1) airfoil (Reference 1), we decided to optimize an airfoil specifically designed for World Class sailplanes: the SM701 airfoil designed by Dan Somers and Mark Maughmer (Reference 17). All design specifications are reported in cited reference and we summarize here the principal ones.

The objective was to obtain the Cl_{max} equal to 1.6 at Re = 0.5e6 with a negative moment coefficient not greater than 0.1 and the maximum thickness greater than 16%c.

The optimization was performed at same time for two angles of attack and Reynolds numbers: specifically at alpha = 0° and Re = 2.5e6, corresponding to cruise, and alpha = 3° and Re = 1.0e6, relative to climbing condition. Figure 2 shows the initial and optimized pressure distributions at alpha = 0° along with the two geometries. In Figure 3 the enlargment of the airfoils forward part is presented.

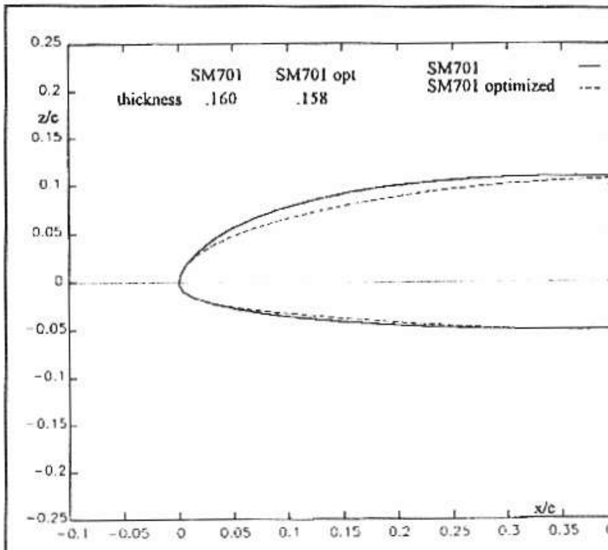

FIGURE 3. SM701 airfoil – leading edge shape for original and optimized shape.

The optimized airfoil presents a greater laminar flow region on the upper surface with a lower drag coefficient. Figure 4 shows the Cl versus Cd curve at Re = 2.5 millions; a Cd reduction at cruise Cl coefficient of about 15% can be seen. In this figure we also report the experimental values obtained at Stuttgart (Reference 17): in both investigations, there is a little underestimation of Cd in the laminar bucket range.

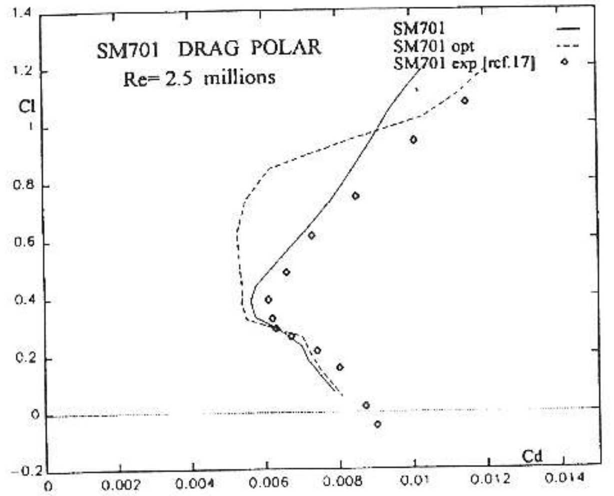

FIGURE 4. SM701 airfoil – Cl-Cd polar for original and optimized shape (Re = 2.5e6) – Comparison with experimental results (Reference 17).

The moment coefficient was also kept close to that of the original airfoil (see Figure 5); furthermore, we ran the code for both SM701 and optimized geometry up to stall conditions and the results seem to predict almost the same maximum lift coefficient. The resulting curves are not shown here, because the code is still under validation for stall and post stall conditions.

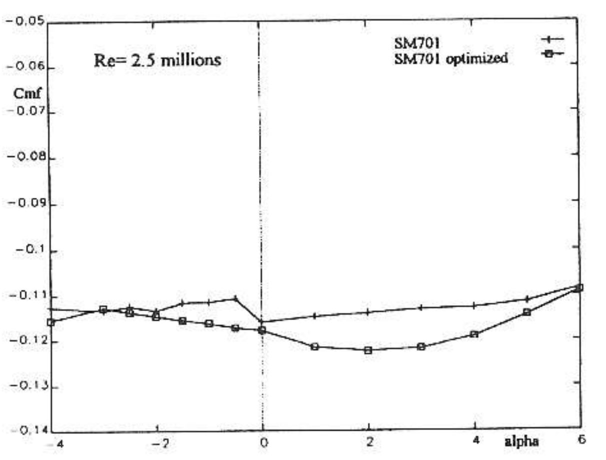

FIGURE 5. SM701 airfoil– moment coefficient for original and optimized shape.

Figure 6 shows the two drag polars at the two design Reynolds numbers: it can be clearly seen the extension of low Cd range obtained with respect to the SM701 original airfoil at both Reynolds numbers.

A typical run to optimize an airfoil for two design conditions takes about 1 hour on a 486 cpu based PC.

Multi-component airfoil design

The shape optimization of a multi-component airfoil

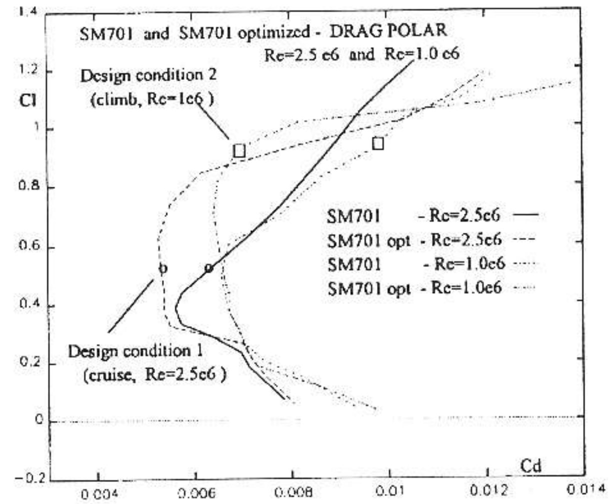

FIGURE 6. SM701 airfoil – Drag polar for original and optimized shape at Re = 2.5e6 and 1.0e6.

consists in finding the best position and shape of the flap. The results, in terms of lift and drag coefficients are greatly affected by the values of the gap and overlap.

We have performed the optimization of the UAG92 slotted flap airfoil designed by D.Marsden (Reference 6), introduced to increase the climb capability of his glider. The sink speed of a glider is proportional to the airfoil Cd/(Cl$^{1.5}$) ratio. Our goal was to optimize this parameter. It is clear that to decrease the above mentioned ratio, the direction is to increase the lift coefficient without big change of the drag coefficient value.

The optimization has been performed at an incidence at which the original geometry has the minimum value of the Vsp = Cd/(Cl$^{1.5}$) parameter choosing the Vsp itself as objective function, using the code to optimize flap position and shape.

In Figure 7 the pressure distribution relative the original

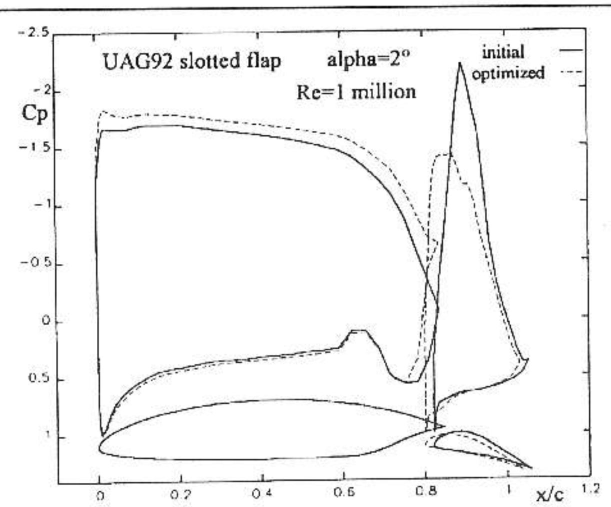

FIGURE 7. UAG92 slotted flap – Pressure distribution on initial and optimized configuration.

and the optimized airfoil is shown. The flap shape and position, along with the original ones, are represented in Figure 8. It is seen that with the gap reduction and flap geometry modification shown, the main component trailing edge pressure has become lower than that of the original airfoil. In this way we obtained higher total lift coefficient with basically the same Cd (see Table 2). Vsp was reduced by about 10% in all flight conditions.

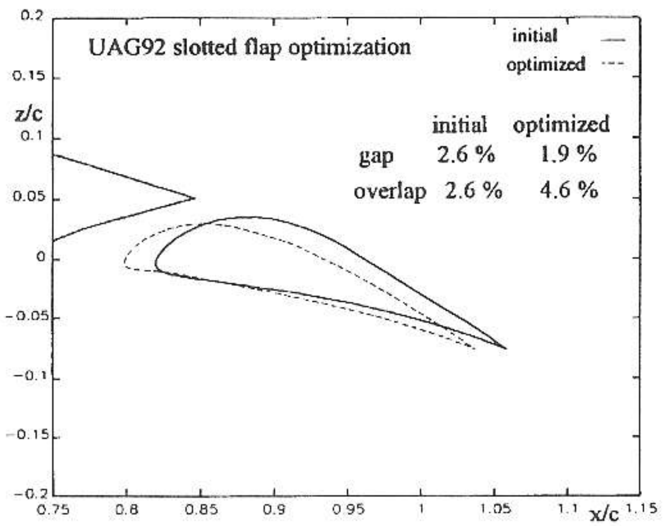

FIGURE 8. UAG92 slotted flap – flap shape and position for initial and optimized configuration.

Three-Dimensional Glider Fuselages Design

With the aerodynamic analysis technique for 3D bodies previously described, we have done many calculations for different fuselage shapes (Reference 16).

Obviously the goal was to obtain a lower value of the equivalent parasite area, indicated by *f* (product of Cd times the dimensionless frontal area) through a greater extension of natural laminar flow area.

To set up an iterative computational procedure, to design three-dimensional fuselage shapes, we had to solve the problem of the parameterization of such shapes. The problem is to represent a modification of an initial shape with a limited number of parameters.

By using a parameterization shape relative to the airfoil shape modification (Reference 18), that is based on the Legendre polynomials equation, it is possible to establish a parametric dependence by using 6 parameters for each of the following functions: $K_u(x)$, $K_l(x)$, $K_y(x)$.

Then we have $K(a_1, a_2, a_3, a_4, a_5, a_6, x)$, where a_1 - a_6 are the 6 parameters ; $K_u(x)$ is used for the upper surface modification, $K_l(x)$ for the lower surface modification, and $K_y(x)$ for the lateral variation of each fuselage section.

For each x constant cross section, the coordinates y and z of each point P = (x,y,z) are modified in the following way:

$y_{new} = y * K_y$

$z_{new} = x_z * Ku$ for P belonging to the upper surface

$z_{new} = z * K_l$ for P belonging to the lower surface

Through these 18 parameters, it is then possible to obtain a modification for each section as shown in Figure 9. This is particularly efficient for fuselages, because it permits preservation of the characteristics of the original shape.

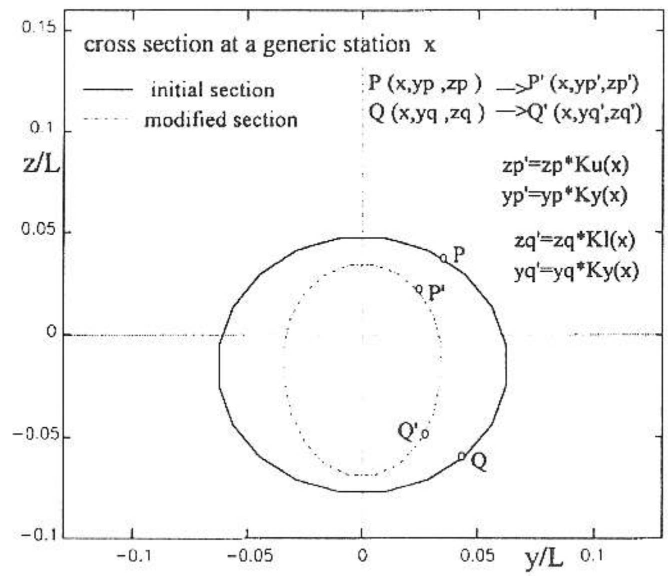

FIGURE 9. Cross section shape modification technique.

Initially no geometric constraints were imposed on the fuselage's shape (for example, maximum fineness ratio). The code found a shape with a greater laminar flow extension, but with a greater maximum frontal area. The result is that the drag of the optimized fuselage is greater than that of the initial shape.

The optimization was then performed with an imposed constraint on the fineness ratio *fr*, attempting to increase the natural laminar flow area. Figure 10 shows the shape modification in the forebody region, favorable dorsal mid-

TABLE 2
Aerodynamic results for UAG92 slotted flap optimization.

$\alpha = 2°$ Re=1 million		
	initial	optimized
Cl	1.84	1.94
Cd	0.0152	0.0149
$Vsp = Cd/Cl^{1.5}$	0.0061	0.0055

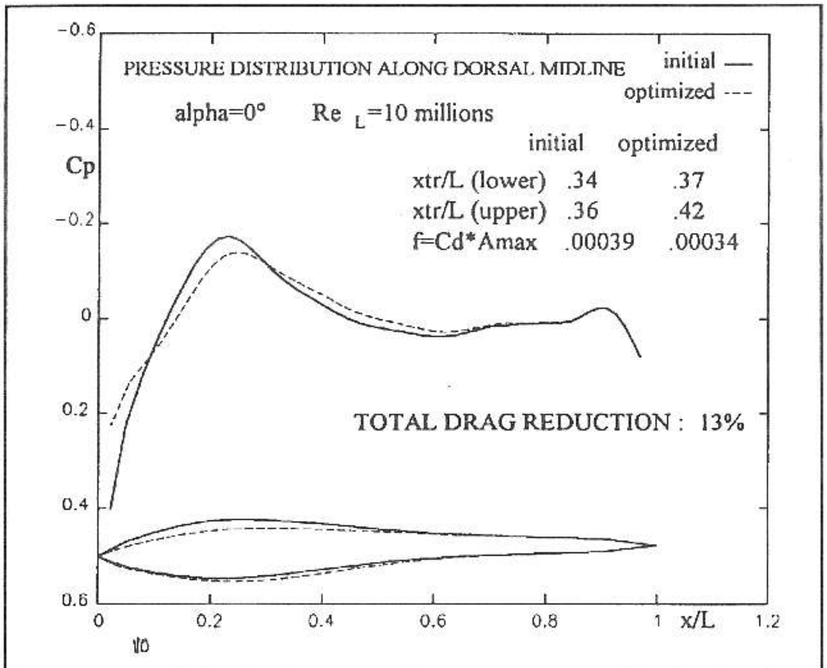

FIGURE 10. Glider fuselage optimization – pressure distribution along dorsal midline for original and optimized shape.

line pressure distribution and transition locations.

A drag reduction of about 13% was obtained.

In Table 3 are described geometric and aerodynamic characteristics of the two fuselages.

Initial and optimized longitudinal section are shown in Figure 11.

A typical run that requires about 400 iterations, using 500 surface panels took about 30 seconds per iteration of a CONVEX 34 cpu time.

Conclusions

A numerical optimization procedure, to design mono- and multi- component airfoils and three-dimensional glider fuselages, has been developed. The code is completely modular, in that it is easy to change a module and address a different problem (for example axialsymmetric bodies design). Sensitivity of the optimization procedure to the choice of the objective function and constraints has been highlighted. Furthermore, we have proposed a way to parametrize the geometry of a general three dimensional fuselage. Presented are results of optimization processes, showing drag reduction up to 15% and an increase in the transition location. We have also shown the multi-point design capability of such an approach. Finally we can state that the proposed methodology is suitable for designing glider components. Extension of the procedure to the optimization of wing-body junctions is under development.

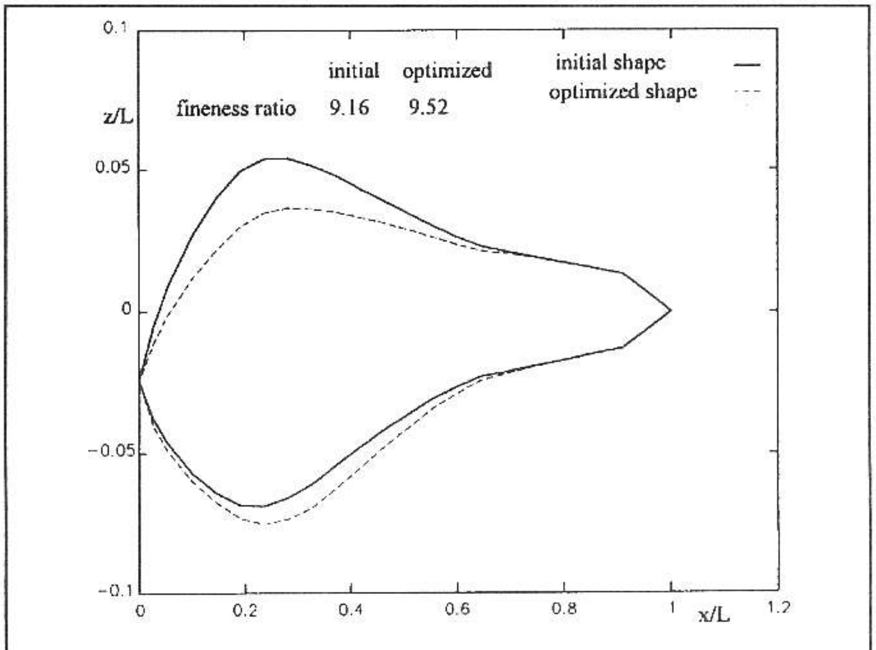

FIGURE 11. Glider fuselage optimization – longitudinal section for initial and optimized shape.

TABLE 3

Geometrics and aerodynamics characteristics for initial and optimized glider fuselage.

$\alpha=0°$ $Re_L=10e6$

	fr	xtr up	xtr low	C_D up	C_D low	C_D tot	$f \cdot 10^3$
initial	9.16	0.36	0.34	0.041	0.043	0.042	0.39
opt.	9.52	0.42	0.37	0.037	0.042	0.039	0.34

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