MULTI-OBJECTIVE OPTIMIZATION OF THE COMPOSITE WING BOX OF SOLAR POWERED HALE UAV

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Abstract

Nowadays there is a growing request for Very Long Endurance Solar-Powered Autonomous Aircraft (VESPAA) flying at stratospheric altitudes of 17-25 km because they can act as artificial satellites - with the advantage of being much cheaper, closer to the ground, and being able to perform missions that offer greater flexibility. A long experience has been acquired by the Polytechnic of Turin, Department of Mechanical and Aerospace Eng. (POLITO/DIMEAS, Scientific Responsible Prof. G. Romeo) in Design of Solar powered UAV as High Altitude Very long Endurance Platform positioned in the stratosphere and with an endurance of more than six months. Two different configurations are under investigation in order to understand the best solution that completely fulfill the a priori imposed constraints. The main goal is to obtain a structure as light as possible that ensures the capability to resist at loads evaluated from the flight envelope without catastrophic failures.

A Genetic Algorithm has been developed to optimize composite laminates subjected to mechanical, thermal and hygroscopic loads. A set of penalty function has been defined in order to guarantee deformations less than the allowable limit and to avoid local or global buckling. A Progressive Ply Failure Analysis code has been developed in order to evaluate the Last Ply Failure Loads of composite laminates.

Since, in agreement with international regulations, the structure must be able to resist at ultimate loads without failures for at least three seconds, Genetic Algorithm and PPFA have been coupled in order to optimize the lay-up of each panel that ensure the minimum wing box weight, a proper safety margin from FPF to LPF and respects of the penalty function. In particular, no failure are allowed at limit load while, at ultimate load, each panel must stay under LPF level.

1 Introduction

HALE UAV related technology has reached a high level of maturity for a wide range of tasks. These kinds of UAVs are still the subject of different research programmes. Research is currently being carried out, under the coordination of the first author, with the aim of designing a Very-Long Endurance Solar Powered Autonomous Stratospheric UAV (VESPAS-UAV) [1–4]; two different configurations (Fig. 1) have been developed during the research programme. The characteristics of HeliPlat (Helios Platform) and Shampo (Solar Hale Aircraft Multi Payload & Operation), which are shown in Table 1, highlight the main differences between the two configurations.

The possibility of long endurance (6-12 months) for a stratospheric platform can be realised with the application of an integrated Hydrogen-based energy system. It is a closed-loop system: during daytime, the power generated by thin high efficiency solar cells that cover the aircraft's wing and horizontal tail supply power to electric motors for flying and to an electrolyser which splits

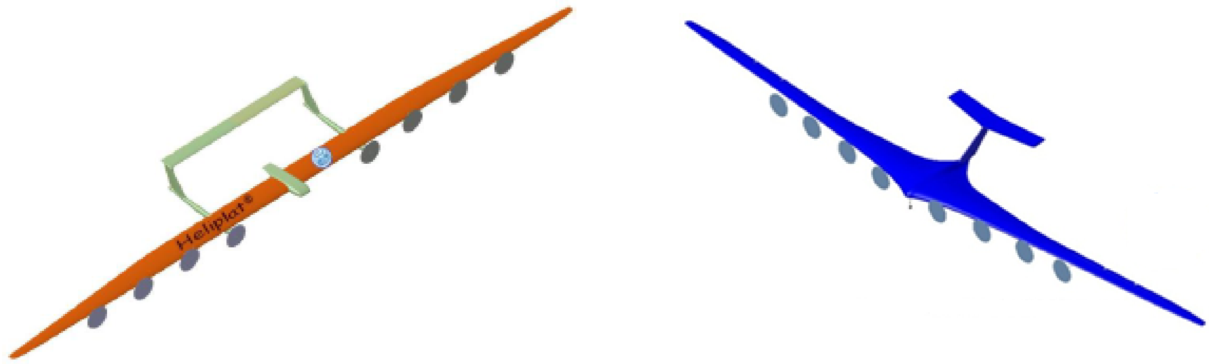

Fig. 1 Solar HALE UAVs under studying respectively, from left, HeliPlat and SHAMPO

water into its two components, hydrogen and oxygen. The gases are stored into pressurized tanks and then, during night-time, used as inlet gases for fuel cells stack in order to produce electric DC power and water to be supplied to the electrolyser. Solar cells and fuels cells are, till today, the weakest element of the project. Good efficiency can be obtained through the proper design of the other elements of the propulsion system.

The main goal of this work is to design the composite wing box of the SHAMPO's configuartion that ensures a high strength to weight ratio, compliance with regulatory requirements and respects the constraints imposed a priori, including maximum wing-tip deflection, mass minimization, buckling and impact threshold limit. As it is worded, the multi-constraints design problem coupled with mass minimization becomes a Multi-objective Optimization problem. To achieve this goals, several different tools, both for design and optimization, have been developed and validated before being used for the final design.

2 Design approach

Simplified beam or plate models of aircraft wing structures are often used for analysis during early preliminary design. Such models are usually required for structural optmization when static or dynamic aeroelastic constraints are considered. These semplified models could faithfully represent the real behaviour of the wing box if some hypotheses, such as the slenderness of the latter, are being satisfied. In these case, differences between different models become low while, the computational cost decreases in such a way that justifies the use of the semplified approach.

Due to the high aspect ratio, in this work, the wing box has been considered be like box beam. The inter-ribs space identifies the computational sections in which the wing box has been divided. Each section is made by a C-spar placed at 25% of the chord and a reinforced nose. The membranal approach suggested in [5], has been choosed to evaluate the wing box's stiffness.

The failure criteria choosed for the desing is the Tsai-Hill criteria, which is a interactive or energetic criteria. These criteria predict the failure load by using a single quadratic or higher order polynomial equation involving all stress (or strain) components. The mode of failure is determined indirectly by comparing the stress/strength ratios. However, some failure mode such as microcracks must be taken into account since the immediate effect of microcracks is to cause a degradation in the thermomechanical properties of the laminate including changes in all effective moduli, Poisson ratios, and thermal expansion coefficients. A secondary effect of microcracks is that they nucleate other forms of damage. For example, microcracks can induce delaminations, cause fiber breaks, or provide pathways for entry of corrosive liquidscan be observed during tensile loading, during fatigue loading, during changes

Description	Symbol	Value HeliPlat	Value Shampo
Flight Alttude	z	17000 m	17000 m
Power available for payload	P_{pay}	1000 W	1300 W
Power available for avionic	P_{av}	300 W	325 W
Cruise flight power supplied to the electric motors	P_{flt}	7300 W	6700 W
Sun Power ($38°N$)	P_{sun}	15000 W	11560 W
Avionic mass	W_{av}	25 Kg	32 Kg
Max. payload mass	W_{pay}	100 Kg	100 Kg
Strucutural mass	W_{str}	345 Kg	430 Kg
Solar cells mass	W_{SC}	78 Kg	127 Kg
Take-off weight	W_{TO}	840 Kg	924 Kg
Efficiency energy storage system	η_{FC}	0.6	0.6
Density energy storage system	D_{FC}	600 Wh/Kg	600Wh/Kg
Efficiency solar cells	η_{SC}	0.21	0.21
Density solar cells	D_{SC}	0.4 Kg/m^3	0.6 Kg/m^3
Efficiency electric motor	η_m	0.95	0.95
Efficiency propeller	η_{prop}	0.85	0.85
Number of motors	N	8	8
Cruise airspeed	TAS	20 m/s	25 m/s
Wing span	b	73 m	73 m
Wing surface	S_W	176.5 m^2	192 m^2
Aspect ratio	AR_W	30	28
Root chord	c_{root}	2.97 m	9 m
Tip chord	c_{tip}	0.95 m	0.9 m
Sweep angle	Λ_W	0 deg	5 deg

Table 1 Heliplat and Shampo main characteristics

in temperature, and during thermocycling. Microcracks can form in any plies, but they form predominantly in plies off-axis to loading directions. The processes by which microcracks form, the effects they have on laminate properties, and their role in nucleating new forms of damage are all important problems in the analysis of failure of composite laminates. A common practice to avoid matrix cracking is to have no more than four plies with the same orientation stacked together [6].

2.1 Progressive Ply Failure Analysis

Historically, most analytical assessments of layered composite parts and structures were based on the concept of First Ply Failure (FPF). In this case the laminate were assumed to be completely failed when the first ply fails. In order to realistically determine design margins, it is useful to know if a laminate that has started to fail is going to come apart catastrophically, or if it is just going to crack a little and hold together. In this sense, it could be useful perform a Progressive Ply Failure Analysis (PPFA).
A Matlab program has been developed in order to carry on the Progressive Ply Failure Analysis (PPFA) of composite laminates. Both Tsai-Hill and Tsai-Wu criterion have been chosen as a failure criterion. After a generic ply failure the lamina contribution to stiffness is neglected regardless to the failure mode. After the FPF, loads increments are proportional to the initial applied loads; the routine continues iteratively until LPF. A block diagram of the developed tool is shown in Figure 2. To validate the PPFA code a series of composite laminates have been studied and results have been compared with both experimental and FE results [7].

2.2 Fatigue

In aeronautics, fatigue of structural elements is a significant design's parameter that affects both the design phase and certification phase of aircrafts. Certification requirements for composite structures are much more conservative and are covered by guidelines issued by airworthiness bodies such as European Aviation Safety Agency [8]. The certification of composite structures is accomplished using the building block approach, which requires tests from coupons to full scale

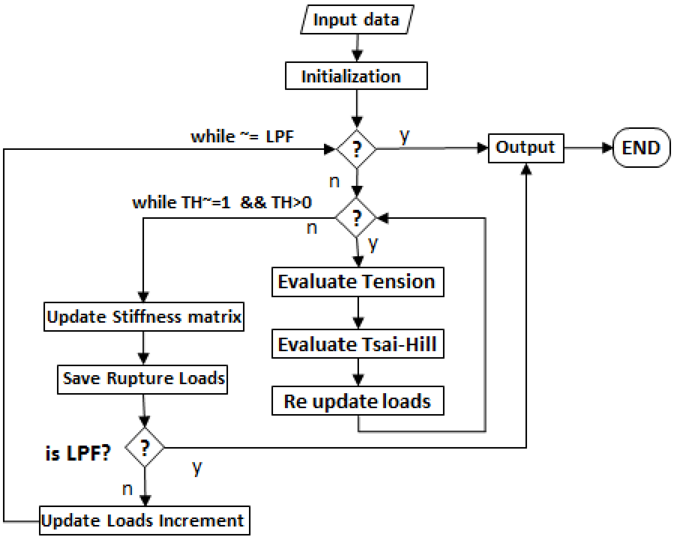

Fig. 2 Progressive Ply Failure Analysis block diagram

components. Various damage types have been classified into five categories by certification authorities based on visual/non visual detectability. Concerning fatigue, the introduction of composite materials, represents a benefit since the growth of a damage in metal is typically much more abrupt than in composite. Typical S-N curves for composite materials are flatter compared with those of isotropic materials, underling a minor influence of the effects of stress concentrations for composites and also, composite materials have a initial value of S higher than isotropic materials. Both, the increased life under cyclic loads and the higher level of specific strength, make the use of composites preferable.
The effects of BVID in the residual tensile and compressive strength has been investigated by several authors.
A first static experimental activity was carried out at the Polytechnic of Turin as a preliminary activity for the introduction of the subsequent fatigue case [9] . The experimental campaign has dealt with the effect of low velocity impact damage produced at different energy levels by the dropped mass test method. Several panels were tested to in-plane shear or uniaxial compression load to verify the effect of the damage on the buckling load, with respect to the non-impacted

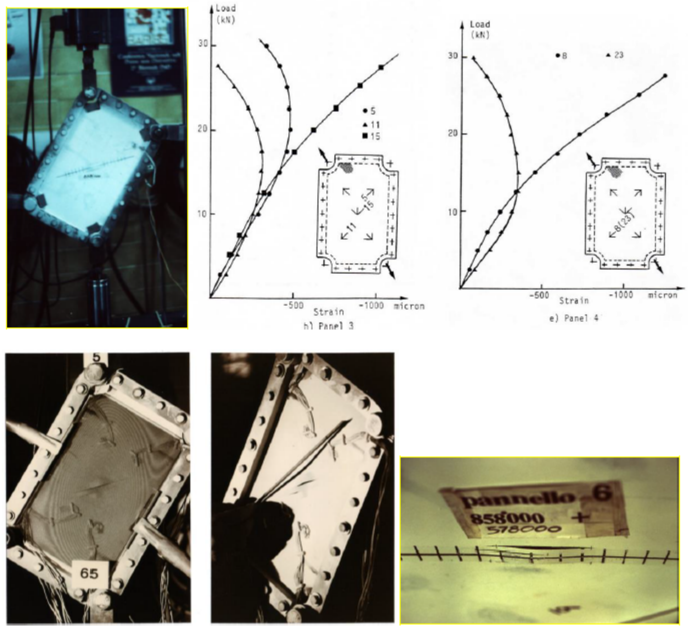

Fig. 3 Undamaged and Damaged buckling results of panels under shear loads

panels. A buckling load reduction up to 30% was obtained for panels impacted up to 41 Joule. In [9] it has been also demonstrated, both numerically and by experiments, that buckling load of panels under in-plane shear is very much affected by the positive or negative direction of load application. The maximum reduction in buckling load has been detected when a negative shear is applied. Static experimental results are reported in Figure 3 for the undamaged case (Panel 3) and damaged case (Panel 4).
A subsequent experimental activity, considering cyclical compressive load condition was performed on different type of panels (flat and stiffened) in order to investigate the effect of fatigue loads on damage propagation and structural integrity.
Both constant amplitude and spectrum fatigue can produce failure from BVID at significant lower strains, thus a fatigue threshold must be defined so that, below fatigue threshold an infinite fatigue life can be guaranteed. This fatigue threshold exist when deformation are approximately under the 60% of the static threshold, thus approximately 3000 $\mu\varepsilon$, as demonstrated in [10].This limitation has been included in the

preliminary wing box design process and optimization.

2.3 Buckling

Upper wing panels may also experience buckling; buckling is characterized by a sudden failure or a reduction of the stiffness of a structural member subjected to high compressive stress, where the actual compressive stress at the point of failure is less than the ultimate compressive stresses that the material is capable of withstanding. For simply supported plate under uniaxial compression, the critical load can be computed as follow

$$N_{x_{cr}} = \pi^2 \left[D_{11} \left(\frac{m}{a}\right)^2 + 2(D_{12}+2D_{66}) \left(\frac{1}{b}\right)^2 + D_{22} \left(\frac{1}{b}\right)^4 \left(\frac{a}{m}\right)^2 \right] \quad (1)$$

A second buckling mechanism is shear buckling. Modeling of this buckling mode has an high computational cost; however, assuming that the panel have an infinite length along x, an estimation of the critical shear load can be done using the following equation

$$\lambda_s = \begin{cases} \dfrac{4\beta_1 (D_{11}(D_{22}^3))^{1/4}}{b^2 N_{xy}} & 1 \leq \Gamma \leq \infty \\[2ex] \dfrac{4\beta_1 \sqrt{D_{22}(D_{12}+2D_{66})}}{b^2 N_{xy}} & 0 \leq \Gamma \leq 1 \end{cases} \quad (2)$$

where

$$\Gamma = \frac{\sqrt{D_{11} D_{22}}}{D_{11} + 2D_{66}}$$

The effect of combined loads, both compressive and shear, can be taken into account by the following interaction equation (Lekhnitskii).

$$\frac{1}{\lambda_t^{(m,n)}} = \frac{1}{\lambda_n^{(m,n)}} + \frac{1}{\lambda_s^2} \quad (3)$$

where λ_n is the normal buckling load factor, λ_s is the shear buckling factor and λ_t is the combined buckling load factor; λ_t is always more critical than normal buckling factor thus, among these, it will be used as an optimization parameter.
The results obtained were compared a posteriori with those obtained with a more accurate programme, EALPATAR. Here, the equations for laminated cylindrical shells (Donnell-type equations) were used in conjunction with the Galerkin method to determine the critical buckling loads.

3 Optimization strategy

From the wide range of optimization algorithms, the Genetic Algorithm (GA), belonging to the class of Evolutionary Algorithms, is one of the most useful for the optimization of laminated composite structures. GA is an abstraction of biological evolution and follows the Darwinian's law of evolution with the aim to find an optimum solution between several thousands of allowable solutions.
Different test cases have been used to validate the developed GA and its behaviour in terms of convergency and time required to converge. The results obtained during this phase were being used mainly to calibrate the termination criteria and to improve some functions in terms of computational time.Two termination criteria have been defined, one deals with convergence of the population's mean fitness to the higher value while the other regards the maximum allowable number of iterations which is fixed a priori. A block diagram of the developed GA is shown in Figure 4.

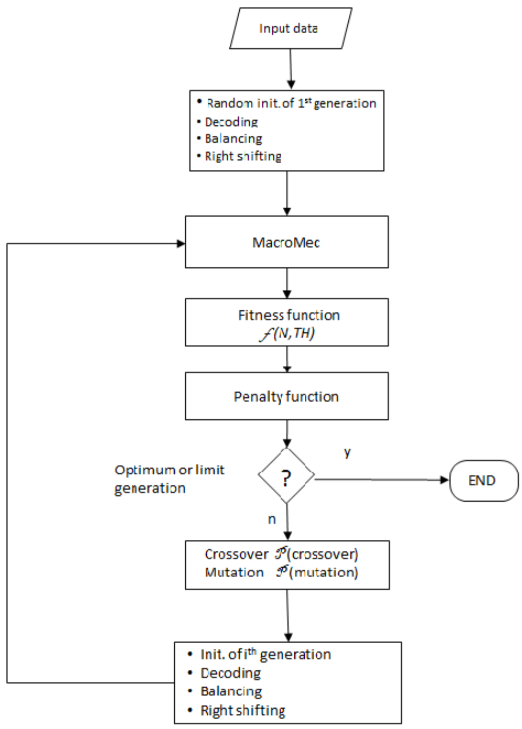

Fig. 4 Block diagram of the developed GA

3.1 Population's initialization

The first step concerns the initialization of the population. Initially many individual solutions are usually randomly generated to create an initial population. The population size depends on the nature of the problem, but typically contains several hundreds or thousands of possible solutions. Traditionally, the population is generated randomly, allowing the entire range of possible solutions, the search space. Occasionally, the solutions may be seeded in areas where optimal solutions are likely to be found. A necessary condition so that the evolution begins is that the initial population has to be feasible with regards of some criteria, typically the ones that define the environment in which the population has to survive.
A common practice in developing GAs is to encode the genes since, in this way, it is possible to move from a spread search space to a discreet one, with benefit in writing the code and in making it faster in case of further modifications. Usually, the most used discrete spaces are binary space and the integer space. For laminates optimization, since the allowable orientations are definitively more than two, the choice is mandatory. In the developed GA the encoding selected is

$$[0 \ \pm 45 \ 90] \longmapsto [1 \ 2 \ 3]$$

where $[0 \ \pm 45 \ 90]$ is the physical space while $[1 \ 2 \ 3]$ is the encoded space. Truely, also 0 belongs to the encoded space while the decoded (or physical) space remains the same. The encoded space is now

$$[1 \ 2 \ 3] \cup \{0\}$$

3.2 Selection strategy

The evolution has the aim to improve the mean fitness of the population in order to find the best individual(s). The search is performed on the basis of some criteria, usually defined in terms of fitness function, and favors individuals that are better performing by means of selection strategy. The selection method chosen for the developed GA is the fitness proportionate selection, also known as roulette wheel selection. This fitness level is used to associate a probability of selection to each chromosome. If $\tilde{f}_i$ is the fitness of the $i-th$ individual in the population, its probability of being selected is given as

$$P_{i,rep} = \frac{\tilde{f}_i}{\sum_{k=1}^{n} \tilde{f}_i} \tag{4}$$

where $\tilde{f}_i$ is the value of the fitness function of the i-th chromosome and n is the number of chromosome in the population.
Since the objective of the optimization is the weight reduction of a composite laminate, the fitness function has been chosen being function of the panel weight (or number of laminae, if they are made of the same material) and of the failure index. The fitness function is given by the following equation

$$\tilde{f} = \frac{1}{TH \cdot q^5} \tag{5}$$

where TH is the Tsai-Hill index and q is the number of plies of the laminate. Once the probability of reproduction of each laminate is known, the algorithm compute the number of couplings that each panel could make. The latter is proportional to the fitness of the laminate, the higher the fitness the higher the number of sons generated. The number of sons that each laminate could generate is given by the following equation

$$n_{son} = ceil(P_{rep} \cdot n) \tag{6}$$

where n is the size of the population or, in other words, the number of chromosomes, while ceil means that the number is rounded to the next integer.
Since it has been chosen to have a population with a fixed number of individuals, the algorithm is able to check the size of population, allowing or inhibiting respectively that best or worst individuals could reproduce. Also elitism is allowed.

3.3 Genetic modification

There are different kind of mutations, some takes inspiration directly from the natural world while others are created intentionally for special purposes. As happens in nature, for each chromosome, the probability of undergoing a genetic modification is low. Generally, in GAs probability of mutation is fixed at $20 \div 30\%$. The set of genetic mutations that has been defined is composed as follow

- **mutation**, changes randomly a gene of the laminate. The position of gene that undergoes the mutation is randomly chosen but with respect to the maximum dimension of the chromosome;
- **swap**, exchanges the position of two genes of the chromosome. Also in this case, the position of genes that undergo the modification and the string length is chosen randomly;
- **deletion**, deletes a randomly chosen gene from the chromosome. In this case, in the encoded laminate appears code 0 mentioned above while, in the encoded one, the correspondence gene have to be removed.

Each one of the previous listed mutations is associated to an integer, the kind of mutation which the chromosome could undergoes is randomly chosen by the algorithm. Each mutation has the same probability to be selected, in this case fixed to 10%.
As above underlined, some functions are required for conditioning a particular problem. A function that balances the laminate's lay-up has been developed. This function checks if the laminate is balanced if not, the function counts the number of $\pm\theta$ plies and randomly adds or removes a ply to balance the laminate. If the function removes a ply, in the encoded laminate appears the code 0. This means that this ply does not exist in the physical laminate since the code 0 has not a corresponding orientation in the physical space. In this case, and in general each time that function deletion has been invocated, is necessary to check if there are some zeroes in the encoded laminate and, if so, shift them to the queue of the encoded laminate and delete them to the chromosome before decoding the laminate lay-up.

3.4 Penalty functions

The straightforward application of GAs to Constraint Optimization Problems (COP) is not possible due to the additional requirement that a set of constraints must be satisfied. Difficulties may arise as the objective function may be undefined for some (or all) infeasible elements, and an informative measure of the degree of infeasibility of a given candidate solution may not be easily defined. Finally, for some real-world problems, the check for feasibility can be more expensive than the computation of the objective function itself. As a result penalty techniques are commonly used; however they require considerable domain knowledge and experimentation in each particular application in order to be effective. It must be underlined that can be difficult to find a penalty function that is an effective and efficient surrogate for the missing constraints. The effort required to tune the penalty function to a given problem instance may negate any gains in eventual solution quality. Much of the difficulty arises because the optimal solution will frequently lie on the boundary of the feasible region. Many of the solutions most similar to the genotype of the optimum solution will be infeasible. Therefore, restricting the search to only feasible solutions or imposing very severe penalties makes it difficult to find the schemata that will drive the population toward the optimum. Conversely, if the penalty is not severe enough, then too large a region is searched and much of the search time will be used to explore regions far from the feasible region. Then, the search will tend to stall outside the feasible region.

3.4.1 Limit strain penalty

A multiplicative penalty function, shown in equation 7, has been defined to take into account strains constraint

$$\mathcal{P}_\varepsilon = \begin{cases} 1 & \varepsilon < \varepsilon_{lim} \\ \prod\limits_{1 \le k \le n} \left(2 - \frac{\varepsilon}{\varepsilon_{lim}}\right) & \varepsilon_{lim} \le \varepsilon \le 2\varepsilon_{lim} \\ 0 & \varepsilon > 2\varepsilon_{lim} \end{cases} \quad (7)$$

where the product is extended to each ply of which the deformation are higher than the chosen limit. It could be noticed that the penalty function becomes zero if one (or more) ply has a deformation higher than two times the limit value. It also must noticed that in the equation (7), ε must be understood as the module of ε.

3.4.2 Buckling penalty

Extensive analyses were conducted about buckling and post-buckling behaviour of composite laminates [11]. These analyses have shown that composite panels might ensure a good strength also in post-buckling field however, regulations does not allow that composite panels work in this field. For this reason, a death penalty has been defined.

$$\mathcal{P}_\lambda = \begin{cases} 1 & \lambda_t < 1 \\ 0 & \lambda_t \geq 1 \end{cases} \tag{8}$$

where λ_t is computed as shown in equation 3.

3.4.3 Matrix cracking penalty

The selected failure index does not distinguish between different failure modes since it is quadratic. In sections 2 and 2.2 has been underlined how matrix cracking could negative affect laminate's stiffness. To prevent matrix cracking, is a common practice to avoid that more than four contiguous plies have the same orientation [12, 13]. This stacking's rule has been implemented by a multiplicative penalty function defined as follow

$$\mathcal{P}_{mc} = \begin{cases} 1 & c < 5 \\ \sum_j \frac{1}{2(c-3)} & c \geq 5 \end{cases} \tag{9}$$

where c is the number of contiguously plies that have the same orientation and j is the number of sub-laminates that violates the matrix cracking constraint.

The choice of multiplicative penalty function is justified to the fact that the structural beam model to determine the stress state gives mostly normal loads N_x and shear loads N_{xy} so, the effect of the off-axis loads, which are one of the most responsible of matrix cracking, can be neglected.

3.5 Objective function

The resulting objective function is shown in the following equation

$$f = \tilde{f}\, \mathcal{P}_\lambda\, \mathcal{P}_\varepsilon^8\, \mathcal{P}_{mc} \tag{10}$$

It can be noticed that, two of the defined penalty (matrix cracking and maximum strains) are indirect, i.e. they enlarge the search space also to the non feasible region. However to avoid that the penalty function enlarge too much the search space, the strain's penalty function is raised to eight, in this way this penalty becomes non linear and favours the laminates with strain around limit level.

4 Parallel computing and distributed computing

One of the major disadvantage of GAs is that, finding the optimal solution to complex high dimensional and multimodal problems requires very expensive fitness function evaluations. In real world problems such as structural optimization problems, a single function evaluation may require several hours to several days of complete simulation. A way to improve the algorithm efficiency, in term of computational speed, is the parallelization.

Optimally, the speed-up from parallelization would be linear. However, very few parallel algorithms achieve optimal speed-up. Most of them have a near-linear speed-up for small numbers of processing elements, which flattens out into a constant value for large numbers of processing elements. The MDCS (MATLAB®Distributed Computing Server) has been used to create a cluster of 12 workers in which the master computer acts itself as a worker during the optimization. Remarkable gain in term of computational time has been reached. One step of optimization, with the whole set of penalities activated, requires approximately 20 minutes of computational time.

5 Results

The highest structural efficiencies are required to minimize the airframe weight and increase the payload mass. They are obtained by wide use of high modulus graphite/epoxy material to obtain a very light and high stiffened structure with good aeroelastic behaviour. The use of carbon fibres (T300/Epoxy), with high moduli and strengths ($E_{11} = 181GPa$, $E_{22} = 10.3GPa$, $G_{12} = 7.17GPa$, $\nu_{12} = 0.28$, $X = 1500MPa$, $Y = 40MPa$, $S = 68MPa$,$X' = -1500MPa$,$Y' = -246MPa$, $\alpha_1 = -0.2 \cdot 10^{-7}$, $\alpha_2 = 2.5 \cdot 10^{-5}$, $\beta_1 = 3.85 \cdot 10^{-5}$, $\beta_2 = 3.1 \cdot 10^{-3}$), can potentially allow a weight reduction with respect to traditional aircraft constructions.

For the adopted configuration, a good wing structural solution to enhance the bending/torsional stiffness, could be represented by an opportunely reinforced high modulus carbon fibre leading edge wing-box as shown in Figure 5. The set of allowable orientations is $[0/\pm45/90]$ as listed in section 3.1. The load condition considered in this

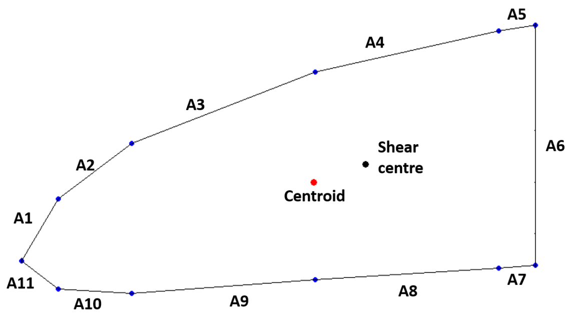

Fig. 5 Wing box cross section discretization, section A (root section)

work, concerns the D point of the flight envelope ($n = 3$, $V_{EAS} = 47km/h$).As shown in [2, 14], the presence of large tip deflection can significantly influence the aeroelastic properties of the wing. In order to satisfy the stiffness requirments the maximum allowable tip deflection has been preliminarly limited to 1/10 of wing span at the ultimate load.

After some oscillations during the first iterations, at the 10-th iteration the convergence starts and was completely reached at iteration 20. The maximum number of plies is reached in the root's region, in the panel (**A4**) and its optimal lay-up is $[\pm45/0/\pm45/45/0/\mp45/0_2/-45/0_2/90/0_3/45/0_3/90/0_4/-45]_S$. Herein the buckling penalty has the major effect thus the required total thickness is 7 mm, considering a ply thickness fixed to 0.125mm. In Figure 6 is shown, for the panel 4, the total number of the layers and the variation of the three possible orientations as a function of the wing span. In each section, panel 4 is the most stressed and it is the one most sensible to the buckling. A notable reduction in 0° plies is shown from root section to tip section due to the variation of bending moment; the slope of ±45 curve is lower than the 0° curve because of the lower shear flows; while for 90°, the curve is pratically flat since the matrix cracking penalty, which guides the presence of 90°, has a minor influence on optimization. This penalty is activated only when, in the stacking sequence, there are more than four plies with same orientation; for example, if there are more than four 0° layers, the minor influence on fitness function is obtained adding only one 90° ply instead of two layers ±45 to have a balanced laminate. In Figure 7 are depicted the minum number of plies obtained with the optimal configuration. Maximum deflection at D point, considering a safety factor of 1.5 ($n_{ult} = 4.5$), is 6.81 m at the tip of the wing as show in Figure 8. The weight trend vs optimization iterations is reported in Table 2, the minum wing box weight is 160 Kg. These results, both tip deflection and weight, are in a good agreement with weight estimation formulas for solar-powered HALE airplanes and with the results obtained in [3] where a main spar sandwich reinforced tube made of

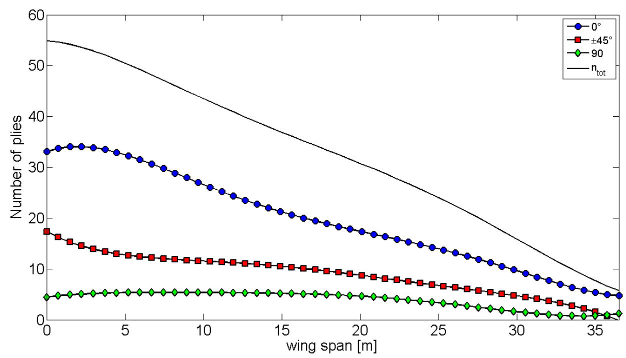

Fig. 6 Number of plies per each orientation and total number of plies along wing span for panel 4

Iteration	Weight [Kg]	Tip deflection [m]	Twist angle [deg]
1	212	3.346	1.037
2	169	6.980	5.468
3	170	6.976	3.120
4	172	6.421	1.806
5	165	6.646	1.244
6	165	6.751	1.676
7	160	6.738	2.456
8	166	6.398	2.260
9	163	6.697	1.207
10	160	6.650	1.514
⋮	⋮	⋮	⋮
19	160	6.883	1.596
20	160	6.812	1.164

Table 2 Weight, tip deflection and twist angle vs optimization iterations

M55J graphite/epoxy pre-preg tape and Korex or Nomex honeycomb materials was considered.

6 Conclusion

A Genetic Algorithm has been developed to optimize the wing structure of innovative composite aircraft. A set of penalty functions has been activated during the optimization process in order to guarantee adequate stiffness and strength requirements and to avoid local or global buckling. A limit of 3000 $\mu\varepsilon$ has been considered to cope with static and fatigue behaviour of damaged composite structures. An optimal lay-up configuration has been identified for a solar powered HALE UAV. A maximum deflection of 6.81 m is reached at the ultimate load; the optimized wing box weight is 160 Kg with maximum number of plies of 56 at the root section. The results are in good agreement with those reported in literature for same kind of aircraft.

GAs, as well as other conventional optimization strategies, for composite wing boxes results in a piecewise thickness distribution through the entire structure. The discontinuities of plies in boundary between the adjacent laminates may result in stress concentrations, loss of structural integrity and also increase manufacturing difficulties and costs. A two level optimization strategy has to be considered for the final design to meet also technological requirements as well as lower the manufacturing costs.

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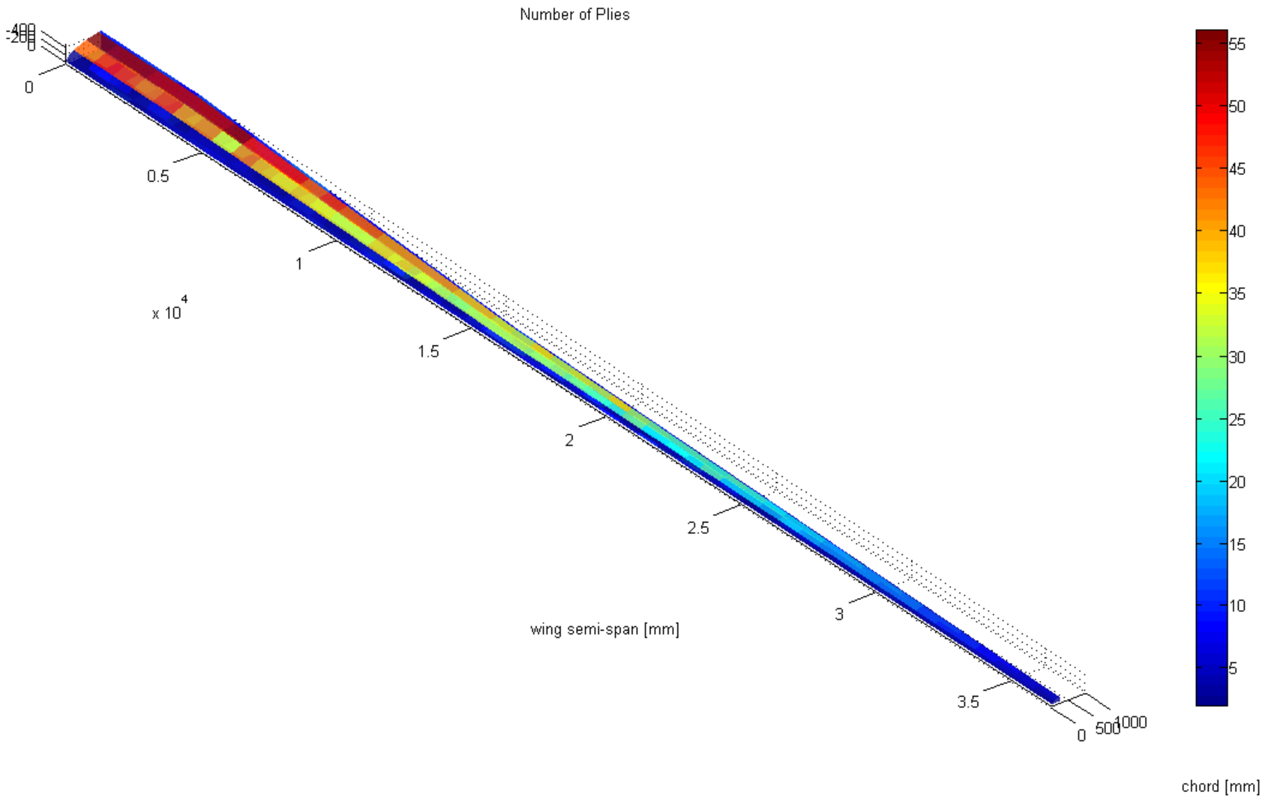

Fig. 7 Number of plies at iteration 20

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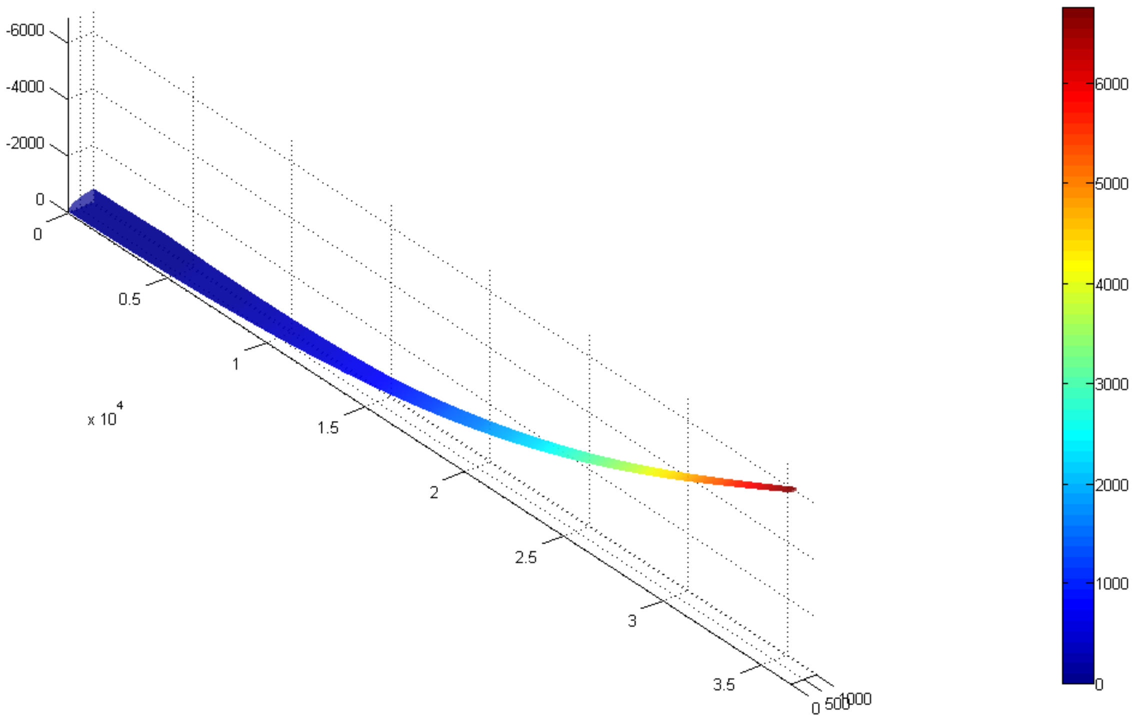

Fig. 8 Wing box deflection at iteration 20