Using 3D fabric on light glider fuselage

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1. Introduction

A new material, which is called the Paraglass, has been used for the development of the ultralight composite glider TST-10 Atlas, unlike usual technology of the sandwich construction. The glider TST-10 Atlas is the cantilever midwing momoplane with the T tailplane. The take off engine is installed on the stow-away arm.
The Paraglass is made by the Dutch company Parabeam in Helmond. The advantage of this material is very simple and cheap technology of the structure production. The material is used as middle lay of all coverings, that are skin of wing, fuselage and tailplane.

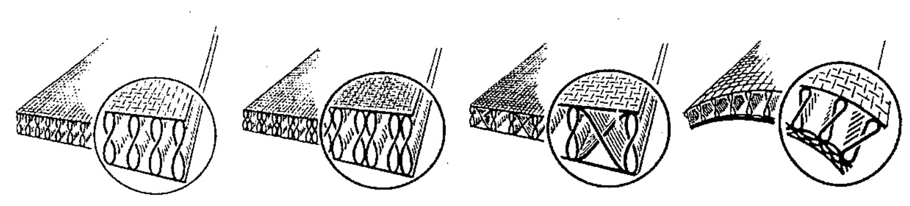

Figure 1. Types of Paraglass material

Figure 2. TST - 10 Atlas sailplanes

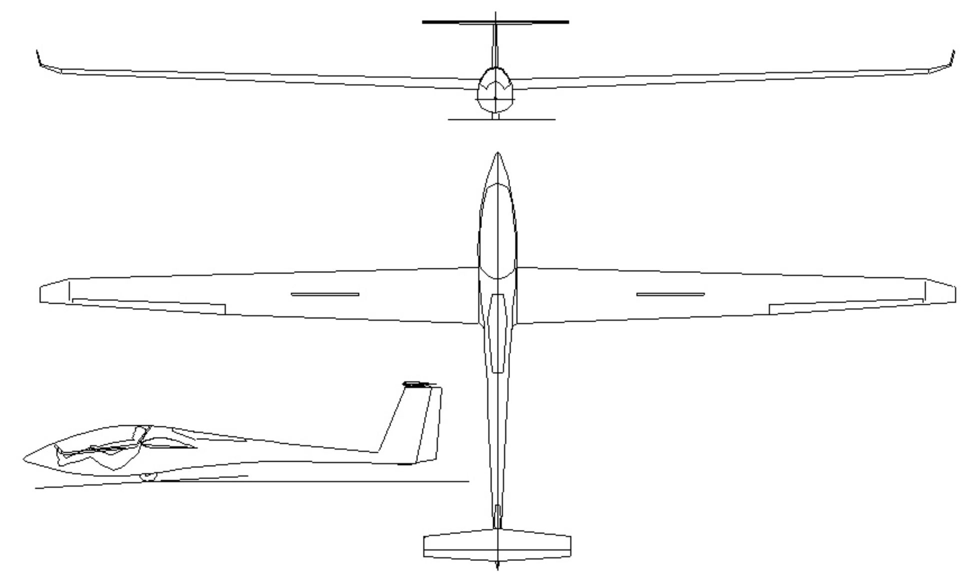

Figure 3. TST - 10 Atlas sailplane

2. Material Tests

Material and technology tests were done during the preparation of production. The poster presents results of the tensile tests and the share tests on a frame.

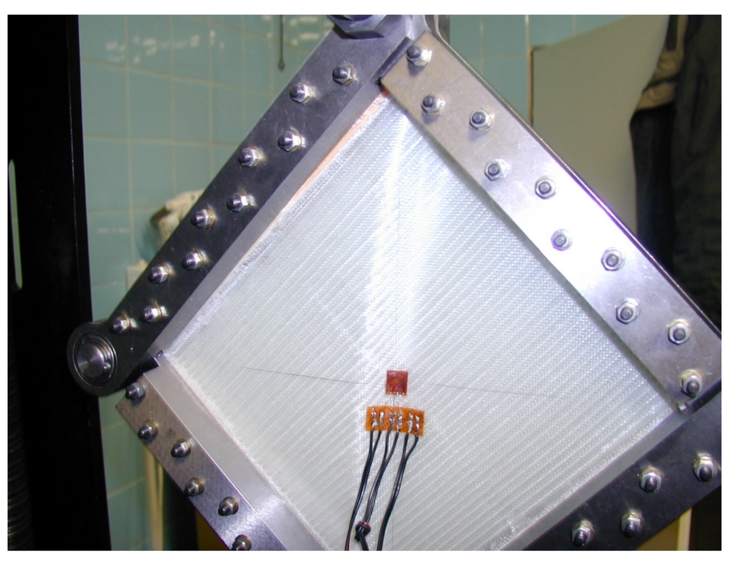

Figure 4. Share test on frame

Specimen		Dimen. (mm)			E modul		Rm	qx
nomber	description	a	b	l	(Mpa)	(1)	(Mpa)	(N/mm)
1	fabric 2x92 110/0st	0.33	19.5	20	12321	-	204.49	67.48
2	fabric 2x92 110/90st	0.33	19.5	20	10641	-	160.41	52.94
3	fabric 2x92 110/45st	0.33	19.5	20	6336	-	102.07	33.68
4	fabric 2x92 110/0st	0.35	19.7	20	11472	-	158.68	55.54
5	fabric 2x92 110/90st	0.34	20.2	20	10048	-	140.97	47.93
6	fabric 2x92 110/45st	0.35	20	20	5804	-	76.64	26.82
7	Paraglass 10/0st	9.7	19.4	20	852	0.162	25.49	247.28
8	Paraglass 10/0st	9.7	19.7	20	794	0.096	18.45	178.94
9	Paraglass 10/90st	9.3	19.6	20	1051	0.404	11.34	105.48
10	Paraglass 10/90st	9.7	19.3	20	996	0.071	17.61	170.78
11	Paraglass 5/0st	6.3	19	20	961	0.216	23.07	145.34
12	Paraglass 5/0st	6.25	19.2	20	1066	0.161	27.04	168.98
13	Paraglass 5/90st	6	19.6	20	578	0.107	17.62	105.72
14	Paraglass 5/90st	6.2	19.2	20	1032	0.290	24.86	154.14
15	Paraglass 3/0st	3.1	19	20	2840	0.094	55.42	171.79
16	Paraglass 3/0st	2.7	19.8	20	3193	0.181	53.40	144.18
17	Paraglass 3/90st	3.2	19	20	2607	0.101	58.08	185.84
18	Paraglass 3/90st	3.2	20.6	20	2831	0.107	48.62	155.58

Figure 5. Results of tensile tests

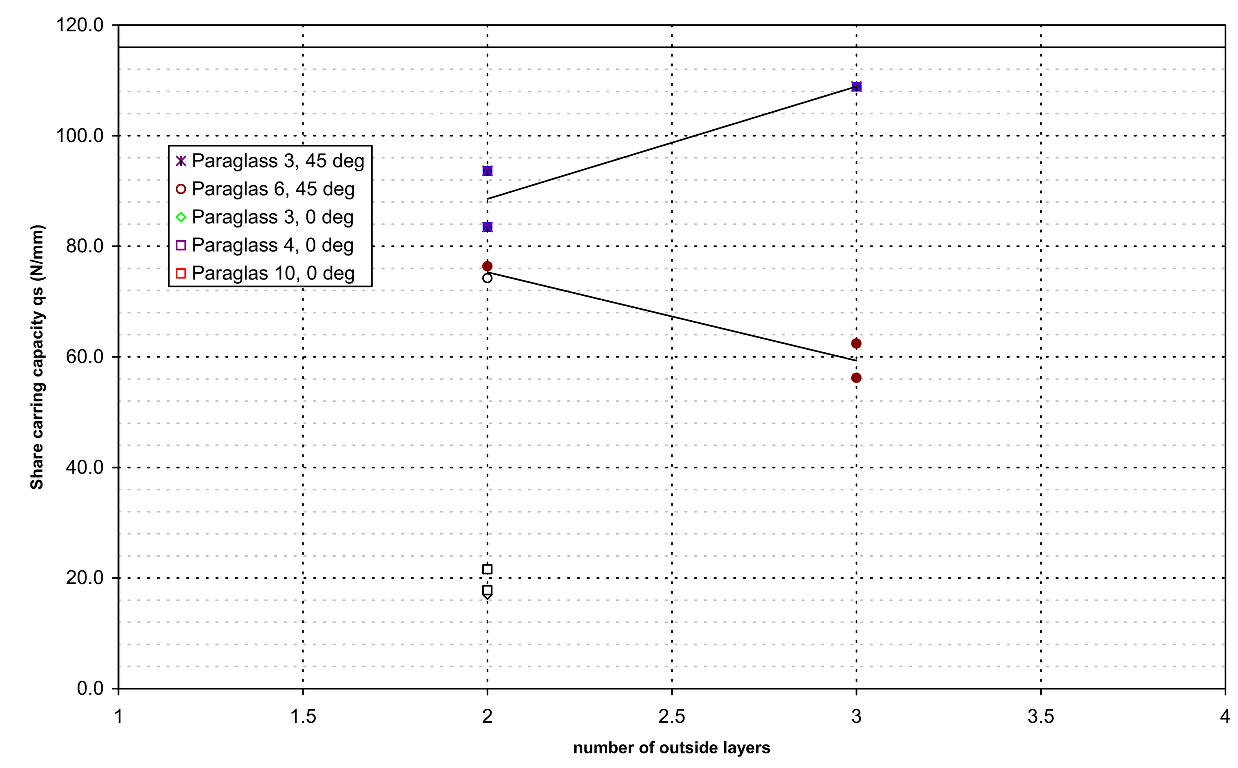

Figure 6. Results of the share test

3. Loading

Several cases are crucial for the loading of fuselage because both the bending and the twist are carried with the fuselage structure. A maximum force on the horizontal tail unit is one of the important bending cases. The resultant force, applied in the horizontal tail hinge F_{HTU}= 811 N, was determined for this case (Case 1). This force presents air load after deducting inertial effects.
The second important case is the bending to a side simultaneously with the twist of fuselage (that is load from the vertical tail unit). The force F_{VTU}= 811 N and the twist moment Mx = 852.95 Nm presented load of Case2.
Load cases mentioned above are limit loads. A carrying capacity was calculated with the safety factor 1.5 and the special coefficient for composites 1.5. Ultimate loads were 2.25 times bigger than limit ones.

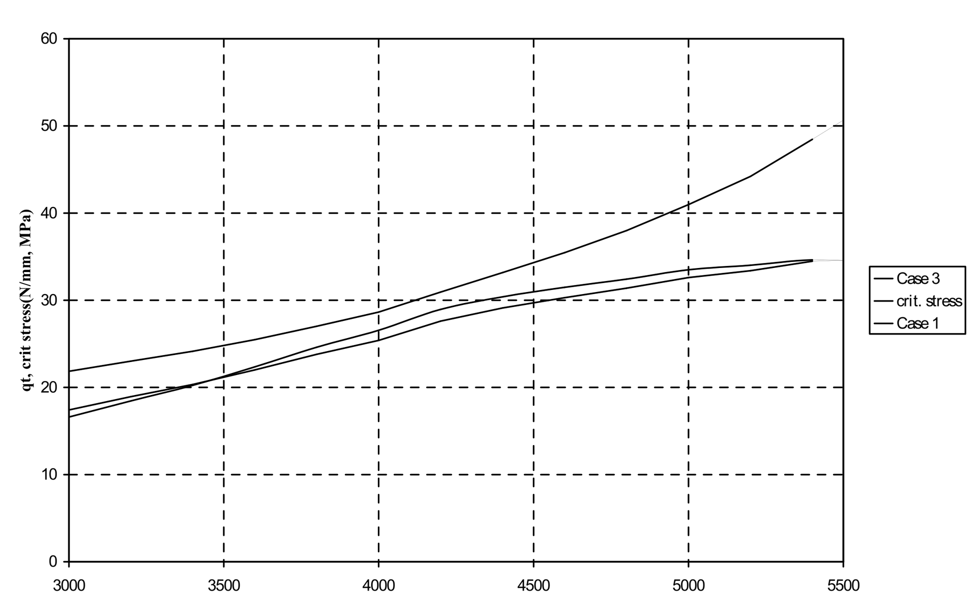

Figure 7. Result of analysis

4. Analysis

The first step of analysis was a simple calculation. The shear forces, bending moments and the torsion moment along fuselage were calculated. Then the tensile and shear flows (forces on the length unit) were calculated. The maximal size of the tensile flow q_t=38 N/mm (Case 1) and the shear flow q_s=14,7N/mm (Case 2) were determined in the area, where the fuselage was changed into a fin.
These loads represent tensile or compression stress 45 MPa and shear stress 23 MPa in the upper lay of laminates. Both values are low for achievement of strength.
However, this judgment is not sufficient for an appraisal of the carrying capacity. The effect of buckling on the fuselage has to be taken into account for the next appraisal. A similar equation, like for curved isotropic sheet, was used for the first establish of critical area and value of critical compression flow determination:

$$q_{KRc} \quad kE\,\frac{t}{b}^{\,2} \quad 0{,}3E\,\frac{t}{R}\;t$$

The critical area of buckling of bending case was predicted from the ratio of critical flow and limit flow in the coordinate range 3.8-4.4 m from the fuselage nose. The value of a reserve factor, by which the given failure comes about, could not be defined without experimental verification at the panel or directly at the fuselage structure.
The second step of analysis was FEM modelling and non-linear solution. MSC Patran/Nastran system was used for FEM analysis. By FEM analysis, the failure was established with reserve factor 2,15in the area with coordinates 3.8 m.
The area, where fuselage is changed into fin, was determined as a critical one from an analysis of shear flow (Case 2) and its influence on structure. But from the view of the carrying capacity we could take the reserve as sufficient.

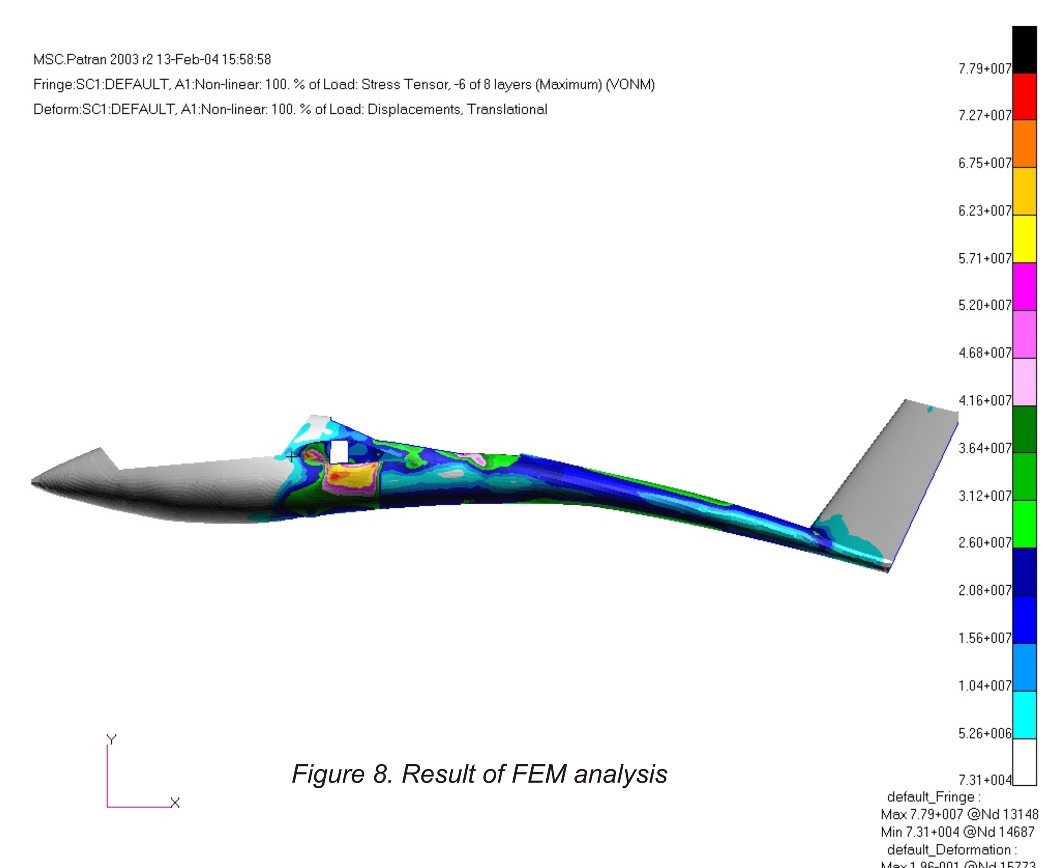

Figure 8. Result of FEM analysis

5. Fuselage testing

The results of analysis and structure carrying capacity were verified during tests. The bending test proved ability to carry the maximal horizontal tail unit force F_{HTUmax} = 1638 N that is 184% of limit load. The failure came about at the point 3.92m. But the failure area had a bad bond of both halves of the fuselage body.
There was another test after the repair. It was for the definition of a further failure point, which was found at the load value F_{HTUmax} = 1946 N (ie. 219% of limit load) and the failure area moved at 3.72m. It means that the area of failure was moved by the repair effect (local reinforcing) about 200mm forward.
The next critical case was Case 2. The fuselage was also tested on the twist and the bending to a side. The fuselage carried a side force on the vertical tail unit F_{VTUmax}= 1890 N and the induced rolling moment Mx = 324 Nm (ie. 240% of limit load for the Case 2 at twist). The test was stopped without failure.

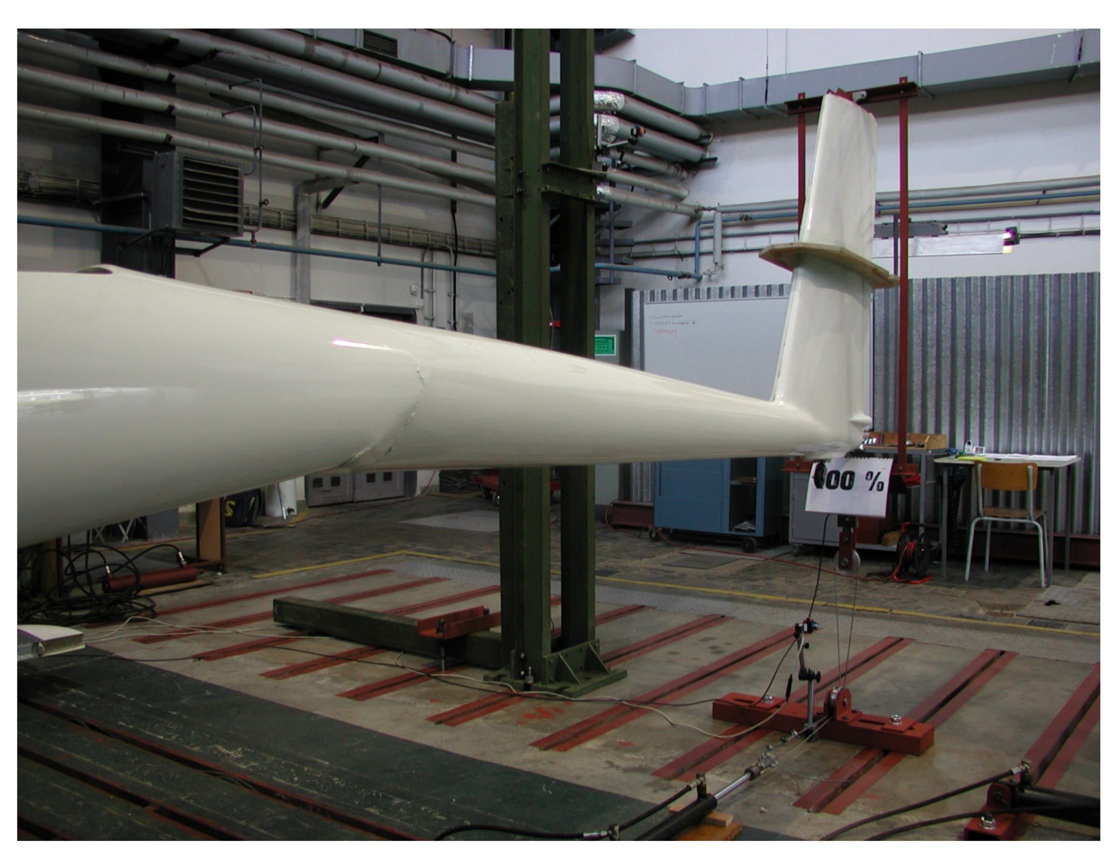

Figure 9. Fuselage test - Case 1

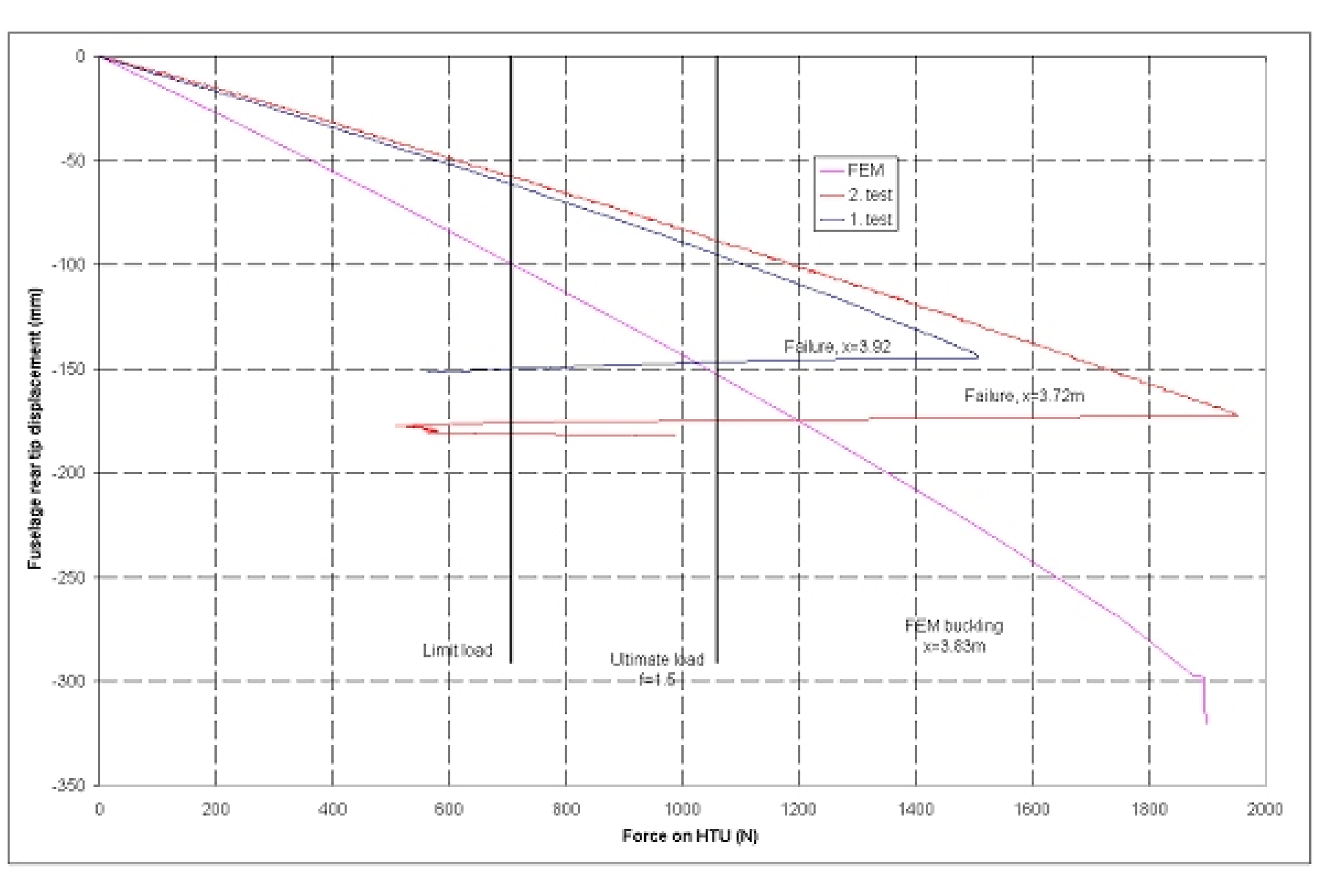

Figure 10. Results of fuselage test - Case 1

6. Conclusion

Tests confirmed the expected assumptions and the ability of fuselage to carry required loading. The agreement was proved at the prediction of the failure area of structure. As we assumed there were no exact agreements at the determination of load amount, by which the failure comes about.

Figure 10. Fuselage test - Case 1