QUICK ANALYSIS METHODS FOR COMPOSITE STRUCTURES

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ABSTRACT

The quick analysis methods applied for the preliminary analyses of the structure of the UAV MILANO are presented in this paper. MILANO is an UAV with the structure mainly produced with composite materials: laminates and sandwich panels. A semi-automated tool able to analyze different parts of the entire structure with the only need of a quite simple input (geometrical data like stringers positions and structural data like layups, thicknesses…) has been developed as the result of the challenge of achieving a methodology for sizing the structure with quick analysis methods. The modifications and adaptations of classical theories for metallic aircrafts adapted to composite materials used for this purpose and the validations performed are also described.

KEYWORDS

Composite structures idealization, modulus-weighted properties, preliminary analyses, composite aircraft structures, quick analyses methods, torsion box and stiffened panels.

INTRODUCTION

The aviation industry is increasingly concentrating on the development of Remotely Piloted Vehicles (RPVs), Unmanned Aerial Vehicles (UAVs) and Unmanned Combat Aerial Vehicles (UCAVs). These UAVs will replace the conventional aircraft in several roles as reconnaissance, scientific research or target practice and even perform novel assignments as monitoring of hostile battlefield environments and difficult access areas without endangering human life. This paper is focussed in the quick analysis methods for composite structures applied to MILANO, an UAV developed at INTA. MILANO, also referred as MILANO System is a reconnaissance, surveillance and target acquisition UAV, able to perform missions at high altitude and long periods of time.

The main feature of the MILANO while attending to its structure is the material it is made of. It will be mainly manufactured with composite material (epoxi-IM7 cabon fibers) and in certain areas with sandwich panels (composite materials in upper and lower skins and foam core) and metal.

Quick analysis methods were required to accomplish the necessity of performing a preliminary analysis of the composite structure in a quite reduced period of time. These methods have been traditionally widely applied to aircrafts analyses and designs, but mostly of them are developed for metal structures. Many hypotheses assumed in metal structures are not valid for composite material structures and these methods must be revised and adapted.

The methods for these analyses are not as homogeneous as they are for metal structures and the information and bibliography for preliminary designs is not as large as in metallic material. More than methods to design, in the bibliography there are many experimental advices [1].

In metallic materials the failure criteria are quite simple (von-Misses stresses for instance), but in composite materials due to the anisotropy of the material the criteria are more complex (there are different approaches which are optimum in different cases: Tsai-Wu, Hill, maximum strain…). The admissible values of the laminates are usually difficult to obtain and it is also hard to find a widely and unique used criterion for the different aircraft elements.

The quick analyses methods for attachments and joints are the usual simplified methods of aeronautical structures [2, 3, 4, 5, 6, 7, 8]. In case of torsion boxes, the analyses are based in the idealization technique widely used for preliminary analyses of metal aircrafts. For stress calculation in composite materials the hypothesis applied is the linearity of deformations and the idealization technique has been modified in order to apply it with the adequate modulus-weighted properties.

The obtained strains of the elements are transformed to stress values and compared with admissible material values. The ten per cent rule [9] has been applied to obtain the admissible values of maximum stress and strain. This rule is valid for axial and shear loads in the linear behaviour zone of the structure, however, while taking into consideration buckling failure mode, new hypotheses and techniques have been done. A description of the chosen bending stiffness for composite materials and its application to simplified buckling equations for the admissible values calculation are presented.

Later, simplified analyses are validated with FEA which simulate the load state for singular load cases. Detailed models of critical areas of the structure have been developed for this purpose. A test campaign is also programmed for validation and tuning of the different techniques. Some element level specimens already produced for these tests are also presented.

ANALYSIS METHODS

The structure has been divided in two groups in order to perform the analyses:

- Laminates and sandwich structure
- Attachments and Joints

Laminates and sandwich components

The aircraft structure is entirely produced with composite material with the exception of the landing gear, joints and attachments.

The idealization of stiffened shell structures theory for metal torsion boxes has been adapted in order to take into account composites particularities. The usual applied hypotheses are: skins only support in-plane loads, panel shear stress is uniform, skin out of plane stresses are supported with longitudinals, longitudinals area is composed of spar or stringer and equivalent skin areas, longitudinals only support axial stresses and ribs and frames are considered rigid in-plane and without rigidity out of plane

The semimonocoque construction is similar to those of metal aircrafts with the exception of the sandwich panels where stiffeners are replaced with foam filling. Thus, the usual discretization used in the stiffeners position is poorer in the case of sandwich panels and the stresses distribution less detailed. In any case it has been considerer enough for the analyses detail level required and with the conservative estimation of stresses.

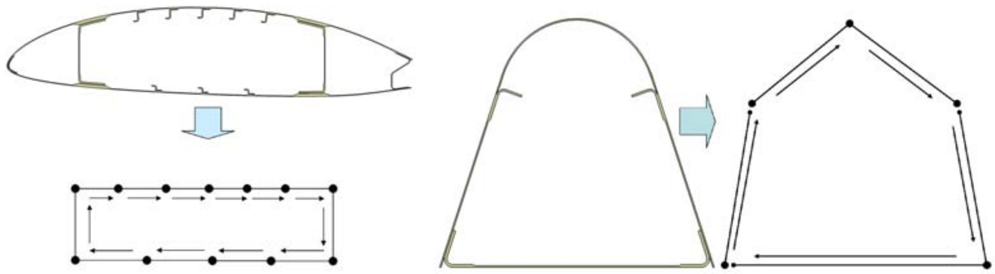


Figure 1: Structure idealization of the wing (left) and fuselage (right)

The main difference in the stresses calculation between composite materials and metals is the unhomogeneity of the Young modulus in the case of composites. Due to this fact, stresses neutral axis is not acceptable and the **strains neutral axis** must be calculated. The stresses are obtained with the deformations of the section and the elastic module of different structural elements (panels or longitudinals). The elastic module has been calculated using the ten per cent rule [9]

The strain neutral axis is dependant on the modulus weighted section properties, and instead of the usual effective or ideal areas, the effective/ideal EA and EI products are computed. The following expression is used for the loop of the strain neutral axis in the wing cross-section (in the XZ plane):

$$\varepsilon_y = -\frac{M'_x}{EI_x} \times z_p + \frac{M'_z}{EI_z} \times x_p + \frac{F_y}{EA_{total}}$$

M' are the effective moment components which differ from the natural components if the axes are not principal axes. A_{total} is the real compressive-tensile area of the cross-section.

In the scope of a preliminary analysis, maximum strain failure criterion has been selected. The ten per cent rule [9] has also been selected for the maximum allowable strain estimations.

Bending failure has also been considered in these analyses and therefore a simplification of the methods available in [1] and [10] has been applied. Thanks to the short range of variation of the laminate parameter $\frac{D_{12} + 2D_{33}}{\sqrt{D_{11} \times D_{22}}}$, this simplification is possible. This parameter varies, in the laminates of the MILANO air vehicle, between 1 and 3, which entails to the capability to determine constant values of k_i in the buckling equation:

$(\sigma_i)_{cr} = \frac{\pi^2 \cdot k_i}{t \cdot b^2} \cdot D_b$, where Db is the bending stiffness, and i represents the load type: compression, shear or in plane bending [11].

The bending stiffness is computed with the formula $D_b = \sqrt{D_{11} \times D_{22}} + D_{12} + 2D_{33}$, being D_{ii} components of the flexural stiffness matrix [10, 11].

The contribution to the bending from the different three load types follows a Shanley relation: $\left(\frac{\sigma}{\sigma_{cr}}\right)^c + \left(\frac{\tau}{\tau_{cr}}\right)^s + \left(\frac{\sigma_b}{\sigma_{bcr}}\right)^b = 1$ where

-*c*, *s*, and *b* are coefficients which must be empirically obtained,

- σ is the mean compressive stress in the panel,

- τ is the mean shear stress in the panel,

- σ_b is the mean in-plane bending stress in the panel.

The following values have been used considering a plate behavior of the composite plates similar to the metallic plates. For a better correlation many test should be performed to determine their values for the MILANO panels' layups, but this is out of the scope of preliminary analyses.

$$\left(\frac{\sigma}{\sigma_{cr}}\right) + \left(\frac{\tau}{\tau_{cr}}\right)^2 + \left(\frac{\sigma_b}{\sigma_{bcr}}\right)^{1.85} = 1$$

This equation is applied to determine the two loads more influential in order to simplify the analysis (3 loads require an iterative method to solve the post-buckling re-distribution of stresses while 2 loads have an analytical solution for the stresses re-distribution). When considering the two main loads, the following equations apply:

Under compression and shear loads: $\left(\frac{\sigma}{\sigma_{cr}}\right) + \left(\frac{\tau}{\tau_{cr}}\right)^2 = 1$

Under shear and in-plane bending: $\left(\frac{\tau}{\tau_{cr}}\right)^2 + \left(\frac{\sigma_b}{\sigma_{bcr}}\right)^2 = 1$

Under compression and in-plane bending $\left(\frac{\sigma}{\sigma_{cr}}\right) + \left(\frac{\sigma_b}{\sigma_{bcr}}\right)^{1.75} = 1$

These bending indexes are also applied to obtain the effective width. This is a typical method which simulates the effect of buckling in plates with a fictitious non-buckled panel with an adjusted width. This effective width represents the portion of the post-buckled panel which still supports load.

The effective width method requires an iterative calculation in order to obtain an accurate approximation of the post-buckled structure.

The upper methods are used for both fuselage and wing/fins type structures, but different hypotheses must be taken for the open crossed-section fuselage and the closed crossed-section wing when obtaining the stress distribution of torsion moments.

While idealized theory is optimal for crossed-section torsion boxes, in case of fuselage the acquisition of shear stresses needed to be adjusted with semi-empirical data for an adequate simulation of the torsion moment effect in panels, which is not possible with the idealized properties of the section.

The value of an average GJ has been calculated with FEA for the open crossed-section fuselage. A mean value was found to be accurate enough for the simulation purposes, with no recalculation required when a modification in design is implemented.

In the case of sandwich panels (composite materials in upper and lower skins and foam core) the additional hypotheses used are:

- The foam does not absorb axial stresses.
- The foam maintains is uncompressible in perpendicular direction to the panel.

These hypotheses allow an accurate allowable bending stress calculation of the panels while not increasing the complexity of this calculation.

Attachments and Joints

For the metal attachments and joints usual methodologies have been applied obtaining usual failure modes safety margins: pure tension, shear tear out, bearing.

Some critical joints have required to be studied individually and FEA have been done due to its complicated geometry as in case of joints in wings. MILANO UAV is intended for quick assembly-disassembly and must fulfil transport requirements which imply removable semiwings, removable fins and a fuselage capable of being parted. The results of the models are integrated in the design/analysis process.

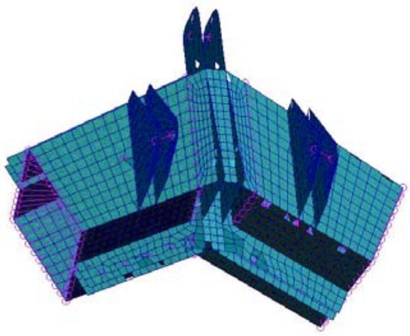


Figure 1: Fins - Fuselage finite elements model joint

Allowable stresses for rivets, bushing and bolts bores in composite material plates are still pending. Several tests are going to be carried out in order to calculate them.

ANALYSIS PROGRAM

The described analysis methods have been gathered in a program coded in MATLAB v7 which manages the analyses.

The input parameters, some of them semi-empirical which must be tuned upon design characteristics, are introduced from a text file. The mentioned parameters are:

-Safety factors for limit load and ultimate load

-Empirical-values which correlate allowable buckling stresses of a load type in panels with the bending stiffness of the panels.

-The GJ value of the opened-section of the fuselage obtained with a FEA.

-Empirical data for the stiffeners bending calculation. If the stiffeners size varies in the geometrical input data, the program adapts the characteristic bending data dependant of the stiffener area following a homethecy.

-Allowable buckling stresses factors in panels due to the curvature

-Allowable buckling stresses factors in panels due to the filling of foam in the fins

-Allowable errors for iterative processes: neutral axis calculation, effective panel width calculation.

-Design parameters related with the effective section area for axial stresses and characteristic data for the computation of the buckling allowable stresses.

The design input data (geometrical data like stringers positions and structural data like layups, thicknesses) is introduced in spreadsheets.

The internal loads are obtained from the mechanical and structural loads using a simplified finite elements model analysis. In the model there have been assumed the next hypotheses and simplifications: infinitely rigid engine, structural mass is linked with interpolation constrain elements (RBE3), non structural mass is taken into account with concentrated mass elements (CONM2), joints are simulated with spring elements, wing and fins lift is elliptically distributed along the ¼ depth points, wing and fins drag is proportionally distributed along different sections, wing distortion and aerodynamic moment is also included. There are nodes in each selected section. Furthermore, there are mass elements between sections which have associated the mass and inertia of the structure between those sections. Those values are imported from CAD design. Consequently the FEM model behaves analogously to the real air vehicle. Another MATLAB program manages the output reports of internal load distribution and joint loads from the FEA in order to be used by the mentioned analysis program.

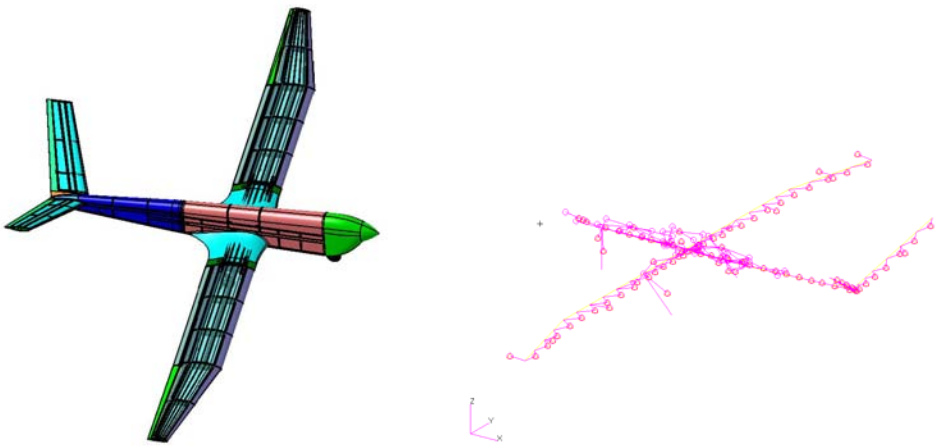


Figure 2: MILANO structure CAD and FEM

The program incorporates:

-The Db “bending stiffness” calculation as a function of the layup of the composite panels, and in the case of sandwich panels as a function of the layup and the foam thickness.

- The buckling stresses following the adapted metal semi-empirical theory

- The allowable compression, tension and shear failure stresses following the ten per cent rule.

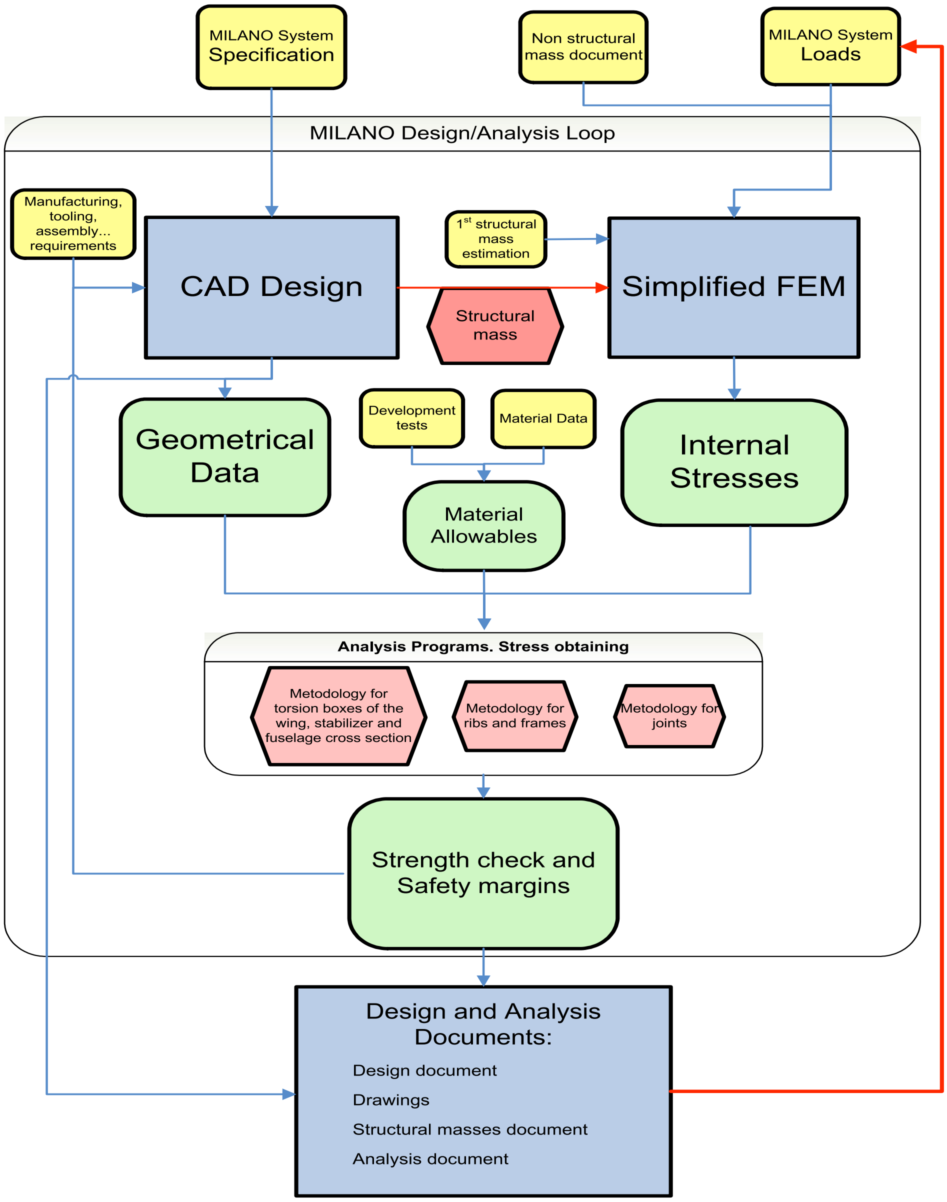
-The buckling stiffeners stresses following the Euler theory.

The safety margins for the limit load and the ultimate load cases are calculated and finally the program generates a summary spreadsheet for every element of all sections of the UAV with the allowable stresses and the most unfavourable safety margin and corresponding load case for each failure mode.

All the safety margins for all the considered failure modes in every analyzed section and for every load case are also stored in spreadsheets. These files contain all the analyses bulk data.

In the figure can be seen the design/analysis loop followed.

Program diagram

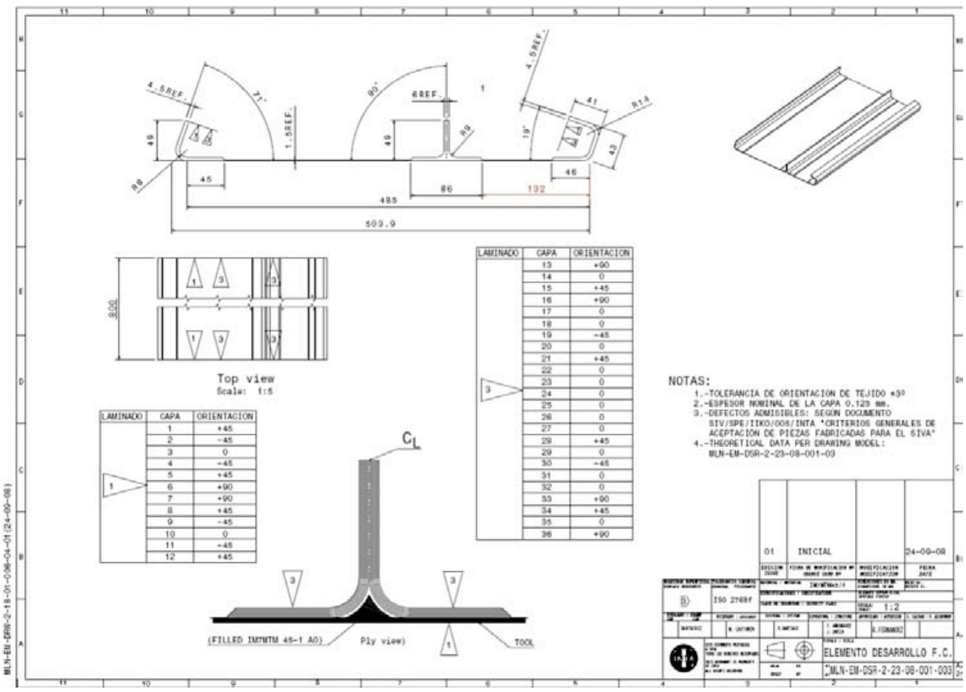


ELEMENT LEVEL SPECIMEN

The specimen with the following plan has been already manufactured and the tests are currently in production phase.

These tests are intended for validation purposes, and additionally for tunning the empirical and semi-empirical parameters which first estimation is based in metal structures, specifically for the constant empirical-values which correlate allowable buckling stresses of a load type in panels with the bending stiffness of the panels.

Test element plan



The specimen has been previously subjected to non-destructive evaluation.

Compressive load and shear load tests will be carried out reaching post-buckling regimen of the plates. Information about strains in different locations of the specimen, buckling initiation of both the plates and the stiffeners, and displacement versus applied load will be obtained.

The specimen will also be subjected to failure compressive load

CONCLUSIONS

In this paper, it has been proposed a quick-analysis methodology for an aeronautical composite structure that allows its integration in the mechanical development first phases, greatly improving the efficiency and time of the process.

As today’s engineering projects demand rapid analysis, advanced development capabilities and high level of prediction MILANO project mechanical development combines state-of-the-technology tools in order to optimize and validate MILANO prototypes.

Moreover, the theory and hypothesis used allow structurally analyzing an air vehicle structure mainly manufactured in pre-preg composite material taking into account different layups and thicknesses, and the use of sandwich structures.

The mechanical development has been supported with commercial software packages. MSC/NASTRAN FEM model and MATLAB programs allow estimating all internal loads and sizing each structural element and storing data in EXCEL spreadsheets. As a result, an initial prototype is being manufactured. The prototype presumably accomplishes all requirements.

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