# Design, realisation and testing of an advanced composite unmanned aircraft

A. F. Accardo, F. Ricci, P. Basso

*Dipartimento di Progettazione Aeronautica – University of Naples FEDERICO II*

**SUMMARY:** The large experience and confidence with the design of metallic aircraft structures that had been maturing in the Dipartimento di Progettazione Aeronautica (DPA) has been utilised for research and educational purposes for many years. The progress in advanced composites technology and the increasing use of such materials in the aerospace industry has leaded to introduce composite material concepts in the Aircraft Structure degree courses. However, due to the gap between the structural engineer and the material technologist, there is not enough confidence with composite structures design.

In order to bridge this gap some activities have been carried out in collaboration with experts of the technology and manufacturing area. This collaboration has been finalised to the design and realisation of a small composite unmanned aircraft.

In the paper are presented the activities concerning the design, realisation and testing of the unmanned aerial aircraft (UAV), pointing out the aspects related to composite material utilisation. Design criteria, based on the reduction of the numbers of structural components and consequently of manufacturing tools, are discussed.

Materials and curing cycle have been selected depending on strength requirements, commercial availability and process cost. Structural calculation have been carried out on the basis of traditional methods and assessed with finite element models.

**KEYWORDS:** Composite materials, aircraft structure, design and testing

## INTRODUCTION

Within the DPA, a research group has been working on the design and realisation of an advanced composite unmanned aircraft addressed to civilian application The project is under the financial support of the Italian National Research Council (CNR); and it is co-ordinated by the DPA and involves also others University Departments, a research centre and an industry specialised in composite manufacturing[1].

Starting from the long experience in the design of aeronautical structures, the main goal of the project has been the improvement of skill in composite airframe design and the support to academic activities in the field of experimental aerodynamics, flight mechanics, composite structure design, stress analysis and testing [1],[2].

The aircraft architecture is traditional: high rectangular wing, two stroke single engine driving a two blade propeller, tricycle landing gear with cantilever legs mounted on fuselage.

All structural solutions are addressed to reduce the structural weight and the number of parts, according with manufacturing problems and costs. In the paper the general design of the aircraft will be described as it concerns composites utilisation and manufacturing aspects. Furthermore wing static test will be described and discussed the comparison between experimental and numerical results.

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[1] Dipartimento di Ingegneria dei Materiali e Produzione - University of Naples FEDERICO II
Dipartimento di Ingegneria Aerospaziale - Second University of Naples
Istituto per la Tecnologia dei Materiali Compositi – National Research Council
Lavorazione Materiali Compositi S.p.a.

## PRIMARY DESIGN AND DESCRIPTION OF COMPITES PARTS

As the full scale composite vehicle will be tested in the main wind tunnel facility of DPA for both aerodynamic and structural investigations, the dimensions of the airplane are limited by the wind tunnel test section. This limitation allow to transport it as far as the starting mission location by a conventional road vehicle.
The starting point of the structural design has been the choice of the material and appropriate manufacturing techniques. A preliminary evaluation of mechanical properties, costs and commercial availability has leaded to utilise a graphite epoxy prepreg to be cured in autoclave at a temperature of 125 °C and at a pressure of 5 bar [1].
Simple structural solutions have been considered as an important requirement in the structure design, in order to reduce the tooling costs and to simplify the manufacturing techniques.
As result, the number of the structural parts has been reduced, riveted junctions avoided, bonding and co-curing techniques widely utilised.
The wing has been realised through a two spar box beam with four ribs, covered by a skin closed on the trailing edge, as shown in Fig.1.

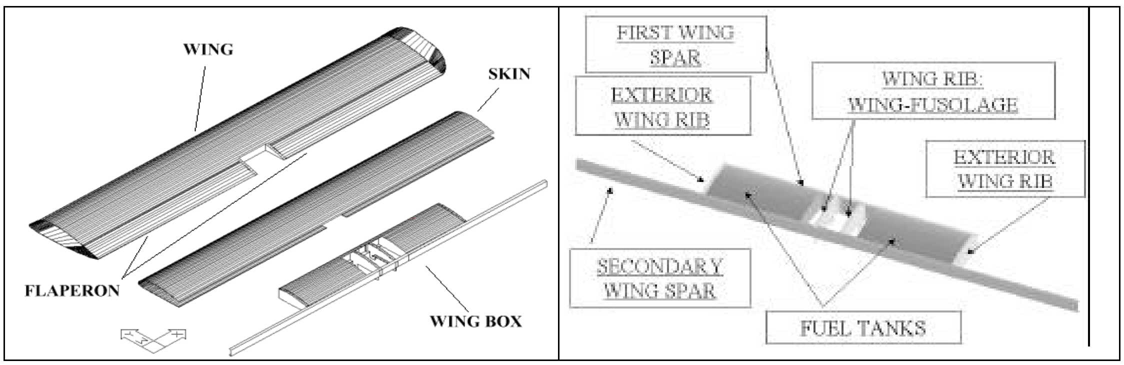


Fig. 1 – Wing structural solution

With exception of fabric ribs, all the wing elements are realised with unidirectional plies.
Similar criteria have been used for the fuselage that consists of two symmetrical semimonocoque assembled together with five bulkheads. The fuselage skin has been realised with unidirectional prepreg whereas bulkheads in fabric.

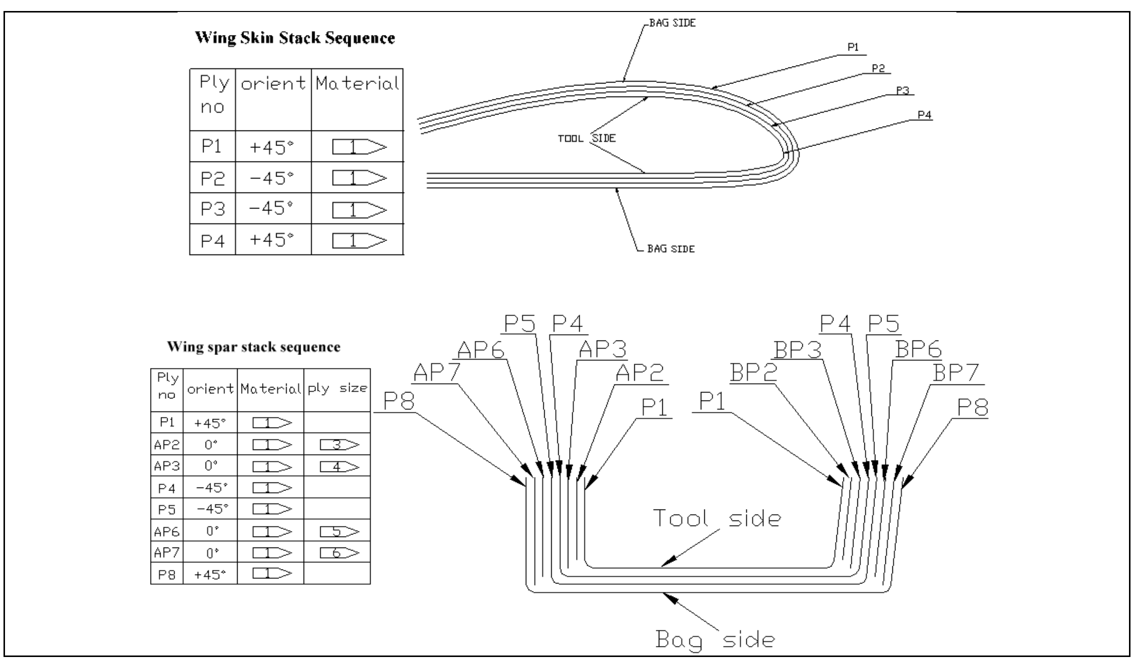


Fig. 2 – Wing skin and wing spar stack sequence

## CONCEPTUAL AND FEM ANALYSIS

In spite of reduced dimension, a rigorous procedure has driven the structural design of each part of the UAV. From the material mechanical properties (relative to a cured unidirectional or fabric lamina) [3] a first conceptual analysis [4], [5], [6] to evaluate the stress, strain and displacement distributions and their peak values has been performed for wing [7], fuselage [8], vertical and horizontal tail elements, according to the algorithm showed in Fig.3. In particular, for wing and fuselage skins, also shear and compression critical loads were determined via theoretical analysis [9], [10], [11] for various edge constraints conditions.

1. Input: Single ply mechanical properties E1, E2, G12

2. Analytical detetermination of laminate mechanical properties Ex, Ey, Gxy using the classical lamination theory

3. Analytical determination of the structure response (mean values of stresses) under the critical flight loads [1] applying the classical theory of orthotropic shell beams (modulus weighted section properties)

4. Analytical determination of the stresses for each one of the plies in the laminate: single ply stress calculation via classical lamination theory

5. Comparison between calculated single ply stresses and carpet plots [3] values for lamina ultimate allowable stress (Maximum strain failure criteria)

Fig.3 – Used procedure in the theoretical dimensioning of laminates

As it regards the material ultimate allowable stress, experimental investigations have been carrying out, according ASTM and BSS standard specifications, in order to substantiate the theoretical evaluations [12], [13]. The theoretical analysis conducted, in facts, has been utilised as a preliminary laminates dimensioning tool and as a smart key in FEM modelling and analysis results understanding. As a result, good agreement between theoretical and numerical analysis has been achieved, see Fig. 4

**Wing skin Laminate Medium Stresses comparison**

| | | FEM | Theoretical | |
|---|---|---|---|---|
| $\sigma_{max}$ | daN/mm^2 | 2,52 | 1,73 | Tensile stress |
| | | 2,98 | 2,7 | compresion stress |
| $\tau_{max}$ | daN/mm^2 | 0,76 | 0,5 | |

**Fuselage skin Laminate Medium Stresses comparison**

| | | FEM | Theoretical | |
|---|---|---|---|---|
| $\sigma_{max}$ | daN/mm^2 | 1,56 | 1,4 | Tensile stress |
| | | 1,9 | 1,95 | compresion stress |

**Wing skin first compression critical eigenvalue**

| | | |
|---|---|---|
| $\lambda$ | 0,2 | Simply Supported |
| | 0,4 | Fixed |
| | 0,9 | SSFF |
| | 1,3 | FEM |

Fig. 4 – Some FEM and Theoretical results comparison

Taking advantage of FEM capabilities, all critical load conditions have been investigated. An accurate mapping of the interlaminar stress field has been achieved as showed in figures 5 and 6 for the wing spars, fuselage and bulkheads. Careful numerical and theoretical investigations are in progress for constraint elements, such as the wing-fuselage junctions shown in Fig. 7, for the wing flaperon coupling and for the attachment of the torsion tube of horizontal tail plane to the rear bulkhead [11], [14].

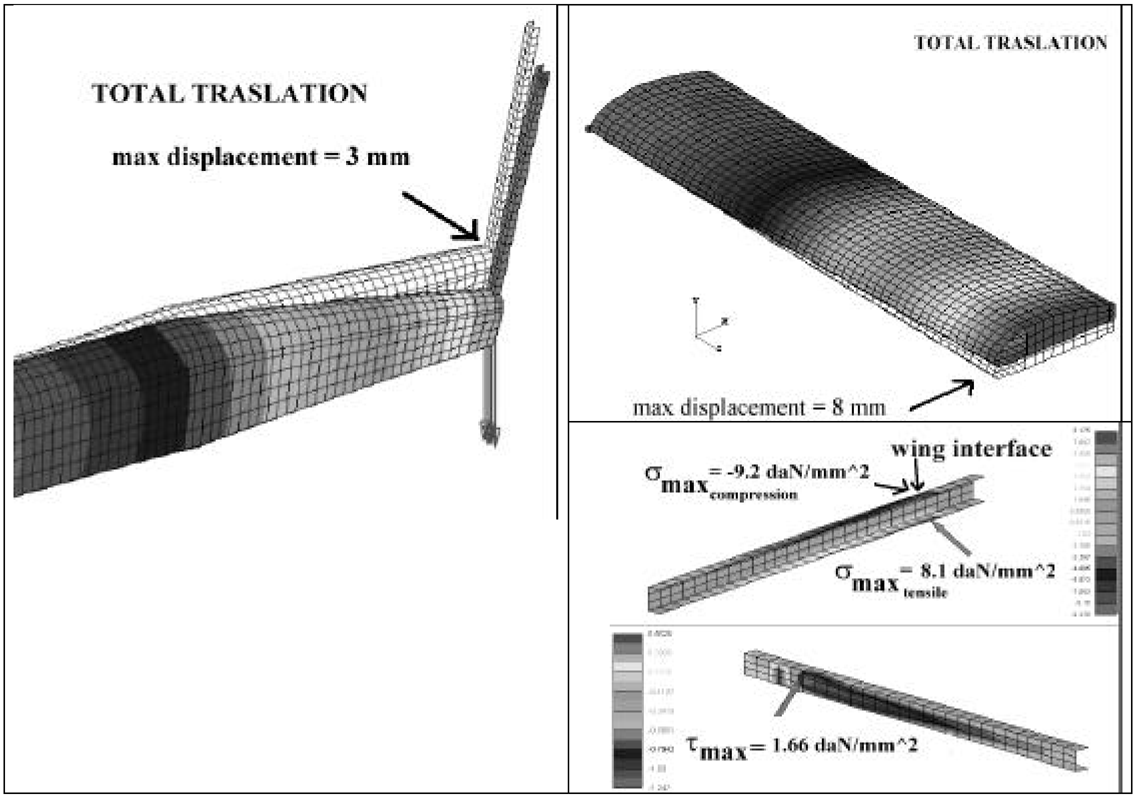


Fig. 5 – Finite element outputs for fuselage, wing assembly and stress values for wing spars

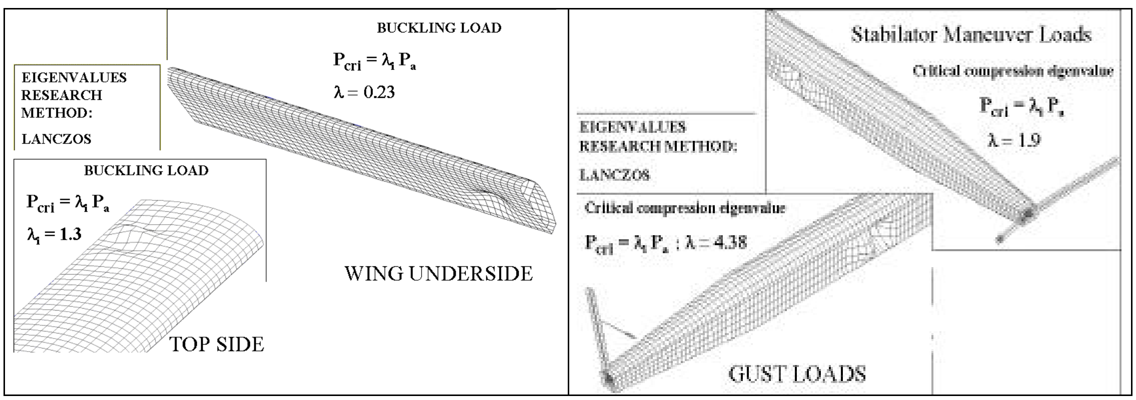


Fig. 6 - FEM Critical Compression loads analysis

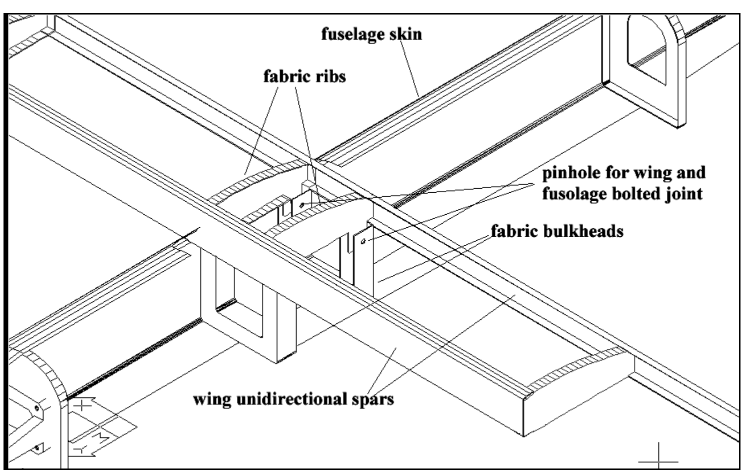


Fig. 7 – wing fuselage junction

## STRUCTURAL FINAL DESIGN AND MANIFACTURING

As mentioned above, the final structural configuration has been obtained according to the requirements of low manufacturing cost, easy assembly, fastening, inspection and tooling. Other main drivers have been the low roughness of the aerodynamic surfaces and the possibility to obtain relative small curvature radii as for wing, flaperons and tails leading edges, due to the small dimensions of such elements. A good aerodynamic roughness can be achieved only with female tools. On the other hand male tools are more capable in fitting desired curvature radii. Therefore male tools have been adopted for all of composites parts using a pressure cold composite plate for the roughness control, obtained with the same tool. This technique has been considered a valid solution for the small thickness of the skin (0.6 mm).

Some practical considerations have driven the choice of tape or fabric layers. The tape layer is characterised by an higher structural efficiency if compared with the fabric one, but it is not suitable for manufacturing problems in the case of tools with small radii of curvature, as for the wing ribs, fuselage bulkheads and control aerodynamic surfaces.

Fig. 8 – Wing Skin after curing cycle on male tool and wing box beam assembly

Fig. 9 – Fuselage skin tool and the unidirectional wing spars

Fig. 10- Fabric wing root, bulkhead tools and fabric bulkheads

## STATIC TEST SET UP

Static tests have been setting up. The more complex behaviour of composite structures compared with metallic ones requires more care in tests execution and results interpretation. At present a first evaluation of macroscopic phenomena has been carried out for the whole structure of the UAV in order to justify the main design criteria adopted. In Fig. 11 is showed the wing test set up: a rigid structure has been used to fix the wing root ribs and simulate the connection to the fuselage. The load has been applied using calibrated weights, and the actual load distribution along the span has been simulated with a leverage system. Bending and torsion limit load condition have been investigated, according to the Joint Aviation Requirements.

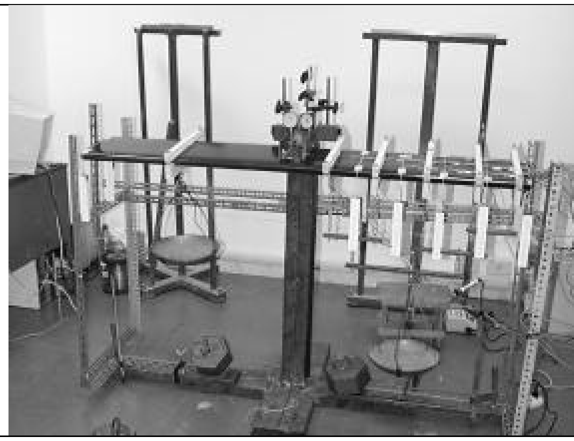

Fig.11 – Wing static test set up

The displacements have been evaluated with high precision mechanical displacement transducers near the constraint region and with decimal references along the wing span. On the skin, ribs and spars monoaxial and triaxial strain gages have been applied and signals acquired with an HBM data acquisition system.

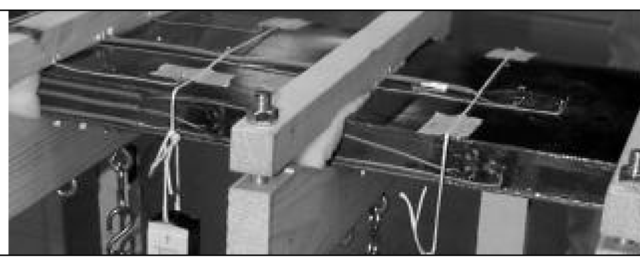

Fig. 12 – Wing skin and spar strain Gauges close up

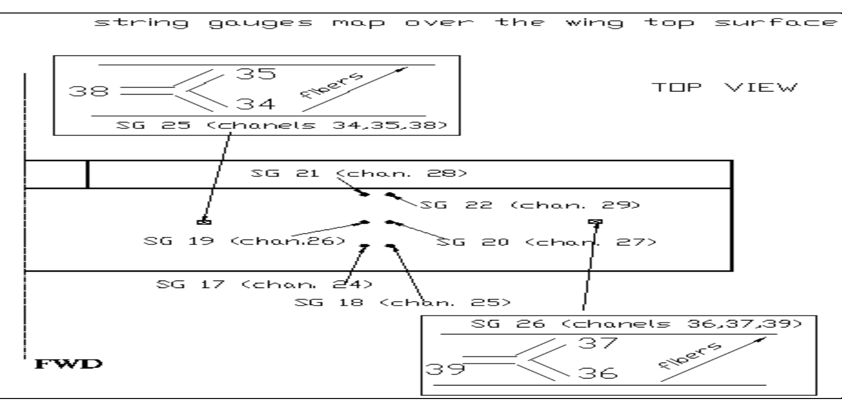


Fig. 13 – Strain gauges distribution scheme

As visible in the previous picture, a strain gauges concentration is located at the end of the main wing spar in order to deeper analyse the transition zone.

## STATIC TESTS RESULTS AND COMPARISON WITH THEORETICAL PREVISIONS

An interpretation of static test results is still in progress. Here we present for brevity just some of the several data acquired. Also considering the small value of displacements that this high strength material exhibit, a good agreement between numerical and experimental data is obtained for displacements, as shown in figures 14 and 15 relative to the flexural test up to limit load.

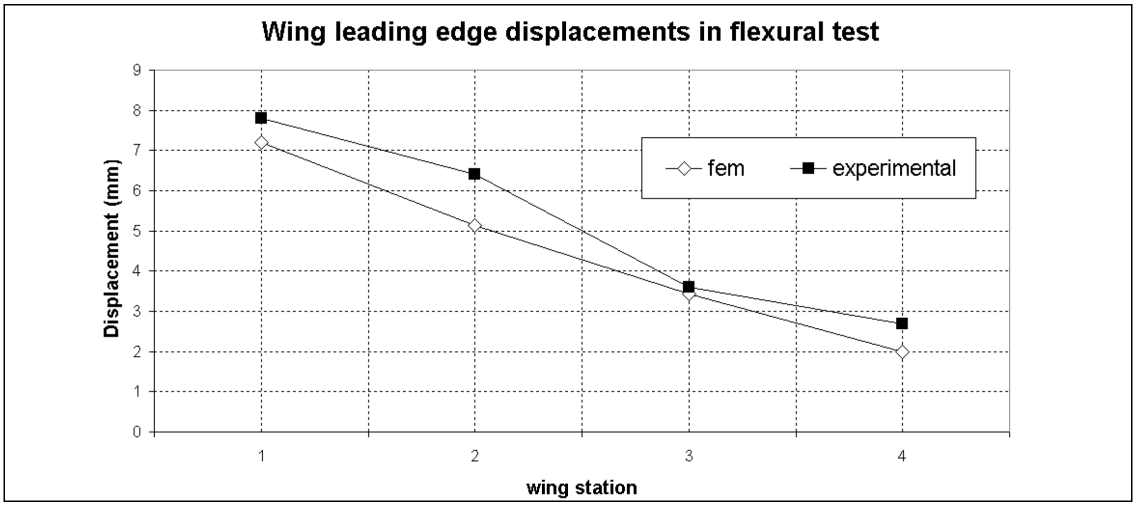

Fig.14 – Wing leading edge displacements derived in flexural test versus wing span

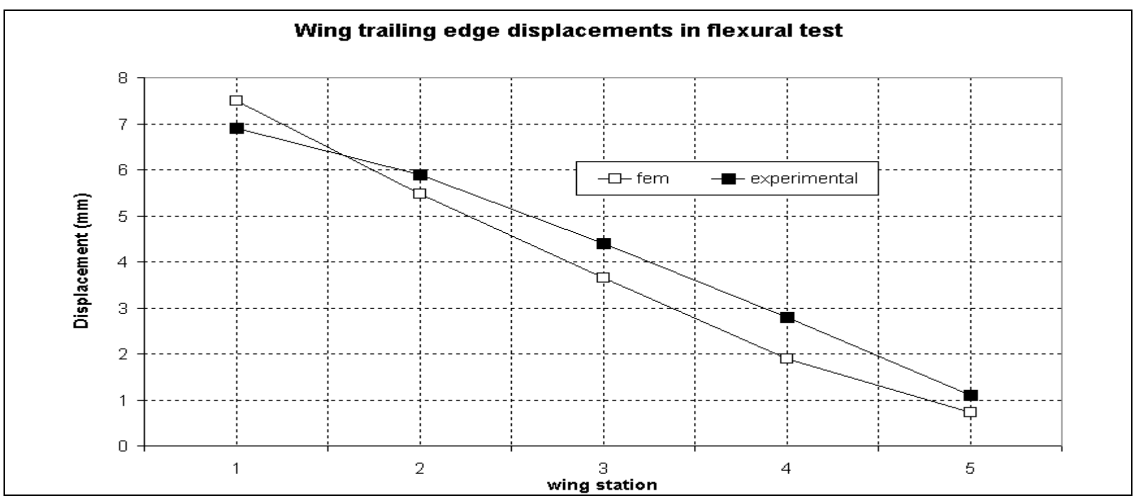

Fig. 15 – Wing trailing edge displacements derived in flexural test versus wing span

A detailed analysis of the large amount of acquired strains is still in progress and here omitted.

## CONCLUSIONS

In the paper has been presented the progress of research activities focused on the investigation of design techniques suitable for composite structures for aeronautical applications. Some design solutions regarding the assessment of the structure configuration, the reduction of the number of the parts and the lay up material choice have been pointed out. FEM analysis have been described together with some preliminary experimental results of the wing static tests.

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