THE CFRP SAILPLANE WING SEGMENTS MANUFACTURED BY FILAMENT PLACEMENT

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Abstract

The application of filament placement technology on sailplane wing is discussed in the paper. Two structure designs, based on high speed CNC milling foam core mandrels, are compared. Performed finite element and real experiments shows that used low cost foam mandrel has some limitations when it is used in wing structure. On the other side automation of sailplane wing manufacturing can offer reasonable benefits.

1 Introduction

The main examined idea of the presented research was to assess the possibilities of manufacturing of sailplane wing structure by filament winding placement. Such manufacturing process can be almost fully machine-operated what can minimize all negative aspects of manual work (e.g. hand lay-up process). These factors can increase the efficiency of wing production. And there is also potential to improve some of the specific mechanical characteristics of wing structure what should be evaluated.

Two different types of sailplane wing design are presented in the paper. The first one – compact sandwich structure was manufactured by filament placement onto positive foam core wing model (Fig. 1). The theoretical lay-out of the second structure manufactured by filament placement onto multi-web CFRP sandwich core is designed and described. A standard wing box (main spar, rear spar and sandwich skin) from sailplane HPH G304 S is used as a referential design.

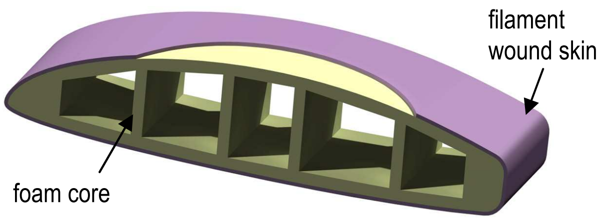

Fig. 1. Schematic visualization of investigated sailplane wing structure

The main goals of the experimental wound wing were to carry out design and parametric study of foam core mandrel and stress analyses of wound CFRP wing skin. In past decade, the cost of CNC manufacturing technology has been considerably reduced therefore its larger application in small aircraft structures is becoming economically efficient. Also many types of low cost polyurethane foams are available at the market. Promising progress of small aircraft production automation can be accomplished by combination of both those factors.

1.1 Experiment lay-out

The geometry of examined segments is adopted from the wing of glider HPH G304 S. Particularly; it is the section from root rib (c = 855 mm) with overall length 3800 mm (Fig. 2). The segment itself is only load-bearing part of wing (Fig. 3). It is the part from leading edge to former position of rear spar, which was situated slightly in front of the flaperon.

The transition between wound structure and flaperon should be in real application additionally bonded to it. And same approach could be used also for all others transitions in the area of trailing edge (e.g. A, Fig. 2.).

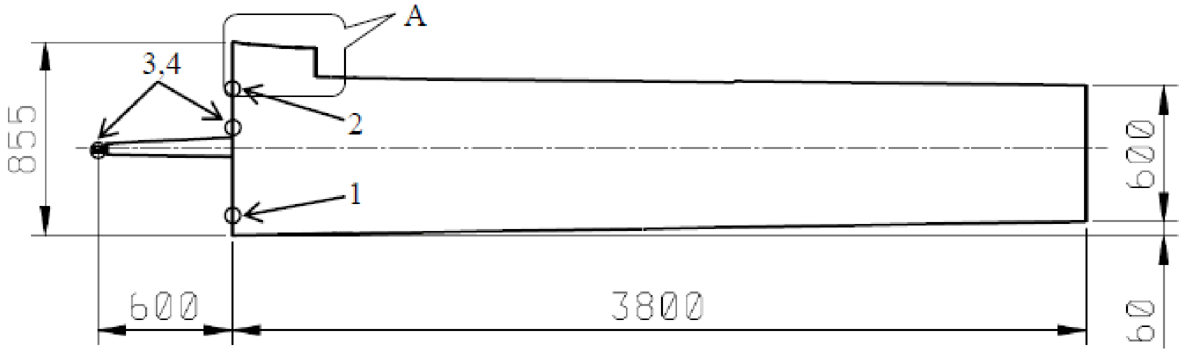

Fig. 2. Schema of original sailplane wing

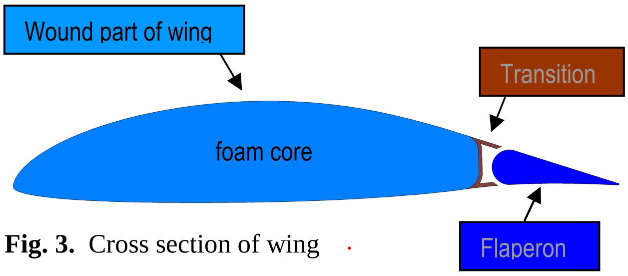

Fig. 3. Cross section of wing

2 Structure design

As the segment is primarily technological and structure design demonstrator some simplifications are used mainly in the area where forces are acting during testing. The following restrictions and assumptions were applied on the experimental filament wound wing structure:

1. The out of wing part of cantilever spar which serves as a wing hinges was removed. Wing root cross section of tested segment was extruded 750 mm in direction of removed outer spar. The new wing part was used for clamping of the prototype during a static test of experimental wing (Fig. 14).
2. A constant laminate thickness was used over wing span to simplified manufacturing process of wing segment prototype.
3. The constant thickness laminate stacking sequence was calculated analytically so that bending stiffness in root cross section of wound wing was equal to the wing bending stiffness in root area of the real HPH glider wing.
4. The wing load was reduced on single force acted on wing tip. The bending momentum at the root cross section of the wing was equal to momentum at HPH glider wing.

2.1 Materials models and sandwich modeling approach

Selection of foam materials was compromise between material characteristics, costs and weights. The “Foam 1” (Tab. 1) is light enough with acceptable price but with the low stiffness and strength material characteristic. On the other hand the “Foam 3” has higher material properties values but from economic point of view it is on a border of applicability in low cost aircrafts. The foam material characteristics were derived from compressive test (ASTM C365-03) and rail shear test (ASTM C273-00). The “perfect plastic” isotropic material model was used for FEM simulations.

	Foam 1	Foam 2	Foam 3
E [MPa]	24	48	63
μ [-]	0.22	0.22	0.22
G[MPa]	9.8	19.6	25
Yield stress [MPa]	0.85	1.6	2
Density [kg/m^3]	50	100	160

Tab. 1. Foam material characteristics

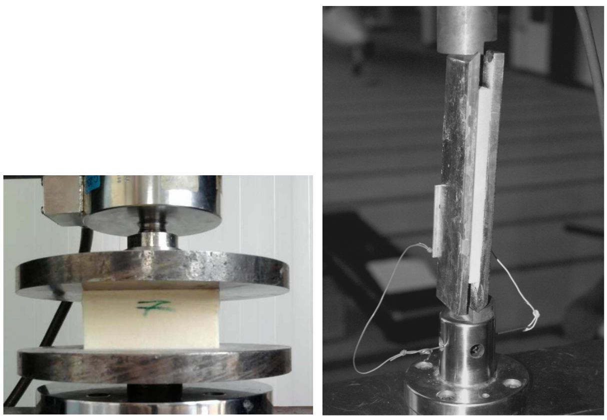

Fig. 4. Foam material testing

The beech plywood and fabric CFRP were also used in simulations. Their material characteristics correspond to required characteristic for hand – lay up laminate manufacturing process accepted LBA [1] for building of gliders.

	CFRP	Plywood
E11 [MPa]	39470	10000
E22 [MPa]	39470	
μ [-]	0.037	0.3
G12 [MPa]	1620	
G13 [MPa]	1620	
G13 [MPa]	1620	
Xt [MPa]	146	
Xc [MPa]	146	
Yt [MPa]	146	
Yc [MPa]	146	
S [MPa]	72	
ILSS [MPa]	58	
t [mm]	0.32	18

Tab. 2. Fabric CFRP and plywood materials properties

Simulation strategy of sandwich structure was tested and compared with four point bending laboratory test (Fig. 5) according ASTM C393-00. The MSC.Partan/Nastran with application nonlinear solver 106 was used for all simulations. Composites skin was created using two plies of fabric CFRP and material Airex 70.55 serves as a sandwich core. Orthotropic linear elastic material model was used for fabric CFRP and isotropic perfect plastic for foam core. Three nodes TRIA3 elements simulated composite material skin and volume elements CTETRA (Fig. 6) was used for sandwich core.

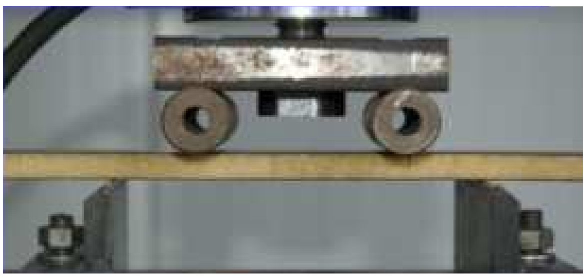

Fig. 5. Sandwich specimen bending test

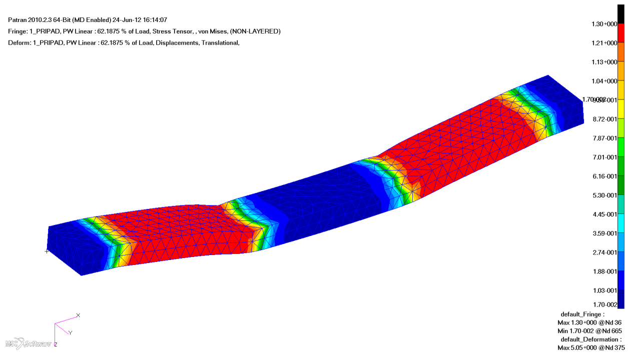

Fig. 6. FEM analyses of bending test of specimen

Figure 7 shows force displacement comparison curve between simulation and series of bending test experiments. The displacement was measured in the center of specimen. The slightly nonlinear responses of measured data, in area where linear slope of curve would be expected, were caused by inaccuracy in simulation of the contact between specimen and loading points. Highly nonlinear response at the maximum load was caused by reaching of yield stress point of sandwich core.

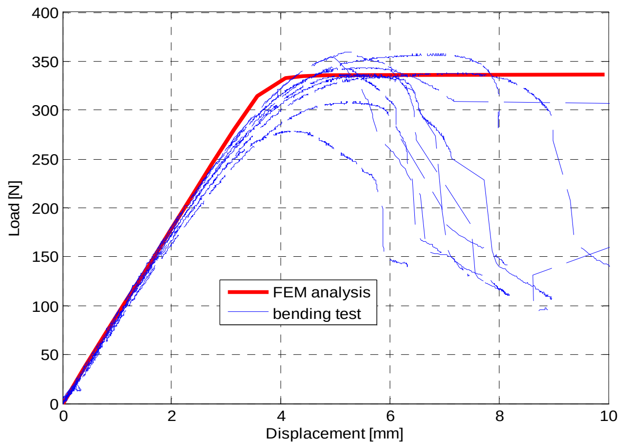

Fig. 7. Comparison of FEM analyses and bending test

2.2 Design of mandrel

The whole mandrel was divided by four ribs onto three parts (Fig. 8). The first two ribs (from left side) reinforced the structure in place of clamping and the last rib on the other side of the wing supported the structure in place of acting force. The third rib was placed into step change of wing profile height. Modeling strategy was adopted from four point bending test analyses of sandwich specimen. The wing was constrained directly using nodes connected skin and the first two ribs. MPC rigid element was used for application of loading force to wing tip rib. The TRIA3 elements with global size 35 mm modeled laminates and CTETRA elements simulated foam.

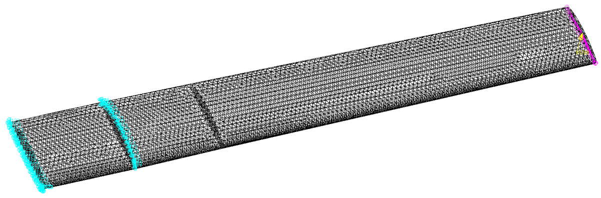

Fig. 8. Experimental wound wing FEM model

Two types of mandrels – compact sandwich and multi web were considered as applicable structure concepts for purpose of winding of low cost wing. The compact sandwich wing model was simplified. Internal holes along the span necessary for fixing the mandrel to winding machine were omitted in FE model. Internal surfaces of multi – web mandrel (Fig. 9) were covered by two layers ± 45° laminate made of manual placed CFRP fabric. Initial value of core thickness 15 mm was chosen for preliminary design. Higher thickness would have been ineffective from weight point of view in combination especially with “Foam 3”. Global stiffens characteristic and sandwich failure criteria were used for evaluation of preliminary wound wing analyses.

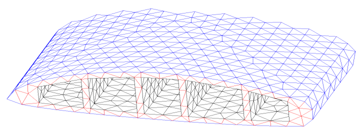

Fig. 9. Cross section detail of the multi-web structure model

The percentage of ultimate load on wing tip displacement curve for executed simulations with different foams is shown in figures 10 and 11. The global stiffness of wings was not strongly affected by type of foam. Increased global stiffness of multi web wing was caused by application of diagonal CFRP layers on the webs. Thus shear forces were carried by laminates on the web without help of the foam.

Perfect plastic foam material behavior had a strong effect on result global response. The compact sandwich wing simulation terminated calculations at the point when foam yield stress point was reached. Achieving of yield stress point in foam at multi web structure caused nonlinear bending stiffness of wing.

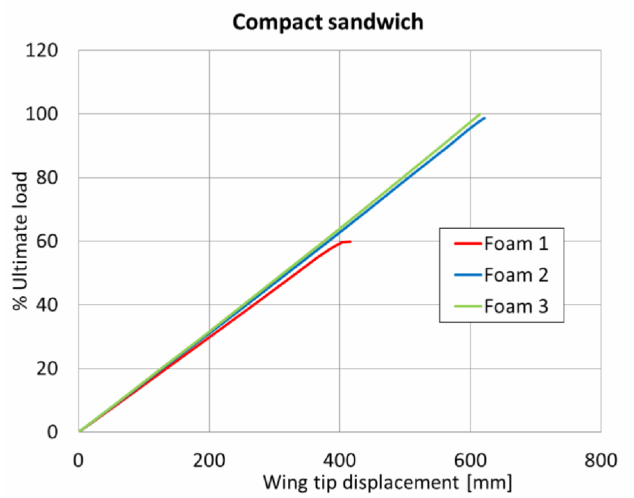

Fig. 10. Compact sandwich analyses results

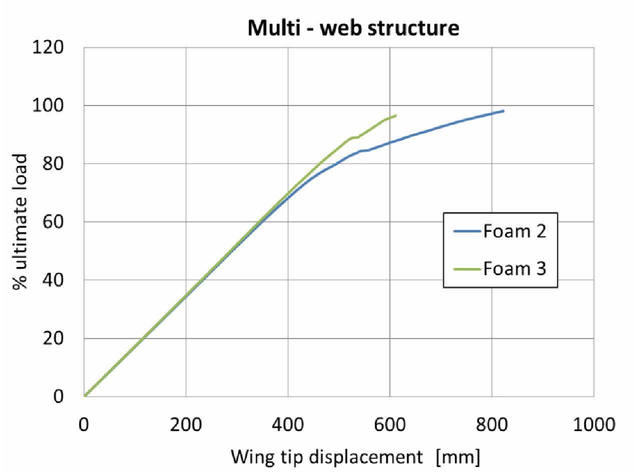

Fig. 11. Multi-web structure analyses results

Furthermore standard sandwich and failure criteria were used for analyses evaluation. Following criteria were considered: face wrinkling, shear crimping, shear core. Maximum stress criterion assessed carrying capacity of the solid laminate skin and also local skin buckling was taking into account. Short wave local buckling of wound skin is shown in figure 12.

Face sheet wrinkling is short local wave buckling pattern of the sandwich skin. It can buckle inwards or outwards, depending on the relative strengths of the core in compression and the adhesive in flat wise tension.

$$\sigma_{wr}=k(E_f\,G_c\,E_c)^{1/3} \qquad (1)$$

The criterion was calculated using equation (1) where E_c exhibits core compressive Young modulus in direction of sandwich thickness, G_c shear core modulus, and E_f face Young modulus in principal compressive stress direction. Wrinkling factor k is dependent on core thickness [2] and differs for both structure types 0.76 for compact sandwich and 0.63 for multi web structure.

The shear crimping is caused by low core shear modulus, or low adhesive shear strength. The generic relation (2) was adopted from literature [3].

$$N_{cr}=G_c\,t \qquad (2)$$

The value N_{cr} represents force per length unit and t is core thickness.

Shear core strength was evaluated directly from von Misess stress result on volume elements.

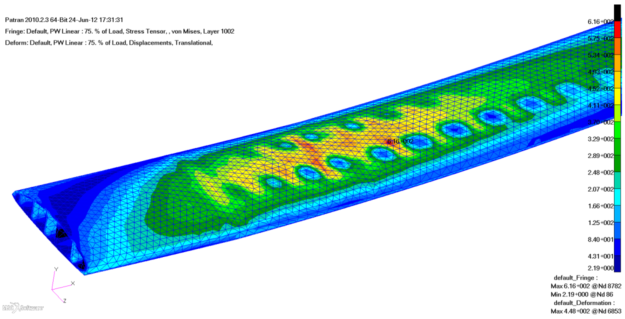

Fig. 12. Skin local buckling example on multi-web wing

The onset of sandwich failures and their values are displayed in table 3a and 3b.

Compact sandwich			
% ultimate load (critical value)			
	Foam 1	Foam 2	Foam 3
Wrinkling	50% (220)	76% (349)	91 (414)
Crimping	-	-	-
Core shear	60% (0.85)	98% (1.6)	-
Local skin buckling	57%	97%	-
Composite skin failure	58%	98.75%	-

Tab. 3a. Values of sandwich failures

Multi-web structure		
% ultimate load (critical value)		
	Foam 2	Foam 3
Wrinkling	49% (289)	58% (344)
Crimping	25% (294)	32% (375)
Core shear	62.5% (1.6)	75%(2)
Local skin buckling	55%	65%
Composite failure	72%	80%

Tab. 3b. Values of sandwich failures

The compact sandwich mandrel made from “Foam 3” was selected as a final design for manufacturing of wound wing segment prototype.

3 Manufacturing process

The tested segment is consisted of the milled foam core onto which the skin in the form of carbon filament is wound. The filament placement process is based on Compotech’s zero degree axial fiber laying technology which allows laying of fibers by filament winding at the small helix angles and also in zero degree direction. This process can offer:

- High bending stiffness and strength
- Predictable and controllable high fiber content
- Consistent and repeatable thin or thick walled structures

This is due to the optimization of the axial fiber placement, controllable fiber fraction and low porosity.

The first (axial) layer was wound onto the surface of the foam mandrel, spread by the adhesive to interconnect the foam and fibers firmly. The winding process progressed slower than expected. Therefore the first layer started to cure before the second cross-layer was wound. Thus the resulting surface became rougher and it had to be repaired additionally by spreading the resin over the surface and again over wound by a PP-tape. Finally, the segment was cured at temperature 100°C.

Fig. 13. Wing segment during winding

4 Experiment, results and discussion

The static bending experiment was carried out to find out bending stiffness. The layout of static bending test is shown in Fig. 14. The specimen was fixed in the area of root rib. The loading force was applied to the opposite side into the area of segment's tip rib. The deflection along the specimen span and strain in the selected areas was measured. Applied loading was proportionally derived from testing of standard sailplane wing. The main aim of the static test was to achieve desire global stiffness characteristics and to prove quality of wound skin.

After the test the compact sandwich FE model was revised according manufactured wound segment. Some slight changes were made in laminate thickness and in ratio between zero angle plies and cross plies. The new simulation was done and compared with static test.

Force and displacement dependencies measured during experiment and from FE simulation on wing tip are compared in figure 15. The difference in global stiffness characteristic (ratio between force and displacement) is 14.5%.

Fig. 14. Static test of compact wound segment

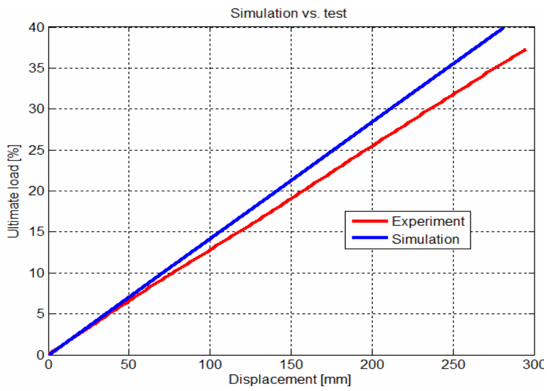

Fig. 15. Comparison analysis and experiment

5 Conclusions

Performed parametric study during mandrel design shows that low cost polyurethane foams are not stiff enough for applications in combination with high stiffness and strength carbon fiber laminate. The compact sandwich structure using lightweight foam is prone to sandwich short wave buckling failure and shear core failure. The sandwich core in compact sandwich configuration which would be stiff enough for carrying desired loading is not applicable because of increased weight.

The multi-web mandrel structure is promising from weight point of view but used low cost polyurethane foams are not sufficiently stiff. This structure is sensitive on sandwich crimping failure and furthermore brings increased cost.

Couple of problems occurred during winding of segment. Most of them were caused by introduction of filament winding process to wing style mandrel. Technological problems like premature start of curing can be solved easily which also improved quality of outer surface. Efficiency of manufacturing was also affected by application of new technology and it will be improved as soon as all technological parameters are set properly. On the other hand the precision of filament placement and automation of wing manufacturing are very promising for future applications.

Future design of wound wing could be based on multi –web structure concept with application of high quality sandwich core.

References

[1] Interessengemeinschaft Deutscher Akademischer Fliegergrupen, Dimensionierungsrichwerte für den Segel und Motorsegelflugzeugbau, März 1988

[2] Ley R.P., Lin W. Mbanefo U. *Facesheet Wrinkling in Sandwich Structure*, NASA/CR-1999-208994

[3] *ASM Handbook volume 21 – Composites*, ISBN 0-87170-703-9, 2001

[4] DoD Handbook *Composite Materials Handbook*, *MIL-HDBK-17*. Volume 1-3, 2002.

[5] Lopes C. S. Progressive failure analysis of tow-placed, variable-stiffness composite panels. *International Journal of Solids and Structures 44*, 2007, 8493-8516.

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