COMPOSITE WING DESIGN UNDER STOCHASTIC DYNAMIC LOADING

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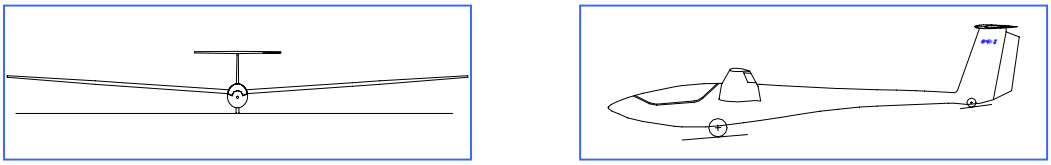
SUMMARY : Minimum weight is desired, in normal aircraft, as a performance requirement. The use of composite materials permits a considerable weight reduction with an incremental material price due to high cost of these materials. A good design approach including aerodynamic and structural non-linearities is necessary to optimise the material quantity. The purpose of the present work is to show the use and integration of different commercial software to perform these calculations during the design stage, from aerodynamics to fatigue life estimations based on atmospheric turbulence models. In addition an approach to obtain the cycle counting for fatigue life prediction based on air turbulence excitation frequencies is presented.

KEYWORDS : design composites materials, fatigue of composites, non-linear finite element, optimisation, random loading.

AERODYNAMIC MODELING

Aerodynamic is first step in sailplane design. All dimensions are adopted based on international standards [World Class Sailplane (IGC)(FAI) and JAR22 (JAA)] and own design criteria [stall velocities, maximum Lift/Drag ratios,…]. The main dimensions are shown in fig. 1.



Wing span : 16 m Wing Area : 11.13m^2 Length : 6.99 m Estimated weight : 350 kg

Fig. 1 Some views of the sailplane and main dimensions

Using concepts of flight dynamics for estimating sailplane movement as a function of aerodynamic forces [1][2] a whole model of the sailplane is generated in MATLAB. For this purpose the sailplane is considered as a rigid body with no aeroelasticity effects. This model is used to check and improve the aerodynamics in terms of both, perfomances and stability. Typical sailplane polars are shown in fig. 2.

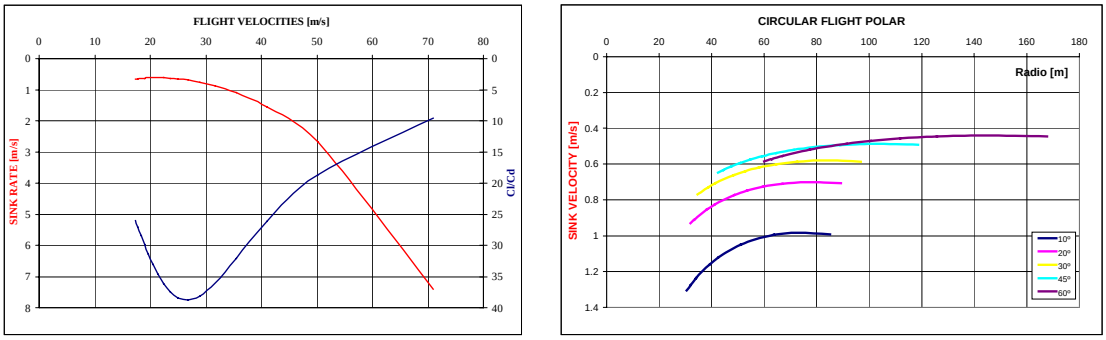


Fig. 2 Horizontal and Circular Flight Polar at sea level

LOAD ANALYSIS

European standards JAR22 [3] for sailplane design define general maneuver diagrams, discrete gust envelope and ground operations to determine limit load and ultimate load cases which cover all possible sailplane operations, including some dynamic aspects in a roughly manner. Sailplane structure will be design to observe these load cases.

The sailplane model is used to generate several surfaces containing all the information about these load cases : basically load values for different velocities and positions alongside the sailplane. One of the bending moment surfaces is shown in fig. 3.

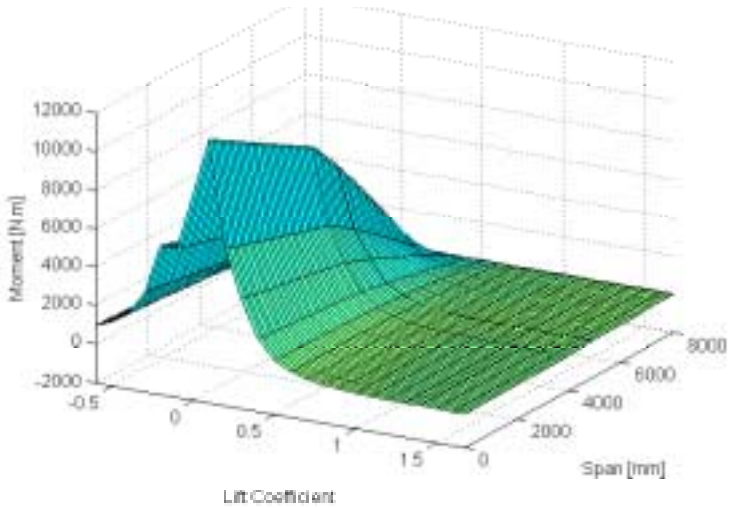


Fig.3 Bending Moment [N.m] due to Lift along wing span for different sailplane velocities

Focusing analysis on the wing, with 16 m span, all actions could be expressed as a linear combination of : a couple of bending moments and shear forces in two principal axes transversally to wing axe and a torsional moment along this axe. Based on surface information it could be determine for every point of the wing a whole set of actions that appears at same time depending on the velocity. This information will be used later for laminate optimisation at every section of the wing beam. The whole set of actions that happens at wing root for different velocities is shown in fig. 4.

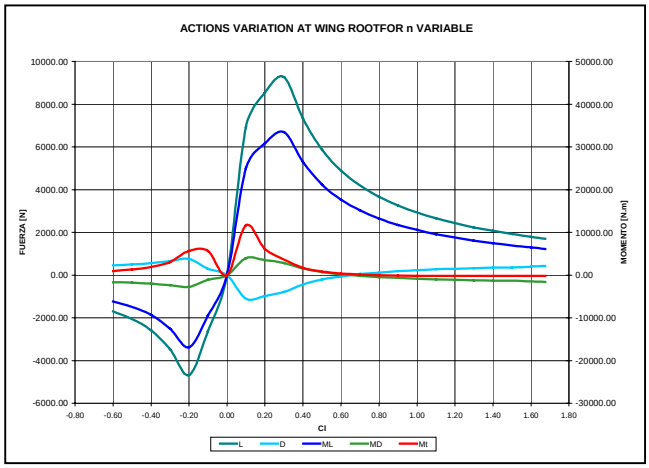


Fig. 4 Actions at wing root for different velocities

STRUCTURAL PRELIMINARY DESIGN

The wing structure is composed of a square beam, ribs and skins. Wing beam is thought as a rigid structure for supporting all the aerodynamic loads and ground operations, whereas ribs and skin laminates are designed to obtain flexible parts whose functions consist in keeping the aerodynamic shape and transmit all the aerodynamic loads to the wing beam with enough strength and strain against impacts. For this reason carbon fibre is used in responsible parts as the beam, and glass fibre is used for the rest of the structure. The main reason to use Carbon Fibre is its high strength-weight ratio and its fatigue behaviour.

The stacking sequences and the thickness are varied along the wing with aim of obtaining optimum weight, taking into account structural and production considerations. For this purpose a own developed MATLAB routine is used to study and optimise laminate sequences at every section of the wing beam based on load information.

As a consequence of the study of the airfoil available space and load types it is adopted a rectangular cross sectional area due to its bending behaviour and enough characteristics in torsion. The beam position along airfoil chord is defined to increase the bending effects instead the torsional ones.

An analysis of the different loads ratios is done to determinate laminates sequences. The next laminate constrains are adopted : first it has to be symmetrical to avoid displacements and rotations coupled; second the ply angles have to be sequential to minimise free edge effects in delamination and fatigue life; and finally to obtain a balanced laminate it must have repeat sublaminates. Taking into account these criteria a sublaminate of [0_3/45/90/-45] is adopted.

The optimization scheme is as follows : the beam is divided into parts with short length. All of these sections are checked for all the load cases and the global laminate is optimised based on several rules implemented in the algorithm. At every section the laminate is checked and modified for all load cases until it will reach the established failure criteria. During this stage the well known Classical Laminated Theory is used :

$$\left[\frac{N}{M}\right]=\left[\begin{array}{c|c}A & B\\ \hline C & D\end{array}\right]\times\left[\frac{\varepsilon}{\kappa}\right]$$

Several failure criteria are available : Maximum Strain, Maximum Stress, Tsai-Hill, Tsai-Wu, etc… [4][5].

Initially the laminate is based on a [0_3/45/90/-45] sublaminate and number of sublaminates is limited to 4. Additionally an 0º laminas are added in sections where it is needed to increase bending stiffness and due to fact that bending moments are greater in one of the principal axes the material increase is different at every beam wall. Finally a minimum thickness of 2.4 mm is adopted as design criteria.

Based on these constrains the final laminate sequence is obtained from the algorithm, taking into account a gradual variation in lamina quantity to minimise interlaminar effects in thickness variation. The laminate varies from $[0_6[0_3/90/45/-45]_2]_S$ at the root to $[0_3/45/90/-45]_S$ at the tip with 7.2 mm and 2.4 mm thickness respectively. The optimise stacking sequence for the upper wall is shown in fig. 5.

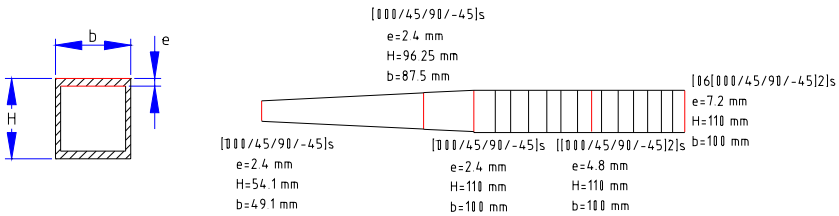


Fig. 5 Final Laminate Sequence for the upper wall

To validate this design a F.E model of the beam is done with MARC commercial code. All information about actions distribution, load cases and laminate sequences is transferred from MATLAB model to MARC code via a own developed Fortran routine. The whole model consists of 4 node shell elements with bilinear interpolation. The material behaviour is considered orthotropic and the quadratic Tsai-Wu Fail Criteria is used[6].
The beam model is analysed with 23 static cases obtained from the load analysis. During calculations the beam is considered isolated and no effects of ribs and skin are taken into account.
Results data show a maximum fail index of 93.55 % for one of the extreme loads with 5.3g acceleration; at this situation the maximum vertical displacement is 3100 mm. Figure 6 shows fail index distribution where the effects of laminate optimisation could see in regions near the root where index values are very similar.

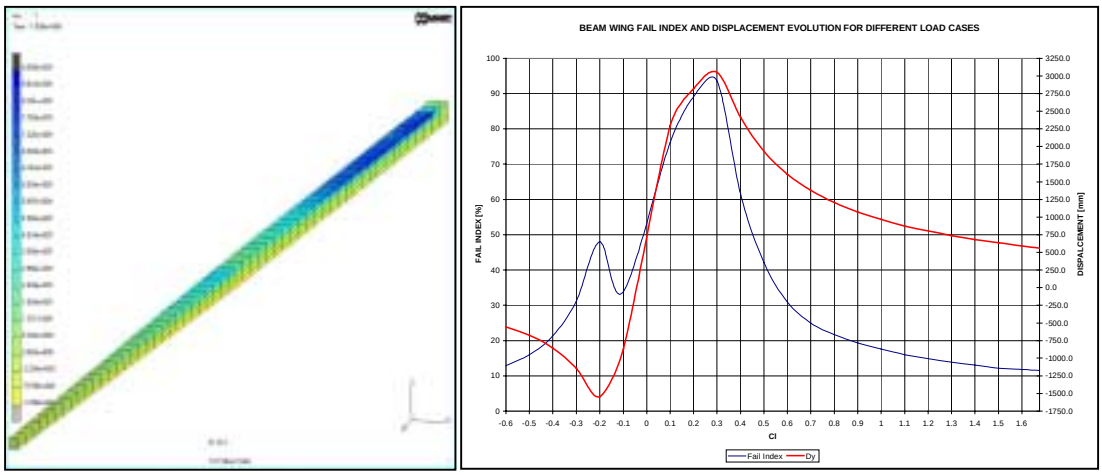


Fig. 6 Fail index evolution at wing beam

As a relevant conclusion the laminate sequence is validated but stress values for extreme loads have to be analysed in more detail with the whole wing model to determine operational and maintenance conditions of the glider. Since the effects of ribs and skin are not considered, both stresses and displacement are expected to be less.
The ribs with lengths between 800 mm at wing root to 387 mm at wing tip are considered as a flexible element in torsional work whit enough strength to transmit the aerodynamic loads from the skin to the wing beam with no shape variations. There are 3 holes inside ribs: two circular for airbrake and ailerons systems with 60 mm diameter and another rectangular one for the wing beam. Stacking sequences are defined with the objetive of obtaining bending flexibility and enough strength for supporting the torsional moments of the aerodynamic loads. A global laminate of $[45/-45]_{2S}$ in carbon/epoxy is defined with 1.6 mm thickness.
A F.E model of 4 nodes shells is used to check the ribs characteristics, they are checked for extreme loads cases for determining the shape variation. Details of ribs are shown in fig.7.

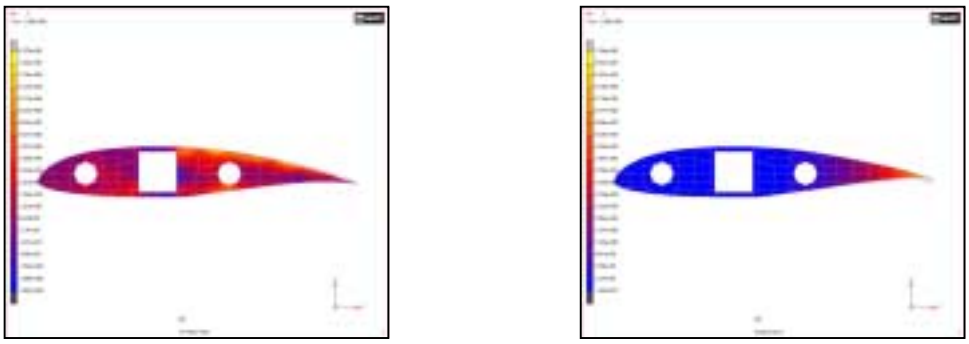


Fig. 7 Details of ribs behaviour : deformed shape and fail index for different load cases

Maximum displacement values appear at the aft part of the rib with a maximum displacement of 7.3 mm for a extreme load of 4g force. These deformed shape affects mainly the airfoil part in which for the majority of attack angles the laminar transition is done**.** Based on these reasons no airfoil aerodynamic behaviour variation is to be expected. Although no great fail index appears on free edges at circular holes they will be checked in the whole model to take into account the global effects.

The skin laminate is designed to keep the aerodynamic shape of the wing with enough flexibility to transmit the loads to the ribs. The design criteria to define the laminate are quite similar to those of the beam and ribs. Bending and torsional stresses have to be supporting with no 0º laminates due to the overload that they produce on the skin. A minimum thickness of 2 mm is fixed as a design criteria for reasons of impact. Finallly a $[45/90/-45/45/-45]_S$ laminate is adopted.

F.E. MODELLING

A FE model of the whole wing was created in MARC in order to validate and perform a more accurate analysis. The use of FEA capabilities permits the study in greater detail of the structure. In particular it will be studied the global fail index variation under different criteria, total displacements, interlaminar stresses, delamination, structural buckling, and non linear progressive failure. Moreover, even more detailed FE models will be analysed in relevant areas.

A total of 5073 elements are used to modelize all the stacking sequences and the whole geometry of the wing. Some views of the model are shown in fig. 8

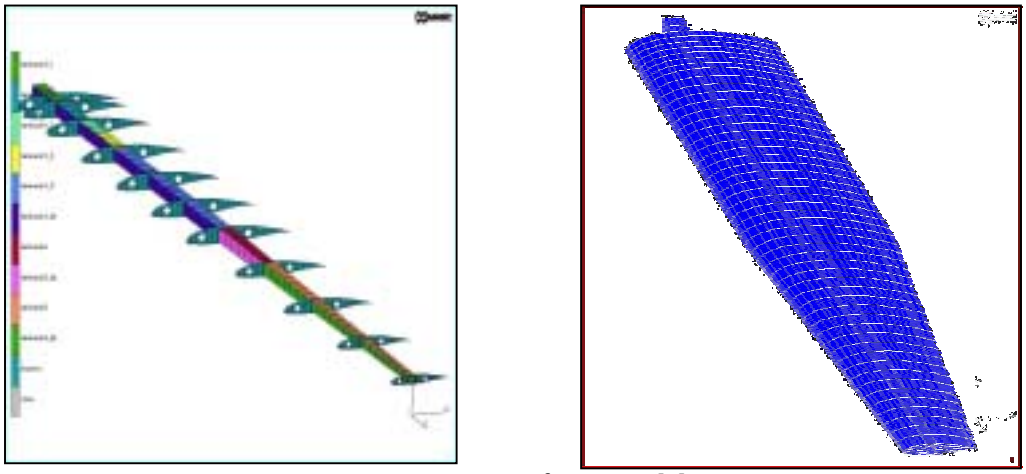


Fig 8 Views of F.E. model

All elements are four-node isoparametric shell elements with bilinear interpolation. The numeric integration is done trough thickness and mathematics of these elements have a three dimensional approach that is enough to reproduce in-plane stresses in each lamina and interlaminar shear stresses : τ_{xz},τ_{yz} [6]. The way MARC obtain τ_{xz},τ_{yz} is very accurate : inside the element formulation there is a parabolic distribution of transverse shear strain instead the normal constant distribution. It is demonstrated that this approximation gives better results in interlaminar shear stresses [7].

The material behaviour is considered orthotropic with the values shown in fig. 9.

E_{11}	E_{22}	E_{33}	μ_{12}	μ_{23}	μ_{31}	G_{12}	G_{23}	G_{31}
110000	9000	9000	0.02	0.525	0.02	4000	2800	4000
S_{XTmax}	S_{XCmax}	S_{YTmax}	S_{YCmax}	S_{Ztmax}	S_{ZCmax}	T_{XYmax}	T_{YZmax}	T_{ZXmax}
1500	1000	80	200	80	200	100	30	30

Fig. 9 Values in MPa for a carbon/epoxy prepeg lamina

For failuring analysis Tsai-Wu quadratic criteria is adopted mainly for the capacity of distinguish between compressive and tensile strengths of a lamina and the integration of the three unidirectional lamina strength parameters. The criteria is evaluated at every lamina and at every integration point because Marc performs trough thickness numeric integration.[7]
The whole model is used to check a total of 23 static load cases of the different sailplane perfomances. Load data are transferred directly, with Fortran code, from MATLAB model as in the beam preliminary design.
Results were carefully analysed for displacement and fail index variation on different substructure of the wing : a maximum displacement of 1991 mm is obtained with longitudinal variation that could be considered uniform. The skin aerodynamic shape variation should be enough to keep the operation safe. The inclusion of all the laminas deformations for limit loads in the elastic field permits operations with no permanent deformation. Regarding the stress for different load cases, limit and ultimate, stresses values were found below failure limits and no static failure at ultimate loads was found. Stresses at beam wing are less than expected in preliminary design due to ribs and skin effects. On the other hand a overload appears at wing root in the joints between root rib and the skin due to the different substructure strength ratio . Despite this, the flexibility and strength ratio between beam and ribs with skin is considered sufficient for security reasons in thickness minimum values. Some details are shown in fig. 9.

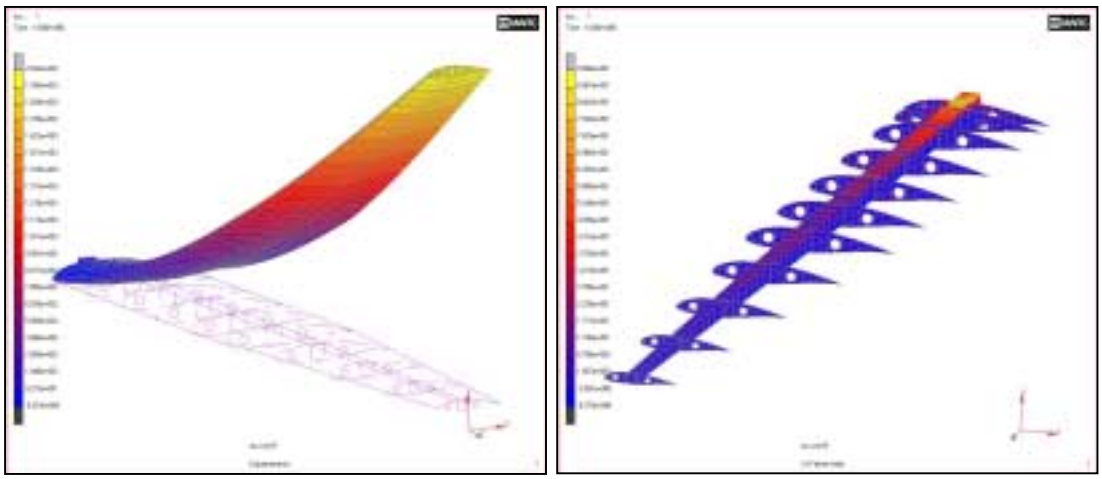


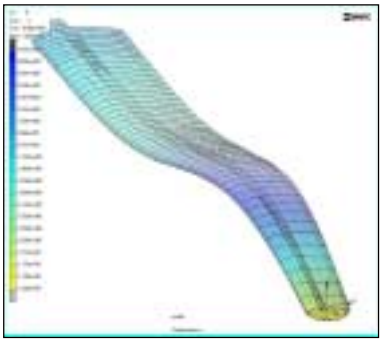
Fig. 9 Wing displacement and fail index for one load case

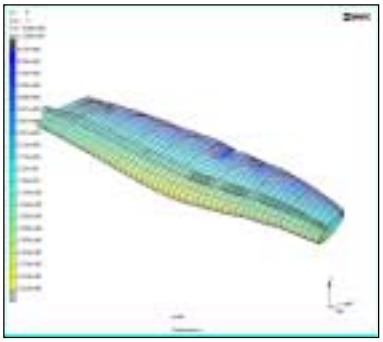
To study delamination a maximum value of 15 MPa is fixed as a design criteria for all the interlaminar stresses, τ_{xz},τ_{yz} , focusing the analysis on free edges. They are checked at all normal flight cases and stresses below 15 MPa are obtained. At extreme load cases a maximum of 35 MPa is obtained at wing rib root, this value is considered inside safety range but after maneuvers higher than 5.3g the wing root has to be checked.
Buckling and torsional buckling effects are studied with a non-linear progressive failure analysis taking into account the effects of broken laminas. This type of analysis considers several decreases in the lamina strength when it is broken, so it is useful to determine the structure instabilities. For the wing case no significant buckling is observed at extreme loads.

DYNAMIC CALCULATIONS

The study of structural dynamics in flexible structures is quite necessary to analyse the real behaviour of the structure due to non-stationary loads. The origin of these loads are gust velocities, discrete or continuos, inertial effects during sailplane manoeuvres, and others. Large composite structures as the wing usually are very flexible and their natural frequencies are close to excitation frequencies of loads. These non-stationary loads could lead wing to instabilities and damage due to actions that are greater than the static ones.

First step in dynamic analysis is to obtain several modes and associated frequencies of the wing to identify the structure natural way of behaviour. This information is useful to determine critical frequencies or mode shapes in the wing structure. Considering the wing structure undamped, the F.E. model is used to obtain the 10 first modes. Some of the modes are shown in fig. 10.





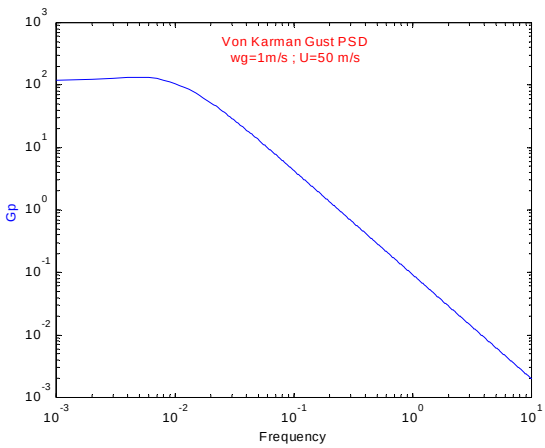
Mode	1	2	3	4	5	6	7	8	9	10
Hz	2.36	5.98	10.50	25.90	30.77	40.75	45.94	51.39	56.04	58.13
Type	Bending	Bending	Bending	Bending	Torsion	Mixed	Mixed	Mixed	Mixed	Mixed

Fig. 10 Some mode shapes and their associated frequencies

An analysis of the results shows the first natural frequency at 2.36 Hz in a bending mode. This low frequency value appears due to high flexibility of wing structure. Assuming that wind energy frequency content is inside the range 0-1 Hz, the proximity in frequency values of the loads and the natural response of the wing, produces amplification of load effects which have to be analysed in greater detail through a Force Response analysis. On the other hand first torsional mode appears at 30 Hz so no big flutter problems is expected.

In order to perform a Force Response analysis it is necessary to analyse the interaction between the wing and the air to obtain the real aerodynamic forces. These forces at every time depend on sailplane velocity, gust velocities induced by air turbulence, and the wing response (velocities and deformed shape).

Air turbulence and its induced gust is a stochastic process that could be modelled trough Von Karman (VKM) models for the air turbulence velocity [1]. Basically the model is expressed as a Power Spectral Density (PSD) of gust velocities in the frequency domain. This PSD is obtained for every sailplane velocity. The general VKM shape is shown in fig. 11.



$$G_p(f) = w_g^2 \frac{\alpha}{\pi} \frac{\left[1 + \frac{8}{3}(1.399\alpha f)^2\right]}{\left[1 + (1.399\alpha f)^2\right]^{\frac{11}{6}}}$$

$G_p(f)$ = V. K. Atmospheric Turbulence PSD

w_g^2 = RMS Gust Velocity

$\alpha = 2\pi L/U$

L = Scale of Turbulence

Fig. 11 Von Karman PSD expression and shape for U=50 m/s

Assuming the turbulence as a Gaussian process, the PSD is used to obtain a stochastic equivalent time series of gust velocities. It must keep in mind that these series only reproduce a time signal whose PSD is the same that the Von Karman one, doing the Root Main Square of PSD as the Standard Deviation of gust velocities. In fig. 12 the equivalent time series of one of the PSD is shown.

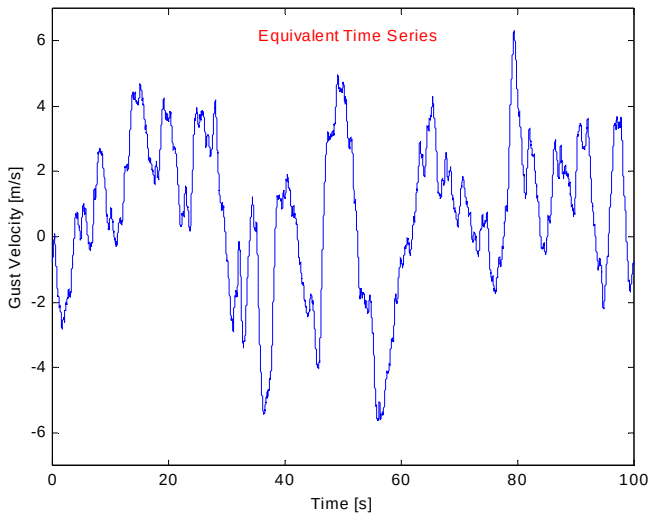
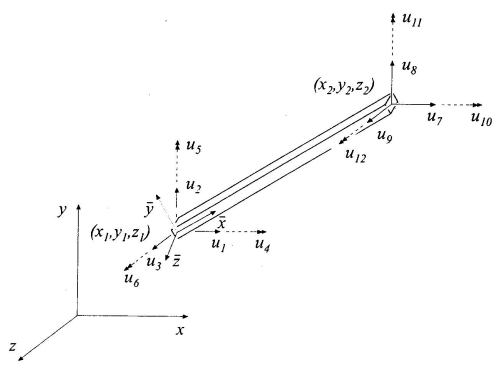


Fig. 12 Time series for one of the PSD.

Combining these time series of gust velocities with wing velocities and torsion angles at every section the whole set of aerodynamic forces is obtained at every time. In addition the associated actions over the wing could be obtained. It is used a own FE model (developed in MATLAB) to carry out these calculations. The model is a simplified one based on 3D beam elements formulation with 6 degrees of freedom per node : 4 for bending, 1 for torsion and 1 for axial loading as could be seen in fig. 13. For this latter element all the interpolation functions are linear.



$$
\begin{matrix}
k_1 & 0 & 0 & 0 & 0 & 0 & -k_1 & 0 & 0 & 0 & 0 & 0 \\
0 & \frac{12EI_{\bar{z}}}{L^3} & 0 & 0 & 0 & \frac{6EI_{\bar{z}}}{L^2} & 0 & -\frac{12EI_{\bar{z}}}{L^3} & 0 & 0 & 0 & \frac{6EI_{\bar{z}}}{L^2} \\
0 & 0 & \frac{12EI_{\bar{y}}}{L^3} & 0 & -\frac{6EI_{\bar{y}}}{L^2} & 0 & 0 & 0 & -\frac{12EI_{\bar{y}}}{L^3} & 0 & -\frac{6EI_{\bar{y}}}{L^2} & 0 \\
0 & 0 & 0 & k_2 & 0 & 0 & 0 & 0 & 0 & -k_2 & 0 & 0 \\
0 & 0 & -\frac{6EI_{\bar{y}}}{L^2} & 0 & \frac{4EI_{\bar{y}}}{L} & 0 & 0 & 0 & \frac{6EI_{\bar{y}}}{L^2} & 0 & \frac{2EI_{\bar{y}}}{L} & 0 \\
0 & \frac{6EI_{\bar{z}}}{L^2} & 0 & 0 & 0 & \frac{4EI_{\bar{z}}}{L} & 0 & -\frac{6EI_{\bar{z}}}{L^2} & 0 & 0 & 0 & \frac{2EI_{\bar{z}}}{L} \\
-k_1 & 0 & 0 & 0 & 0 & 0 & k_1 & 0 & 0 & 0 & 0 & 0 \\
0 & -\frac{12EI_{\bar{z}}}{L^3} & 0 & 0 & 0 & -\frac{6EI_{\bar{z}}}{L^2} & 0 & \frac{12EI_{\bar{z}}}{L^3} & 0 & 0 & 0 & -\frac{6EI_{\bar{z}}}{L^2} \\
0 & 0 & -\frac{12EI_{\bar{y}}}{L^3} & 0 & \frac{6EI_{\bar{y}}}{L^2} & 0 & 0 & 0 & \frac{12EI_{\bar{y}}}{L^3} & 0 & \frac{6EI_{\bar{y}}}{L^2} & 0 \\
0 & 0 & 0 & -k_2 & 0 & 0 & 0 & 0 & 0 & k_2 & 0 & 0 \\
0 & 0 & -\frac{6EI_{\bar{y}}}{L^2} & 0 & \frac{2EI_{\bar{y}}}{L} & 0 & 0 & 0 & \frac{6EI_{\bar{y}}}{L^2} & 0 & \frac{4EI_{\bar{y}}}{L} & 0 \\
0 & \frac{6EI_{\bar{z}}}{L^2} & 0 & 0 & 0 & \frac{2EI_{\bar{z}}}{L} & 0 & -\frac{6EI_{\bar{z}}}{L^2} & 0 & 0 & 0 & \frac{4EI_{\bar{z}}}{L}
\end{matrix}
$$

u_i : degrees of freedom in global symtem

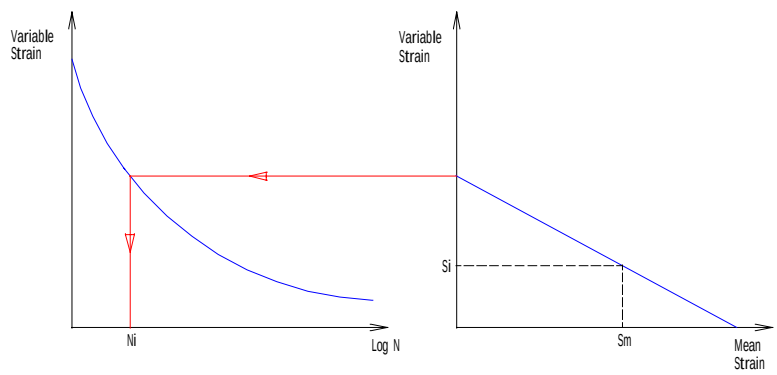
k_1: axial stiffness / k_2: torsional stiffness

Fig. 13 General view of 3D beam element and its stiffness matrix.

Based on stacking sequence information, the wing is discretesed to generate a model which contains information about wing aerodynamics and stiffness. The CTL is used to obtain flexural modulus E_y, E_z, torsional modulus G and other parameters at every section of the wing. The stiffness matrix of each element is constructed using the different modulus as it is shown in the latter figure. Finally all individual matrix are assembled in the general stiffness matrix of the model. Additionally, information about wing aerodynamics will be considered at every node of the wing. Basically this information is : main section chord, airfoil type, airfoil aerodynamic polar and equivalent wing section surface. Furthermore, the general mass matrix and dumped matrix are obtained to solve the well known problem of structural dynamics :

$$[M]\times[x'']+[C]\times[x']+[K]\times[x]=F(t),$$

where F(t) are the aerodynamic loads which depend on the relationships between the velocities of the sailplane, the structure and the gust. The problem shown is a non linear one that is solved using a Runge-Kutta algorithm with a variable time step. During calculations at every time the aerodynamic loads are obtained based on previous time step.
Once the problem is solved a whole set of time series for the actions at every section of the wing will be obtained. These time series can be used as a basis to perform a fatigue approach for the sailplane wing during its operational life.
A cycle counting for a period of time can be obtained from an analysis of the time series based on peak-counting and its inversion. It must keep in mind that this cycle counting correspond to one of the sailplane velocities. Hence, by performing these calculations for every sailplane characteristic velocity is possible to obtain the number of cycles associated at each one of them. Finally taking into account the different probabilities of the sailplane velocities, during a characteristic period of time, the whole number of cycles are calculated for the wing operation during this time.
The next step in fatigue approach was done with an analysis of stresses and strain levels in the wing sections for every variable action. At this stage, have to be consider the effects of mean loads in the stresses and strain levels to obtain pure equivalent alternating loads which will be used in a damage analysis later. This is done using the Modified Goodman Rule which express the relation between a state of mean and variable load and a pure alternating load. The rule is shown in fig. 14.



$$N_i = f(S_a) \qquad \frac{S_i}{S_a}+\frac{S_m}{S_{ut}}=1$$

Fig. 14 Modified Goodman Rule.

Now, for every cycle the equivalent pure alternating load is obtained. Based on Carbon Fiber data for the relation between stress or strain level and the number of cycles to failure we can obtain the maximum number of cycles for every stress or strain level as it is shown in the latter figure. These data are used with the Miner rule for equivalent damage to analyse the damage level corresponding at every action. Finally, the number of total time periods available is obtained from the amount of damage produced during the characteristic time period of sailplane operation.

CONCLUSIONS

In the present paper, it has been presented the integration of different commercial software in order to calculate a sailplane composite wing during the aerodynamic and structural design. An optimisation for the stacking sequences based on load information was carried out, taking into account the different envelopes of sailplane maneuvers. Non linear analysis of the material was also introduced to study buckling and non progressive failure of the wing structure. In addition wing dynamics and its fatigue life were analysed trough finite element calculations with aerodynamic non linear behaviour. The use of Von Karman models for atmospheric turbulence and peak counting techniques, permits fatigue analysis of the wing structure under random excitation loads.

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