Manufacturing process, mechanical behavior and modeling of composites structures sandwich panel

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Abstract: *The complexity of sandwich structures is a challenge for aeronautics designers. Sandwich construction is widely used in both the aerospace and commercial industries because it is an extremely lightweight structural approach with high rigidity and strength/weight ratios. Although today's technology offers the possibility to combine a variety of materials for these structure solutions, in aviation only a few materials are accepted. This paper presents the technological process of making these sandwich structures, as well as a study of the characterization and testing of a sandwich structure to analyze the behavior from a mechanical point of view. The conclusions of the paper represent an experimental basis on which further research will be built.*

Key Words: *sandwich structure composite, mechanical test, modeling, manufacturing process*

1. INTRODUCTION

The percentage of composite parts used in the aviation industry grows steadily. The Airbus A 350 and Boing 787 have more than 50% percent composite materials in the structure [1]. The composite sandwich structure is a challenge for designers due to their complexity of manufacture and the advantages offered by these materials. According to Bruno Castanie et. al [2], the sandwiches split into two categories: symmetrical sandwiches used mainly for their resistance for buckling and their bending stiffness suitable for pressurized structures or aerodynamic loads and the asymmetrical sandwiches, less popular, but useful for light aircraft, moderately loaded structures of helicopters or drone type [2].

For the latter one, the core provides the buckling resistance of the skin and the "stabilizing skin" which is the second skin designed at the minimum allowed [2].

The manufacturing technique used for sandwich composite panel production is impregnation with vacuum bag pressing and autoclave curing [4].

The structural sandwich concept is defined as two thin and stiffness skins combined with a honeycomb, integrally bonding them together, obtaining a superior bending stiffness and low weight [5].

As a result of the core higher shear strength and compression stiffness, the sandwich panel can efficiently absorb tensile and compressive forces in the top layers.

The tensile and compressive forces in the top layers stem primarily from bending stress, meaning from a load that acts perpendicularly on one of the top layers of the sandwich panel [3, 6]. Commonly used materials for skins are metal and reinforced polymers, while the basic materials for the core are balsa wood, Nomex or aluminum honeycomb and some polymeric foams.

Material combinations are made according to the desired application and, of course, one or more benefits, such as low cost, high mechanical and thermal properties, soundproofing, fireproof, low smoke emissions, compliance, ease of processing and formation [5].

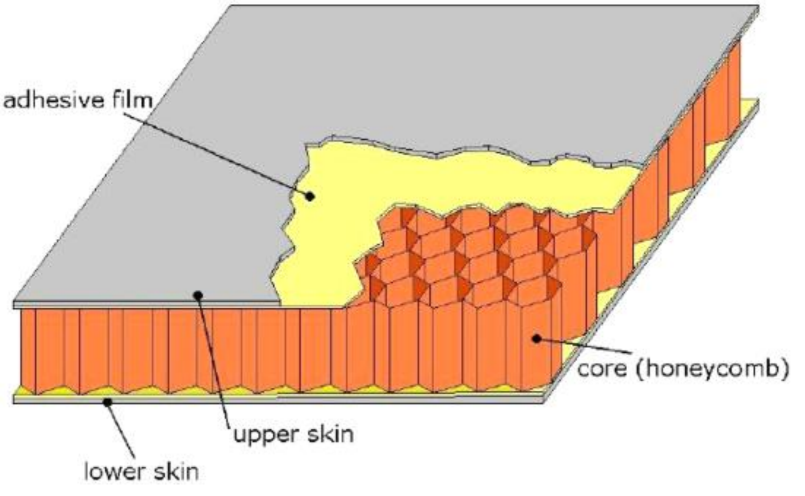

Fig. 1 – Sandwich design [5]

The pre-preg lay-up is a popular technique and it is at best limited to moderately loaded structure, due to the materials involved, the curing conditions, and to the form in which the impregnation is accomplished.

The use of pre-pregs ensures the good impregnation of the reinforcement with resin, and besides this the polymeric resins used in pre-pregs generally present superior properties compared to those used in wet lay-up method.

However, compared to the wet lay-up process, pre-preg lay-up process requires well-controlled curing conditions and the manufacturing technology requires a vacuum bag and autoclave for a good consolidation of the structure and controlled increased temperature and pressure [5].

Aeronautical quality sandwich structure can be achieved via three ways: 1) One curing: co-curing by bonding the fresh skins to the core with or without the use of adhesive film; 2) Two curings: co-bonding way by using one skin cured and the other one fresh bonded to the core while curing; 3) Three curings: secondary bonding by curing the two skins separately and bonding them to the core using adhesive film [2].

Automated processes are necessary to satisfy the growing demand and the competitiveness of low-wages-countries.

The automatization decreases the production budget and could be reproducible at any level, saving productive hours by replacing the manual labor [7].

In the aerospace industry, the automated tape lay-up combining the cutting, lay-up, and compaction process is an accepted process [5] to manufacture flat parts, like wing skins.

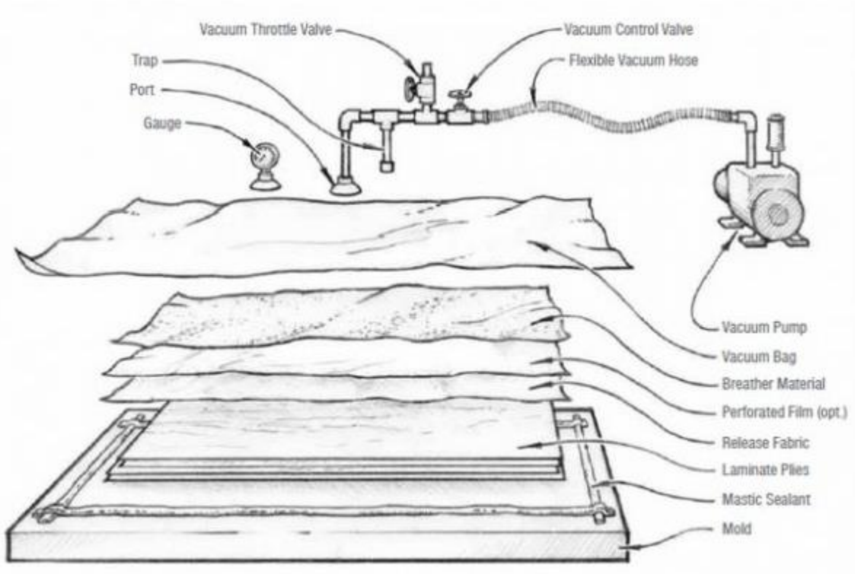

Fig. 2 – Vacuum bagging assembly [8]

Within the projects, it is necessary to validate the design by iterative structural calculation. Validations of the material properties and the allowable forces of the joints are required for this validation.

Because the production process of the test specimens is expected to begin later, numerical methods are used to find out these mechanical properties.

Numerical simulations are used for the virtual testing of specimens with the materials and joints used by the design.

These methods are proven to have a high accuracy of results, being comparable to those in experimental laboratory tests.

Modeling methods for composite materials used in aviation are validated by structural specimen tests that provide results equivalent to finite element (FEM) models. In the case of sandwich structures, the chosen methods use solid elements or shell elements depending on the application and the results pursued.

In such numerical methods the type of structural load is very important, it must respect the real load of the structure. Depending on the application, we can talk about linear static analyzes, which have a constant load, and nonlinear static analyzes, which have a time-dependent load.

If we are talking about the investigation of a sandwich insert configuration, we are talking about a nonlinear static analysis.

Dedicated software, such as ABAQUS or LS-DYNA, is required for such an analysis. Using such software can even highlight local effects such as delamination, local buckling, crushing or adhesive failure.

In the virtual testing of a sandwich insert you can choose the modeling with shell elements of honeycomb cells, “Meso mesh”, or the modeling with solid elements, “3D-continuum”. The insert, potting and adhesive are modeled using solid elements.

Because the laboratory specimens have 2 axes of symmetry, the model can be reduced to only a quarter of the specimen, and the boundary conditions are set using these symmetry conditions [9].

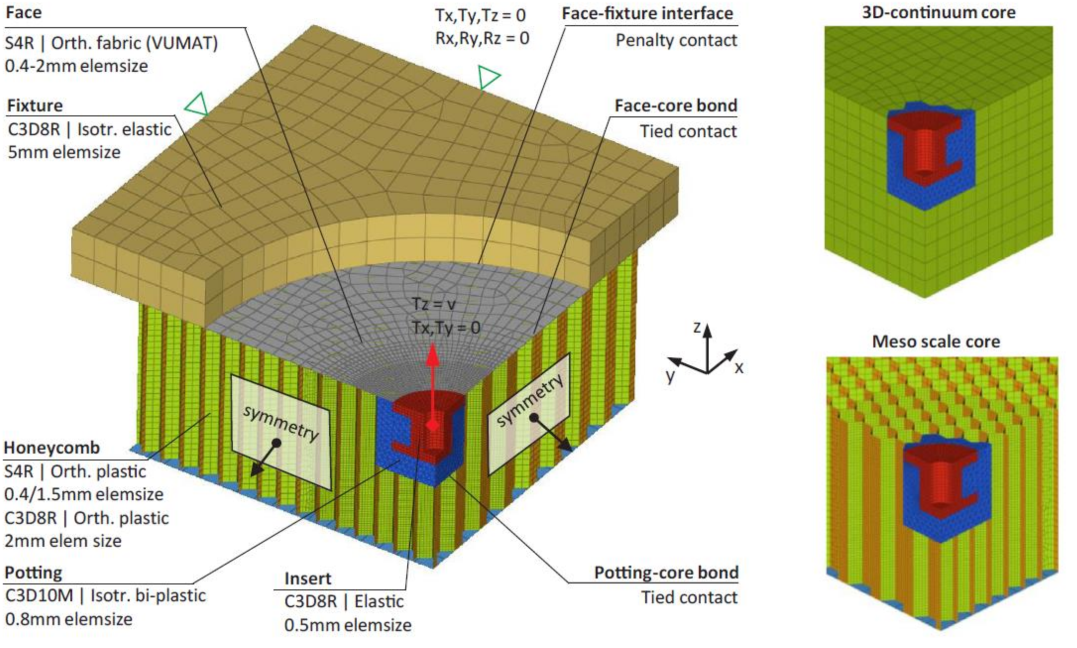

Fig. 3 – Implemented overall models with summary of implemented materials and model details [9]

The virtual testing results of both numerical models in comparison to the experimental results are provided in terms of the force-displacement relationship in Fig. 4. Overall, both simulation models show similar results with regards to initial stiffness, peak force, and stiffness degradation, and match the initial stiffness of the experiments well, while the simulated strength lies in the scatter of the experiments.

The curve progression in the following stiffness degradation phase is not matched as accurately by the simulation models. The test experiments were characterized by a continuous and moderate decrease in stiffness, while the virtual tests indicate a sudden load drop shortly after the peak force.

The carried load then stabilizes at a plateau before entering a continuous stiffness degradation similar to the experiments.

In general, load plateaus were also evident in the experiments, however, they were not as distinct and at considerably lower load levels [9].

In total, the simulation models generally underestimate the stiffness degradation after peak force [9].

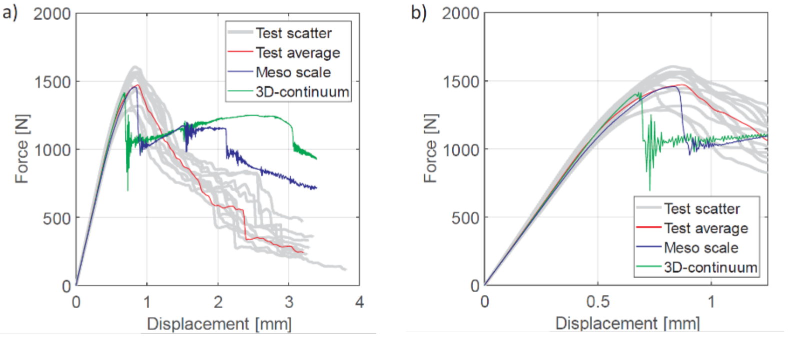

Fig. 4 – Comparison of simulations and experiments in terms of force-displacement relationship a) full displacement range and b) close up on damage initiation [9]

The paper presents technological process for obtaining sandwich structures with aeronautical quality and the analysis of their mechanical behavior during pull out mechanical test. The materials are investigated after mechanical testing using optical microscopy techniques.

The study results represent the basis for further experimental research that is required in the field of aeronautical quality sandwich structures.

2. EXPERIMENTS

Materials

The sandwich composite construction is based on Nomex honeycombs which are used as a core and HexPly® M18/1 prepreg which is used for skins.

The development of sandwich composite structures was performed by autoclave crosslinking technology with vacuum bagging technique. The specification of the prepreg is presented in Table 1.

Table 1 – M 18/1 characteristics

M 18/1 Physical Characteristic	**Value**	**Unit**
Density	1.22	g/cm^3
Fiber density	1.78	g/cm^3
Fiber areal weight	160	g/m^2
Glass Transition Temperature T_G Dry	196	˚C

The prepreg used, HexPly® M18/ 1, is based on epoxy matrix with superior properties, with the ability of self-extinguishing, of high performance suitable especially for use in primary aerospace structures.

It has a low moisture absorption at saturation. The laminate has a staking sequence [0/90] with quasi-isotropic characteristics [8, 10].

Processing and manufacture

The manufacture of sandwich composite structures was done according to the specifications of Airbus Helicopter using the autoclave crosslinking technology through co-curing, by placing the prepreg layers of carbon fiber directly on Nomex honeycomb (Fig. 5).

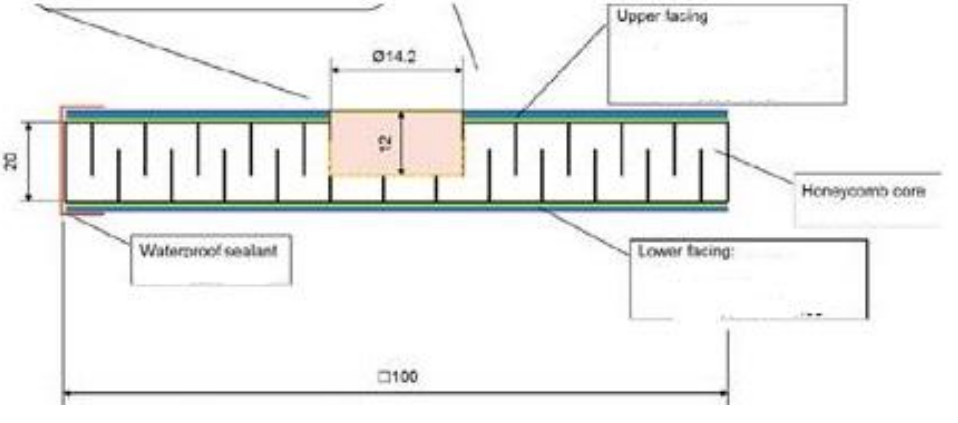

Fig. 5 – Schematic representation of the sample configuration [11]

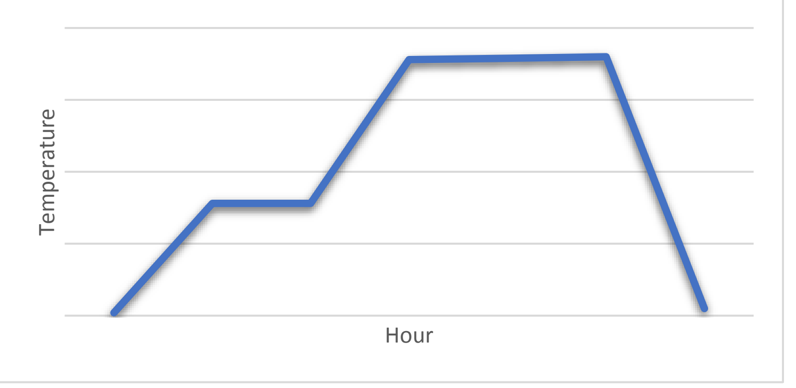

Fig. 6 – Polymerization chart

The composite sandwich samples were obtained by direct curing using SCHOLZ 442 Coesfeld L.W. autoclave, upgraded in 2010.

The pre-impregnated sheets of the material were placed directly on the honeycomb under controlled conditions and co-cured using gradual temperature increase on the two-heating dwell lapse of the oven with a temperature profile of 180°C, following the polymerization chart in Fig. 6. Before assembly, the prepreg sheets were cut at the requested dimensions with CNC cutter plotter.

Testing methods

After the development process was finalized, the obtained composite sandwiches were cut to obtain samples for mechanical tests with dimensions in accordance with Airbus Specifications.

Mechanical characterization. To evaluate the strength of a particular specimen through the dedicated tension inserts attached to it, the samples were subjected to mechanical testing (INSTRON 5982). The tests were performed on a minimum of 5 specimen per test, according to Airbus Helicopters Specification at 2 mm/min rate and pre-load 50 N [11] for pull out test.

Optical evaluation. The fracture mode was analyzed through microscopical techniques, morphological analysis was registered with optical microscopy using MEIJI 8520 microscope equipped with video camera and it was performed in the fracture area of the composite.

3. RESULTS AND DISCUSSIONS

3.1 Pull out testing

For the experimental determination of the maximum torsion force that the insert is able to transfer, before the test it will be checked if all specimens can undergo the limit load of the torque value. During the test it will be verified that neither cracking noise or insert rotation can be observed. The load is introduced in tension in insert axis and normal on the working face of the sample.

To verify the properties the experiment was designed to simulate the straining of the floor in the load direction corresponding approximatively to the pull out of the insert.

In order to analyze the resistance at tension, the mechanical tests were chosen to be able to have a clearer picture of the materials behavior and to observe the mechanism of detachment at different mechanical loads. For pull out tests, a number of 5 tests were tested according to the Airbus Helicopters Specification, applying speed test 2mm / min, speed test 0.5 mm/min until preload 50 N. Torque pre-test undergo limit load torque 12 Nm for each specimen.

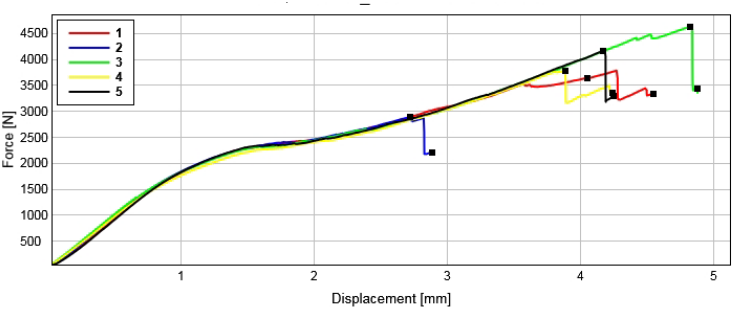

Fig. 7 – Displacement - force curve obtained during pull out test

The results of the pull out test are illustrated in Fig. 7 and Table 2.

Table 2 – Pull out test results for each specimen tested

Specimen No.	Force at tensile strength [kN]
Specimen 1	3.63
Specimen 2	2.88
Specimen 3	4.63
Specimen 4	3.78
Specimen 5	4.16

Analyzing the pull out test results, it can be observed that there is a difference between the 5 specimens (Table 2). Specimen no 2 showed lower values in terms of force at tensile strength compared to the other samples. This difference is most likely due to the torsion pre-test undergo of the samples. For future studies based on these results, the mean value will be calculated without taking into consideration the specimens with minimum and maximum values of the force at tensile strength. To complete the information on the behavior of the mechanically tested specimens, they were analyzed using optical microscopy.

3.2 Optical microscopy

Optical microscopy investigation was registered in the fracture area of the carbon fiber reinforced composite and it was performed with the scope of evaluating the fracture mechanism involved in the composite failure during pull out test. All tested specimens exhibited fracture of the fabric reinforced composite area near the metallic insert, as illustrated in Fig. 9 representing the photograph of one of the 5 specimens tested. No specimen exhibited the detachment of the insert from the composite part, suggesting that the adhesion between the two components was able to sustain the applied force.

Fig. 8 – Specimen tested at pull out (1, 2, 3- areas visualized with optical microscopy)

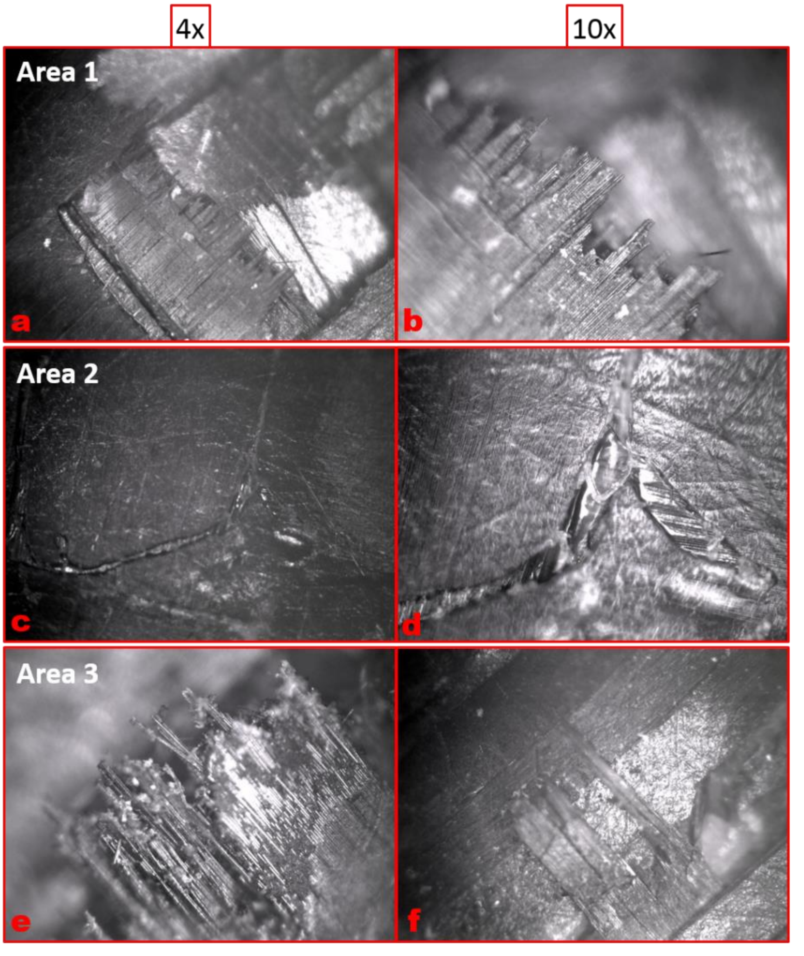

Fig. 9 – Optical micrographs of the 3 areas indicated on the specimen at different magnification levels – Area 1: a- x4, b- x10; Area 2: c- x4, d- x10; Area 3: e- x4, f- x10

Fig. 9 illustrates the optical micrographs captured in 3 areas marked on the specimen surface shown in Fig. 8, at two magnification level (x4 and x10). In area 1 (Fig. 9- a, b) and area 3(Fig. 9- e, f), it can be observed that the fibers fractured both on the external layer, as well as the subsequent layer, suggesting that the facture occurred in depth of the layered structure. The micrographs captured in area 2 (Fig. 9- c, d) suggest that in this zone the fracture seems to have occurred in the external layer, appearing only as a crack in the first fabric ply.

4. CONCLUSIONS

This paper studies the mechanical behavior and the implication of the parameters used in the manufacturing process of aeronautical quality sandwich composite panel obtained in the autoclave and developed under the RoRCraft CompAct grant. These mechanical tests have been performed in accordance with ASTM international standards.

The sandwich composite structures were analyzed from the point of view of the mechanical behavior during pull out testing and the results were corroborated with the optical microscopy evaluation to improve the technological quality of the manufacture process and achieve an optimization of the process.

Regarding numerical modeling methods, even if virtual testing with finite element models offers precise results and the possibility of evaluation of local phenomena, an excessive effort is necessary for sizing of numerous inserts exposed to various local conditions. In this situation, a simple and faster method is to perform laboratory experiments to use the results in “Simple” analytic theory and fast hand calculation, if possible.

The novelty of the paper stands in the experimental research analysis of a dedicated mechanical test consisting of pull out applied on sandwich structures, generally performed for simulations of floor straining. The research focused on corroborating mechanical test results with fracture modes observed by optical microscopy. The study results represent the basis for further experimental research that is required in the field of aeronautical quality sandwich structures.

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