CONCEPTUAL DESIGN AND OPTIMIZATION OF A HIGH ASPECT RATIO UAV WING MADE OF COMPOSITE MATERIAL

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Abstract

The applicability of an optimization routine included in the Ansys® software applied to fiber reinforced structures was analyzed via a practical example. A laminated wing was modeled and optimization parameters were defined such as weight, fiber orientation angle and failure criteria. Two optimization methods were used. Results are analyzed and the method restrictions are discussed and other methods are proposed.

Notation

b	wing span
Cl	lift coefficient
E	Young´s modulus
F^{cu}	ultimate load in compression
FLTHK	flange thickness
F^{su}	ultimate shear load
F^{tu}	ultimate load in tension
G	shear modulus
L	lift
MAXFC	max failure criteria
ν	Poisson ratio
ρ	Density
S	wing surface area
σ	stress
σ^f	failure stress
θy	rotation in y
θz	rotation in z
ux	displacement in x direction
uz	displacement in z direction
V	velocity
WEBTHK	web thickness
ξ	failure criteria value

1 Introduction

Unmanned air vehicles (UAV's) fly typically long endurance missions, at high altitudes. Therefore, it is very important that the aircraft has a high lift to drag ratio. That can be accomplished using a high aspect ratio wing which its inherent low induced drag. Also advanced materials like laminated composites can be employed. They have a high strength to weight ratio, allow an excellent finishing (less friction drag) and can have his properties changed easily, by changing his lamination orientation and sequence, in order to optimize it.

E-GLASS/EPOXY TWO-DIRECTIONAL	POLYURETHANE FOAM
$\rho = 2100\ kg/m^3$	$\rho = 35 kg/m^3$
$E_{11} = 15{,}9$ Gpa	$E = 10{,}3$ Mpa
$E_{22} = 15{,}9$ Gpa	$\nu = 0{,}25$
$E_{33} = 8{,}7$ Gpa	$G = 4{,}1$ Mpa
$\nu_{12} = 0{,}20$	$F_x^{tu} = 250$ Kpa
$\nu_{23} = 0{,}28$	$F_x^{cu} = 250$ Kpa
$\nu_{13} = 0{,}28$	$F_y^{tu} = 250$ Kpa
$G_{12} = 4{,}1$ Gpa	$F_y^{cu} = 250$ Kpa
$G_{23} = 2{,}9$ Gpa	
$G_{13} = 2{,}9$ Gpa	
$F_1^{tu} = 228$ Mpa	
$F_1^{cu} = 165$ Mpa	
$F_2^{tu} = 228$ Mpa	
$F_2^{cu} = 165$ Mpa	
$F_{12}^{su} = 50$ Mpa	

Table 1. Mechanical properties of the bi-directional laminated of E-Glass (60%) with Epoxy matrix and of the polyurethane foam.

Moreover, the minimization of the structural mass of the aircraft is a primary objective of the manufacturers. In consequence, a large number of very efficient private and commercial systems are being created, by using optimization routines together with finite elements modeling [1].

Here one of those commercially available

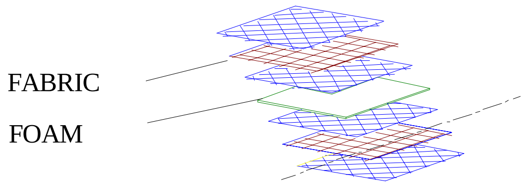

Figure 1. Typical sandwich lamination sequence (45 - 0 -45).

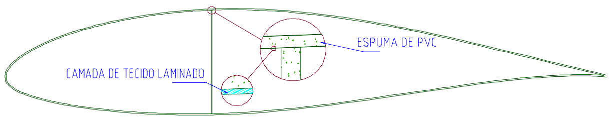

Figure 2. Central wing section cut view.

routines will be studied. It is included in Ansys®, and here is applied to the optimization of laminate composite structures.

The use of fiber reinforced composite leads to many optimization possibilities. Due to the orthotropic characteristic of the composite, it is possible to design the material, favoring the properties of the composite in the "flow" directions of the tensions. Another advantage is the easiness of obtaining complex forms, and the easy alteration of the laminated characteristics, such as thickness, orientation of the fiber and number of layers.

The wing here modeled is laminated, with a sandwich sequence of lamination as seen in figure 1. Bi-directional Glass fabric is used with a matrix of Epoxy resin and polyurethane foam, whose properties are in the Table 1. In figure 2 can be seen a cut of the wing.

The initial sequence of lamination obeys the pattern 45-0-45, symmetrical to the central foam layer. The I spar section is positioned where the section of the wing has the largest thickness. The flanges of the spar are unidirectional woven, aligned with the axis of the spar. Two ribs were placed. One in the root and other in the tip of the wing.

For the sizing of the wing it is necessary to determine which are the cases in that the wing suffers a maximal loading. Here for simplicity just one case will be considered, equivalent to an abrupt pitch of the airplane flying in the maximum speed of maneuver, VA. In this situation the wing reaches the maximum Cl.

2 Theoretical considerations

For the fail analysis in the laminated the Tsai-Wu failure criteria was used. It leads to the lightest structures [2]. This theory postulates that for an orthotropic material [3] and [4], fail occurs when x <1, being:

$$\xi = \frac{1}{\left(-\frac{B}{2A} + \sqrt{\left(\frac{B}{2A}\right)^2 + \frac{1}{A}}\right)} \quad (2.1)$$

$$A = -\frac{\sigma_x^{\,2}}{\sigma_{xt}^f \sigma_{xc}^f} - \frac{\sigma_y^{\,2}}{\sigma_{yt}^f \sigma_{yc}^f} - \frac{\sigma_z^{\,2}}{\sigma_{zt}^f \sigma_{zc}^f} + \frac{\sigma_{xy}^{\,2}}{(\sigma_{xy}^f)^2} + \frac{\sigma_{yz}^{\,2}}{(\sigma_{yz}^f)^2} + \frac{\sigma_{xz}^{\,2}}{(\sigma_{xz}^f)^2} + \frac{C_{xy}\sigma_x\sigma_y}{\sqrt{\sigma_{xt}^f\sigma_{xc}^f\sigma_{yt}^f\sigma_{yc}^f}} + \frac{C_{yz}\sigma_y\sigma_z}{\sqrt{\sigma_{yt}^f\sigma_{yc}^f\sigma_{zt}^f\sigma_{zc}^f}} + \frac{C_{xz}\sigma_x\sigma_z}{\sqrt{\sigma_{xt}^f\sigma_{xc}^f\sigma_{zt}^f\sigma_{zc}^f}} \quad (2.2)$$

$$B = \left(\frac{1}{\sigma_{xt}^f} + \frac{1}{\sigma_{xc}^f}\right)\sigma_x + \left(\frac{1}{\sigma_{yt}^f} + \frac{1}{\sigma_{yc}^f}\right)\sigma_y + \left(\frac{1}{\sigma_{zt}^f} + \frac{1}{\sigma_{zc}^f}\right)\sigma_z \quad (2.3)$$

where,
Cxy = coefficient of joining x-y for theory of Tsai-Wu (=2Fxy)
Cyz = coefficient of joining x-y for theory of Tsai-Wu (=2Fyz)
Cxz = coefficient of joining x-y for theory of Tsai-Wu (=2Fxz)

where Fij are tension tensors, σ_i the principal tensions and σ_i^f - the ultimate tensile loads.

For the determination of the loading, the distribution of Cp along the wing chord was found using a simple panel method, that resulted in the distribution shown in figure 3. Then it was approximated for a polynomial using Mathematica® for the upper and lower wing surface. Along the span the pressure distribution varies elliptically.

The distribution can then be approximated using the following equation:

$$Cp(Y) = \sqrt{1-\left(\frac{2y}{b}\right)^2} \qquad (2.4)$$

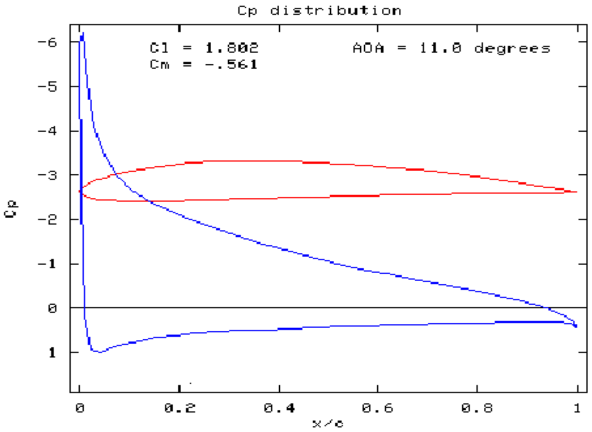

Figure 3. Chord Cp distribution ($\alpha = 11^o$).

3 Modeling description

Structural Optimization is a technique that tries to determine an optimum design [5]. For "optimum" design we understand one that satisfies all the requested specifications but with a minimum expenditure of certain factors as weight, superficial area, volume, tension, cost, etc. In other words, the optimum design is usually the one that is so effective as possible.

Before describing the optimization procedure, some basic definitions will be given: design variable, state variable, objective function, analysis file, etc.

Design Variables (DVs) are independent quantities that are varied with the objective of obtaining the best design. Superiors and inferior limits are specified to serve as "contour" in the design variables.

State Variables (SVs) are quantities that define the specifications of the design. They are also known as "dependent" variables, and are typically resulting quantities that are function of the design variables.

The *Objective Function* is the dependent variable that is being minimized. It should be a function of the design variables.

The analysis file is an input file of Ansys® that contains a sequence of complete analysis, (pre-processing, solution, post-processing). The model has to be defined parametrically, where the parameters are all the optimization variables. After having found the solution, the variables of interest are recovered for use in the optimization routine.

Starting from this file, a file optimization loop is created and used automatically by the software to accomplish the analysis loops.

First was used a optimization tool called *Random Design Generation*, where the program accomplishes a specific number of analysis loops using random values of design variables for each loop. This is a good tool to arrive quickly to an initial configuration of parameters that can be used with a more adequate optimization method. Then was used an optimization technique called *Subproblem Approximation Method*. This is an advanced zero-order method that uses curve fitting to all dependent variables.

As the problem possesses symmetry, just half of the wing was modeled. The mesh was generated automatically on the surface. An element of laminated shell was used, *shell99*. It can be triangular or quadratic, with 6 or 8 nodes. It allows the calculation of the interlaminar tensions and of the value of the failure criteria, besides the displacements and deformations.

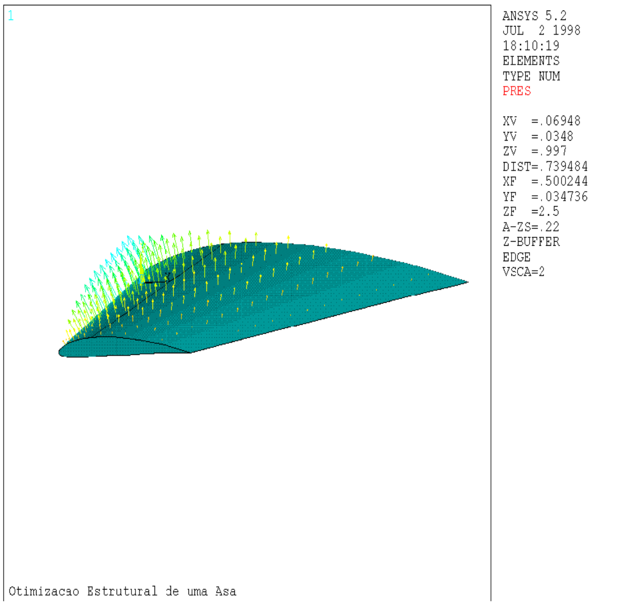

Figure 4. Pressure distribution over the wing model.

For the application of the loading, a constant pressure was applied on each element (see figure 4), whose value was previously calculated for the center of each element using the equations obtained in the previous section. The force resulting normal to the flight direction, lift, is obtained of the following well known equation, being considered that the flow is bidimensional:

$$L = Cl\rho.\frac{V^2}{2}.S \qquad (3.1)$$

For the situation of the wing flying with an angle of incidence of 11^o (angle of stall of the airfoil) at 125 Km/h (here defined arbitrarily, as being VA), the lift force is 4737,91N.

All the spar nodes in the root of the wing were constrained in ux, uz, θz and θy. Two nodes, in the position where the wing torsion pins would be, in the main rib, were constrained in ux and uy, according to figure 5. The initial lamination sequence was previously defined. The skin of the wing was divided in 5 sections of equal length, and the first in the root has 10 layers of fabric and one of foam, in the center. Each following section has minus two layers, until the last, that has only three (see figure 6).

The spar has 5 layers in the web, including the foam. The flanges were modeled as a unique layer of unidirectional fabric (figure 5).

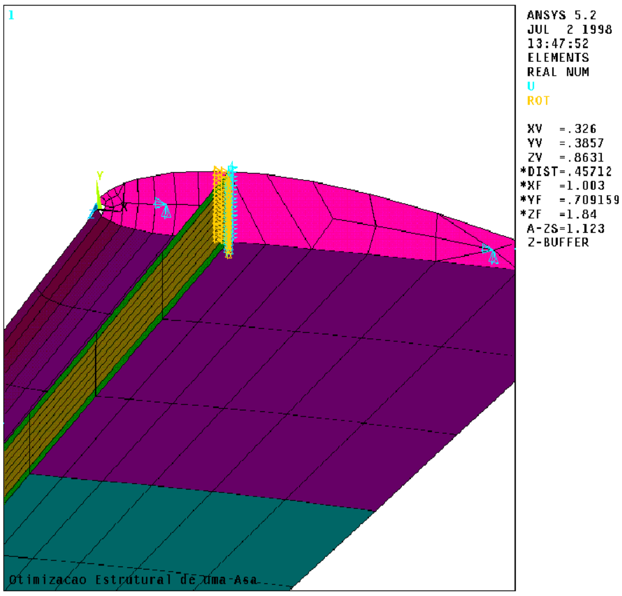

Figure 5. Node constrains and spar mesh.

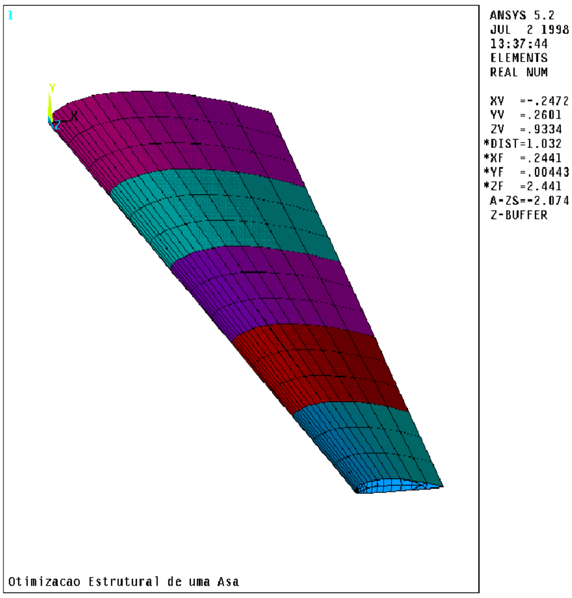

Figure 6. Skin proprieties changing along span.

Twelve design variables were defined, the thickness and the orientation of each layer of the skin and of the spar. The intervals of layer thickness were restricted to the maximum and minimum fabric thickness available commercially. As state variable was defined the value of the failure criteria, that should be below 0,8, giving a margin of safety of 25%. The objective function to be minimized was defined as being the volume, that is proportional to the weight.

Following are presented some data of the modeling:

Variable	Min	Max	Tolerance
MAXFC	0,100000000E-04	0,800000000E+00	0,500000000E-01
THK1	0,159000000E-03	0,558000000E-03	0,399000000E-05
THK2	0,159000000E-03	0,558000000E-03	0,400000000E-05
THK3	0,159000000E-03	0,558000000E-03	0,400000000E-05
THK4	0,159000000E-03	0,558000000E-03	0,400000000E-05
THK5	0,159000000E-03	0,558000000E-03	0,400000000E-05
WEBTHK	0,159000000E-03	0,558000000E-03	0,9190000000E-05
FLTHK	0,100000000E-02	0,800000000E-01	0,590000000E-03
TETHA1	0,900000000E-01	0,900000000E+02	0,899100000E+00
TETHA2	0,900000000E-01	0,900000000E+02	0,899100000E+00
TETHA3	0,900000000E-01	0,900000000E+02	0,899100000E+00
TETHA4	0,900000000E-01	0,900000000E+02	0,899100000E+00
TETHA5	0,900000000E-01	0,900000000E+02	0,899100000E+00
VOLUME	-	-	0,100000000E-01

Table 2. Parameters variables of the optimization.

4 Results analysis

After 15 loops, a configuration that doesn't fail was reached, although safety's margin has been below 25%. In Figure 7 can be observed that the area close to the wing root has the smallest margin of safety. It is also possible to notice that close to the tip of the wing the tensions are low, and it would be possible to obtain a more effective optimization modifying the lamination sequence (extension of each layer, figure 6). It is still noticed the leading edge area is a stress concentration area.

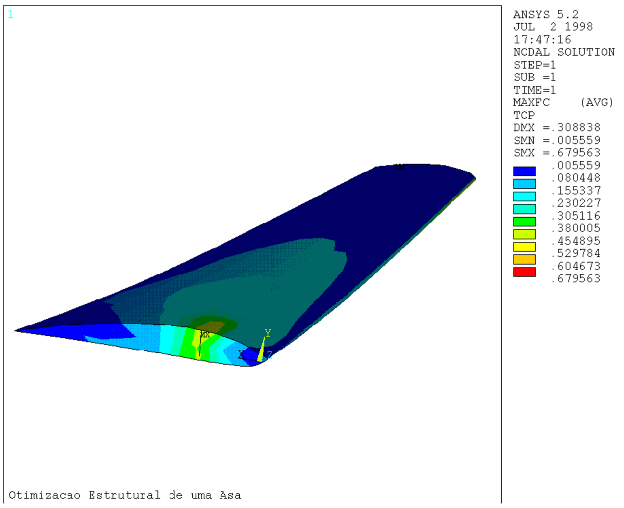

Figure 7. Tsai-Wu failure criteria distribution along the wing skin.

In figure 7 and 8 can be seen the distribution of the failure criteria for the skin and spar.

Variable	Type	Value
MAXFC	(SV)	0,92074
THK1	(DV)	0,39296E-03
THK2	(DV)	0,42252E-03
THK3	(DV)	0,25240E-03
THK4	(DV)	0,35088E-03
THK5	(DV)	0,30880E-03
WEBTHK	(DV)	0,19509E-03
ESPML	(DV)	0,75797E-01
TETHA1	(DV)	37,197
TETHA2	(DV)	40,546
TETHA3	(DV)	72,871
TETHA4	(DV)	60,525
TETHA5	(DV)	83,852
VOLUME	(OBJ)	0,75349E-01

Table 3. Values of the variables obtained after the optimization.

# OF ELEMENTS	639
DEGREES OF FREEDOM	11268
ELEMENTS MATRIX	12,25 MB
TRIANGULAR MATRIX	42,12 MB
PROCES TIME p/ LOOP	4 min
# de LOOPS	15
FINAL MASS OF THE WING	55,612 Kg

Table 4. Process data.

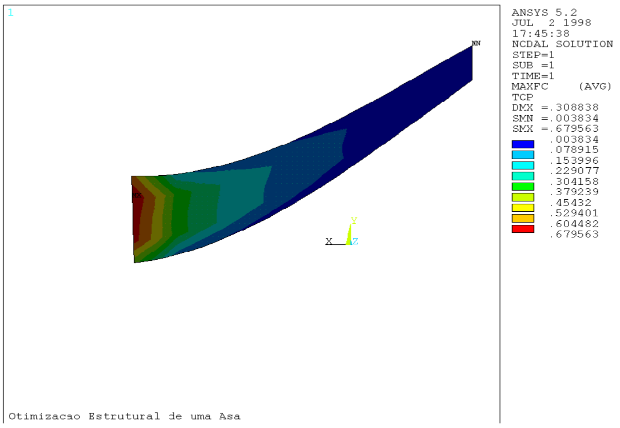

Figure 8. Tsai-Wu failure criteria distribution along the spar.

5 Laminate composite optimization using genetic algorithms (GA)

GA are powerful tools for the optimization of laminate composite [6]. The GA handles easily with the discrete nature of the stacking problem. Optimization of laminate composite structures involve not just orientation to specific thickness, but also the number of layers, the kind of fabric of each layer and the total thickness.

Gürdal et all [6] found, optimizing composite panels under combined in-plane loads, structures that were 5.7% lighter. They found also that GA leads to many different feasible solutions due to its random characteristics.

6 Conclusion

With the present work it was verified that Ansys® allows the use of optimization routines applied to composite laminated structures. The following difficulties were noticed:

-difficulty in defining as design variable the number of layers of the laminate. The difficulty is due the way that the properties of the element *shell99* are defined, and the impossibility of just attributing integer values to the design variable.

-in practice, change the thickness would be only possible for few discreet values, that would be the commercially available fabric thickness. But that could not be made also. Defined the variable, the program attributes any real value that is inside the defined interval. A possible solution would be to eliminate that design variable, and to use different models with different configurations of that variable. Or still, just use a sequence of pre-defined fabrics, and to optimize the extension of each layer in the laminated along the surface of the wing. The difficulty in this case would be the need of a variable mesh.

-it is not possible to define the mass directly as objective function, and in the case of variable density elements, as in the laminated sandwich, it would be more appropriate.

The two first difficulties are easily solved using a GA based optimization routine. Currently a GA based on ANSYS Parametric Design Language (APDL) is been developed to work as a user optimization subroutine at the Ansys session.

It would be also interesting to do, and the program allows it, a sensitive analysis of each design variable in the objective function, allowing a better evaluation of the project parameters.

References

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