SAILPLANE WING BOX DESIGN USING GRAPHITE/ARAMID/EPOXY MATERIAL

Giulio Romeo
Istituto di Progetto di Aeromobili
Politecnico di Torino, Italy

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SUMMARY

A sailplane wing design is being developed that extensively utilizes advanced composite material (graphite/aramid/epoxy). A cellular cross-section has been designed leading to the realization of an integral ribless wing box. The lower panel uses a hybrid graphite/aramid/epoxy material because of the impact resistance of aramid fibers; the upper panel uses graphite/epoxy material because of the proven better properties in compression.

A small panel has been manufactured and tested under uniaxial compression.

INTRODUCTION

In the 1970's, particular attention was paid to advanced composite materials for future aerospace structures because of their potential mass savings.

This objective is very important in soaring flight also for three main reasons: for high performance sailplanes, a larger wing loading variation is available between the minimum and maximum value, improving flight performance; for powered sailplanes, reduced engine power can be used with resulting beneficial effects in fuel savings; for training sailplanes, reduced minimum wing loading allows flight with a lower sinking speed.

In comparison with glass fiber reinforced plastic, advanced composite materials (graphite/epoxy and lower aramid/epoxy) present higher Young's and shear moduli leading to a higher bending and torsional stiffness of the wing.

A mass savings in the range of 11 to 24% has been obtained on sailplane wings by using advanced composite materials instead of some GFRP elements[1]. In Ref. 2, the theoretical mass savings of graphite/epoxy hat stiffened panels is up to 50% of the aluminum stiffened panel data presented by NACA. However, an experimental mass savings of 32 to 42% has been obtained.

A computer program has been designed to evaluate the mass optimization of hat stiffened panels[3]; however, because of the low airfoil thickness of sailplane wings, the utilization of such panels (and also open-section stiffened panels) does not give the same good structural efficiency. For this application, the honeycomb sandwich panels give better structural efficiency.

This paper describes the design and fabrication process of a new advanced composite cellular wing structure. Being a part of a wider program leading to the realization of a "Very Advanced Technology Light Twin" airplane, it has been developed during six month's research at the Delft University of Technology, Dept. of Aerospace Engineering, in cooperation with the technical staff and colleagues of the Composite Materials Work Group[4].

A wing box, 1300 mm long, 600 mm wide and 160 mm high, was manufactured of graphite/aramid/epoxy material and was tested by shear/bending loads (Fig. 1). Previously, a flat cellular panel 500 mm long and 155 mm wide was manufactured of graphite/epoxy by the thermal expansion molding process to get acquainted with fabrication techniques (Fig. 2).

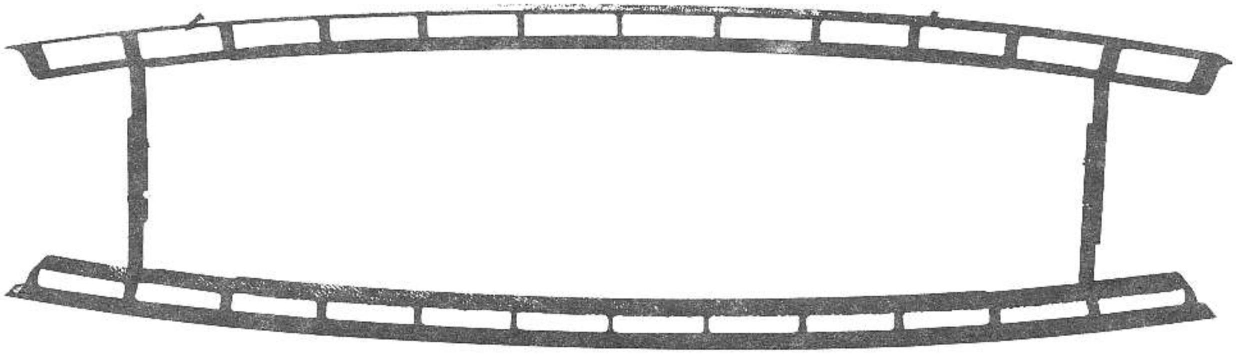

Fig. 1 Cellular Wing Box Specimen

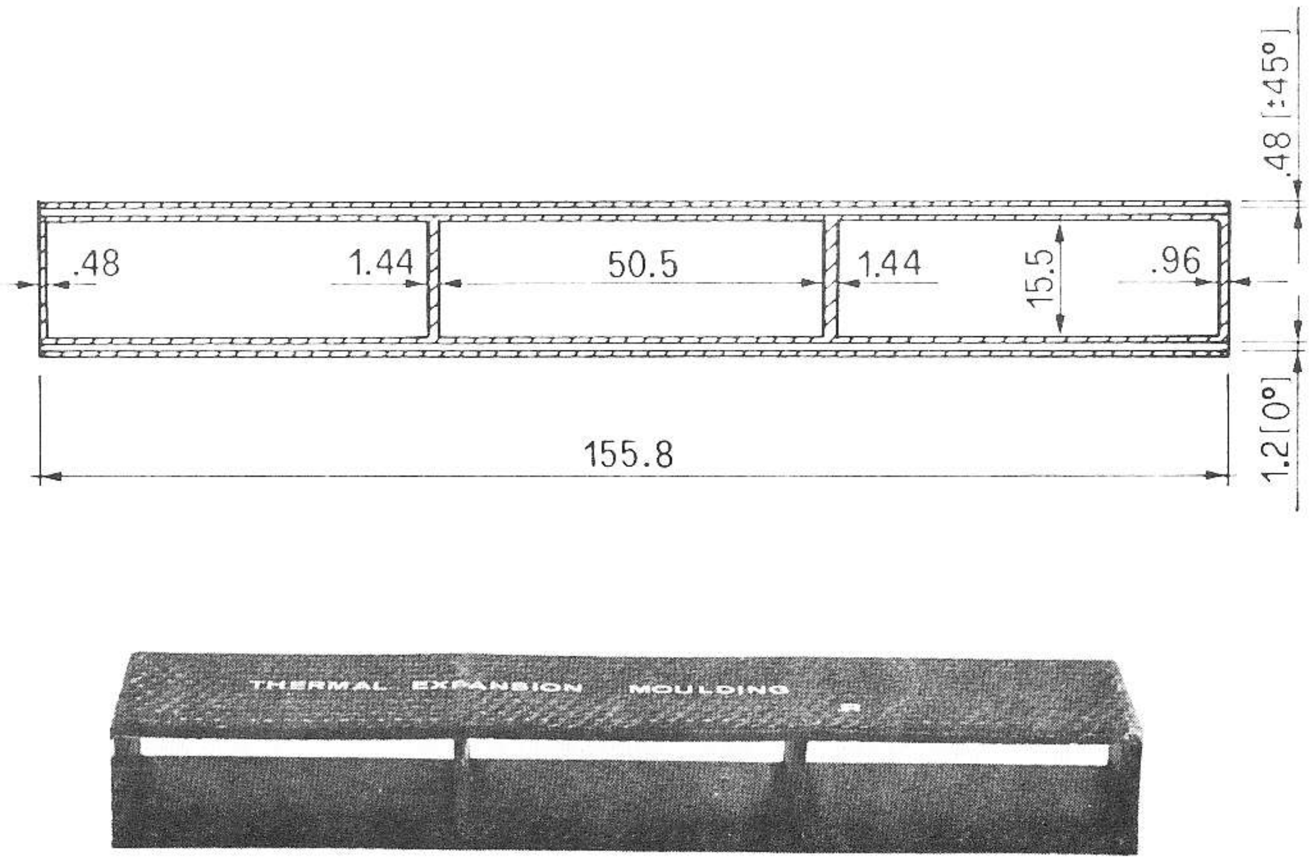

Fig. 2 Dimensions and Specimen of the Cellular Flat Panel

It is possible that such a structure would be of interest in the construction of a sailplane wing also. Unfortunately, I lack any reference data about current CFRP sailplane wings to make the necessary comparison on mass savings, production, maintenance, and repair costs, etc., to substantiate this premise.

Nevertheless, the design of a sailplane wing is reported on; it is related to the modified 15 meter M-300 glider conceived for an extruded wing[5] with a new minimum wing loading of 23 kg/m^2 and unchanged maximum value of 39 kg/m^2.

PRELIMINARY DESIGN

In the design of composite structures, the main advantages should come from realizing a structure with very few components, each one of them manufactured in a single cure cycle.

A possible typical cross-section of the wing is sketched in Figure 3; the central box will carry all bending loads and most of the shear/torsion loads supported in these lasts by the leading edge box. The central box has been designed like a cellular structure, leading to the realization of an integral ribless wing.

An extruded aluminum alloy wing box was manufactured first with such a shape[5], and a shear bending test was performed on a 7.67 m long wing which had only two ribs located at the wing/fuselage connection. The experiment demonstrated the capability of this stucture to withstand nearly the full prescribed loads, with special regard to the crushing loads on the webs; the failure occurred at a very high load factor ($n = 8.72$) in the central part of the structure where the webs were largely cut out to allow for connecting the fittings.

A similar structure made of composite materials could permit a mass savings of about 50 %.

Since a sailplane wing might frequently be subjected to impacts, the lower panel was designed of hybrid graphite/aramid/epoxy material because of the impact resistance of aramid fibers; the upper panel was designed of graphite/epoxy material because of the proven better compression properties.

The upper panel is cured in a single cure cycle including all the skin and the two half-webs; the lower wing box panel is cured in a single cure cycle including the two half-webs. They are connected by riveting the two half-webs along the span. The rear and front-lower panel joints have not been defined yet.

The wing had to withstand (according to the OSTIV Airworthiness Requirements) an ultimate bending moment of 35 kNm and an ultimate shear load of 9.5 kN, corresponding to a load factor of $n = 10$.

The stacking sequence of the panels is sketched in Figure 4. The inner and outer skins have been designed as a ±45°, 0°, 90° laminate, while the bridges consist of a ±45° laminate. More layers of fabric were used in the lower panel to improve both the shear stiffness and the impact protection; however, a reduced number of unidirectional layers were used because of the higher Young's tensile modulus in comparison to the compressive modulus. In this way, the wing box was designed

Fig. 3 Advanced Composition Sailplane Wing Cross-Section

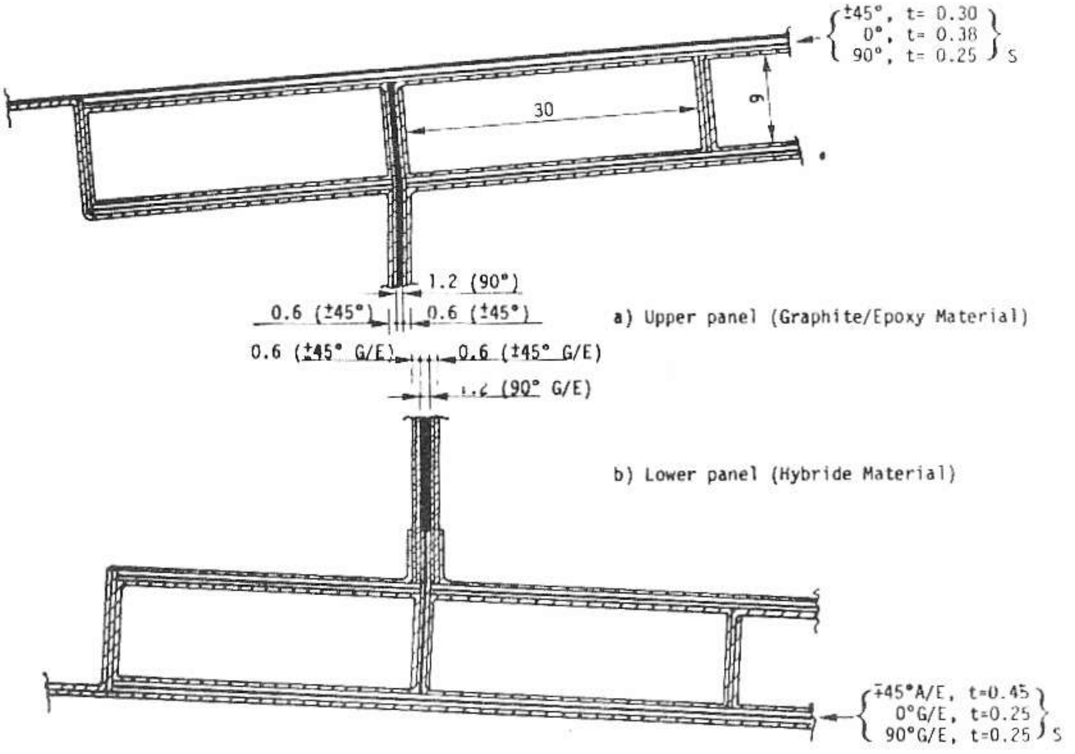

Fig. 4 Stacking Sequence and Dimensions of the Panels

with a structural symmetry. A ±45°, 90°, ±45° graphite/epoxy material was used in all half-webs for improving both the critical cruising loads and the bearing strength in the rivet holes (50% of ±45° and 50% of 90°).

The buckling stresses of the compression panel were calculated by using the simply supported orthotropic plate theory of stiffened panels[6]. It was designed to withstand a load index of 1350 N/mm to make the structure critical for local buckling of the laminate between two bridges.

The theoretical mass of the 9 meter untapered central wing box is 27 kg.

MATERIAL SELECTION

Since the temperature of 54°C is normally not exceeded, the resin system used for sailplane construction is cured at room temperature and postcured at 60°C; the fabric is usually impregnated with resin by hand and several layers are applied in order to obtain the required thickness.

For an airplane structure, a prepreg layer is used instead, realized by advanced composite materials and cured at a minimum of 120°C. Of course, this process is more expensive than the first one; however, some advantages result:

- better mechanical properties at high temperatures
- better alignment of the fibers and close control of the resin content
- reduced time during the layup process
- less toxic and cancer effects on the workers' health

The material used in manufacturing the wing box was unidirectional graphite/epoxy T300/F550, the square fabrics graphite/epoxy T6341/R2503, and aramid/epoxy kevlar 49/R2503. They were prepregged by Hexcell Belgium, aiming at a cured fiber volume content of 60°.

The elastic mechanical properties of the unidirectional material are reported in Figure 5. The discontinuity shown in this figure is due to the use of different specimens for the tensile and compression test; however, in testing the compression specimens starting from a slight tensile load, no discontinuity at zero load was observed[7].

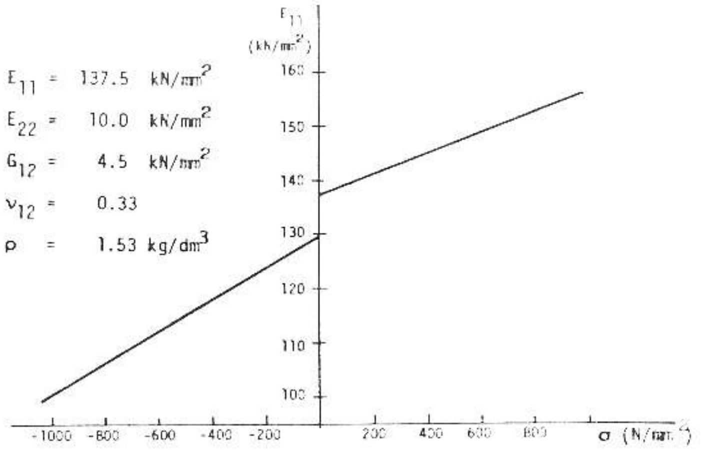

Fig. 5 Young's Modulus/Stress Curve for Unidirectional Material T300/F550 Hexcel

PRODUCTION PROCESS

In an attempt to avoid the use of an autoclave process, some experiments have been carried out with a thermal expansion molding technique and the panel of Figure 2 was produced. Indeed, it became clear that, through this process, the pressure rise caused by solid expanding rubber is badly controlled by the temperature because there will always be an unpredictable volume gap in the closed mold, so this technique has been set aside.

The compression test on this panel to check the capability to withstand the design compression stress without local buckling failure has not been fully satisfactory; the failure occurred at a load of 150 kN with local buckling of the laminate between two bridges. It was followed by an explosive delamination of the skin, Figure 6. However, telegraphing effects and initial imperfections were present in the laminate and both probably caused a reduction of 25% of the local buckling load.

Better results should be possible using the inflated rubber technique[2,8]. However, for simplicity, the panels of the wing box of Figure 1 have been manufactured on a mold that, combined with solid silicon rubber cores, was suitable for an autoclave controlled pressure cycle using a vacuum bag, Figure 7.

In the sailplane production, the main disadvantage of this process is the increased cost of the long autoclave equipment; therefore, from this point of

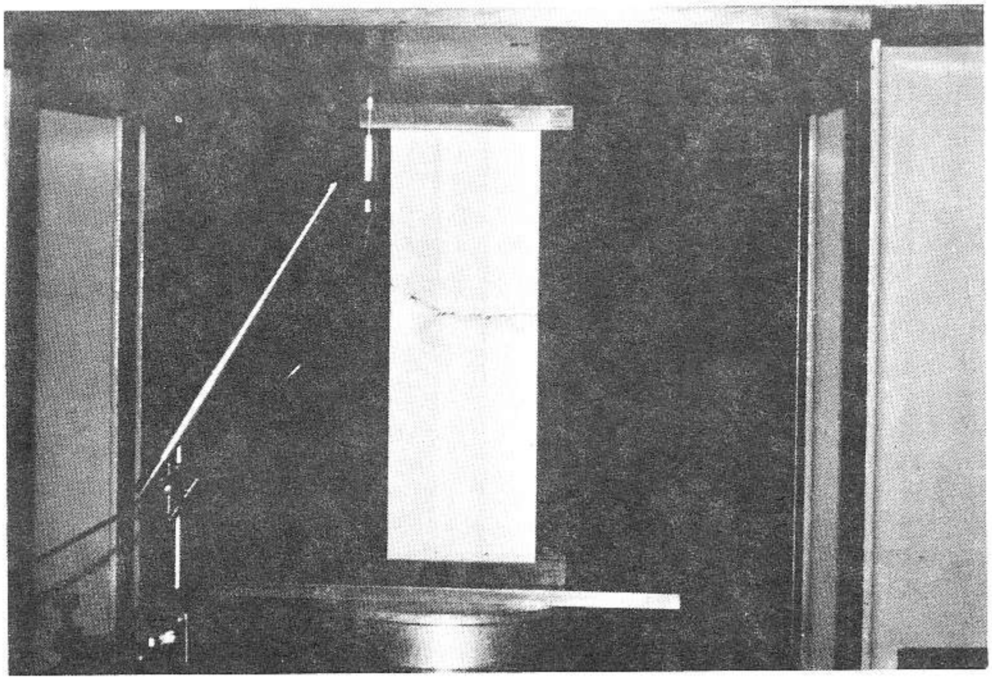

Fig. 6 Compression Failure of the Cellular Flat Panel

view, the inflated rubber technique is preferred since the rubber expansion is easier to control.

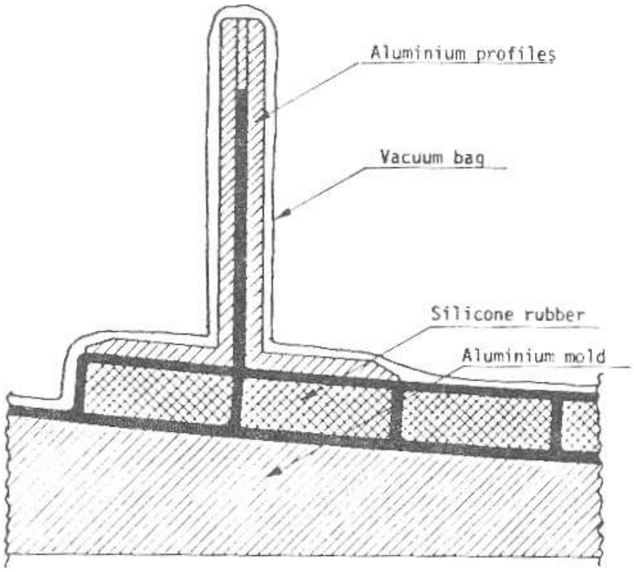

Fig. 7 Vacuum Bag Stacking Sequence

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