FATIGUE TESTING OF A CARBON FIBRE REINFORCED EPOXY TAILPLANE

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ABSTRACT

As part of the structural testing programme for the prototype carbon fibre composite trainer aircraft developed in South Africa, static and fatigue tests were carried out on the tailplane, in general, according to the guidelines of FAA Advisory Circular AC 25.571-1A. The series of tests planned for the tailplane consisted of fatigue tests at room temperature for the equivalent of three fatigue lives (45 000 hours), interrupted at regular intervals by static tests under different environmental conditions to limit and ultimate load. Strains and deflections at several positions were recorded during the testing. Non-destructive techniques, X-radiography and transient thermography, were used in an attempt to monitor the propagation of the defects built into the tailplane to simulate manufacturing flaws and the defects caused by impact loads to simulate in-service damage. The results obtained so far indicate strain levels well below 4000 microstrain and barely any propagation of the defects.

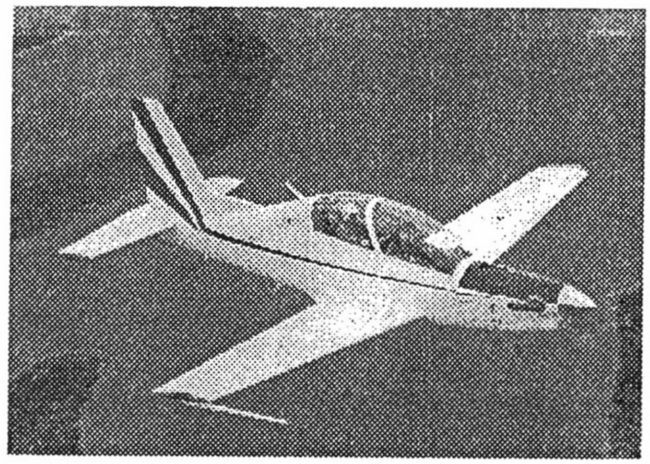

Figure 1 : Prototype Carbon Fibre Composite Trainer Aircraft.

Introduction

Towards the end of 1985, Atlas Aviation and the Division of Aeronautical Systems Technology (Aerotek) of the CSIR joined forces to design and manufacture a carbon fibre composite aircraft. A conventional turboprop tandem two-seat trainer was selected because the South African Air Force were planning to replace their ageing Harvard trainers. The decision to make the airframe from composite materials was taken because wide experience with these materials had been gained, particularly at Aerotek, during the development of products such as autogyro rotor blades, radomes, cowlings and RPV's. The use of composites in the airframe was also in line with observed world trends. During the following five years a prototype aircraft was built as a technology demonstrator. The airframe was subsequently subjected to a fairly comprehensive structural testing programme to satisfy the designers that the structure was sound and this was followed by extensive flight testing. All the design objectives were either met or exceeded. The aircraft is shown in Figure 1 and details of its development are given in Reference 1. Some basic data are presented in Table 1.

Table 1 : Basic Aircraft Data

Maximum take-off mass (aerobatic), kg	2200
Maximum take-off mass (utility), kg	2750
Maximum level speed, KTAS	266
Stall speed (clean), knots	76
Stall speed (flaps down), knots	68
Take-off distance to 50 ft, m	365
Landing distance from 50 ft, m	375
Maximum positive load factor, g	+7,0
Maximum negative load factor, g	-3,5
Maximum endurance at 25 000 ft, hours	5,5
Maximum range at 25 000 ft, nm	1100
Rate of climb, ft/min	3100
Engine	PT6A-34 750 HP

The structural testing carried out on the airframe was considered adequate for a prototype but would not be sufficient for a production aircraft. Testing based on current FAA certification regulations[2,3] would be required and, for certification to FAR 23[4], this would involve the fatigue testing of the wing. As the funds allocated to the project were very limited, a wing for testing purposes could not be built. However, it was considered that the testing of a smaller structure to meet certification requirements would be of considerable value and a tailplane was manufactured for this purpose. Details of the test procedure and some of the results obtained are presented in this paper.

Design and Manufacture of the Tailplane

Tailplane Dimensions

Based on the dimensions of the wing (10,8 m span and 18 m^2 area) and the position of the aircraft's centre of gravity, the span and the area of the tailplane were optimised to be 4,3 m and 4,32 m^2 (including the elevator) respectively. The dynamic pressure at the tail, the wing/fuselage influences and the ground effect were also taken into account when sizing the tail.

Loads on the Tailplane

The loads on the tailplane were determined in accordance with AvP 970 for spanwise loading and FAR 23 for chordwise loading. Eight load cases were considered for different centres of gravity, for different centres of pressure and for the complete speed range. The maximum tail load was found to be 17 088 N with the centre of gravity at 22% m.a.c., its most forward position, and with the centre of pressure at 18% m.a.c.

Construction of the Tailplane

The tailplane is a sandwich structure, carbon fibre reinforced epoxy skins and Nomex honeycomb core, constructed in two halves which are bonded together. The ribs and the front and rear spars are also of sandwich construction. The prepreg used for the skins consisted of an epoxy system that cured at 120°C, reinforced with plain weave carbon fabric of areal mass approximately 200 g/m^2. Unidirectional carbon prepreg was used for the spar caps. The surface layers of the upper and lower skins of the tailplane consisted of glass fibre fabric prepreg, to reduce the risk of barely visible impact damage.

The upper and lower skins were bonded together, and the spars and ribs bonded to the skins, using a 120 °C curing epoxy adhesive. A schematic representation of the tailplane is shown in Figure 2.

Built-in Defects

During the manufacture of the tailplane, circular patches of ptfe were placed in various positions to simulate manufacturing defects. The types of defects considered were:

(1) debonding or bad adhesion between rib and skin;
(2) debonding or bad adhesion between spar cap and skin;
(3) delamination or bad adherence between plies;
(4) debonding of honeycomb core and skin; and
(5) debonding between plies at the edge of the chamfered honeycomb, where the two skins join. (This is shown schematically in Figure 3.)

The defects were located in regions where their presence would be of some significance: where they would increase stress concentration, particularly in shear, or promote local buckling.

Two circular patches of ptfe were used for each defect to ensure that there was an air gap between adjacent surfaces.

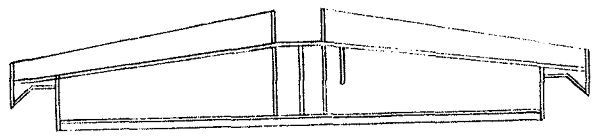

Figure 2 : Schematic Representation of the Tailplane

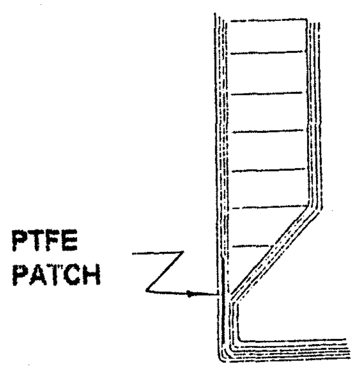

Figure 3 : Built-in Defect between Plies at the Edge of Chamfered Honeycomb

Fatigue Test Spectrum

Experimental tailplane load data for trainer aircraft was not available so a conservative approach towards developing a fatigue load spectrum was adopted. It was impractical to derive a spectrum from the utilisation and mission profiles, so a combination of the Falstaff and CAA aerobatic spectra was used. When compared with the loads measured for the SAAF Impala MK1 aircraft, which has a similar training role, the spectrum was found to be conservative. The spectrum is defined in Table 2.

Table 2 : Fatigue Load Spectrum

Load Factor n	$\frac{n}{n_{max}}$	No of Cycles in 1 Fatigue Life	No of Test Cycles
7	1,0	500	500
6,3	0,9	2375	1875
5,6	0,8	10000	8125
4,9	0,7	31000	22875
4,2	0,6	70000	47125
3,5	0,5	150000	102875
2,8	0,4	300000	197125
2,1	0,3	562500	365375
1,4	0,2	750000	384625
0,7	0,1	250000	210750
0	0	67500	39250
-0,7	-0,1	40000	28250
-1,4	-0,2	15000	11750
-2,1	-0,3	3250	3250
-2,8	-0,4		
-3,5	-0,5		

The fatigue load spectrum defined in Table 2 refers to loads at the wing root. However, tailplane loads can be related to wing loads in any flight condition by considering the equilibrium of the aircraft around its centre of gravity. For a symmetrical pitching condition the fatigue load spectrum for the tailplane consists of the values shown in Table 3.

Table 3 : Tailplane Fatigue Load Spectrum

Test Cycles (1 Fatigue Life)	Tailplane Load (N)		
500	-1648	to	-10834
1875	-1648	to	-9702
8125	-1648	to	-8691
22875	-1648	to	-7619
47125	-1648	to	-6547
102875	-1648	to	-5476
197125	-1648	to	-4404
365375	-1648	to	-3332
384625	-1648	to	-2260
210750	-1648	to	-1189
39250	-1648	to	-117
28250	-1648	to	+955
11750	-1648	to	+2026
3250	-1648	to	+3098

This spectrum representing one fatigue life, equivalent to 15000 hours, was divided into 125 blocks, each equivalent to 120 hours. The cycles within each block were randomized and filtered to remove small spikes.

The final signal was multiplied by a factor of 1,1 to account for the variability of the material and the effect of the mass of the whiffle tree was also included.

Testing Procedure

Test Programme

The structural tests planned for the tailplane consisted of static tests at room temperature, 54°C and 72°C, and fatigue tests for a total of three lives, equivalent to 45000 hours. It was essential that the structure be tested under environmental conditions at least as severe as those expected in service. The planned test sequence was as follows:

1. static tests at room temperature, 54°C and 72°C up to limit load, measuring strains and deflections at suitable locations;

2. fatigue test at room temperature for one life of 15000 hours, using the load spectrum developed for the tailplane;

3. static tests at room temperature, 54°C and 72°C up to limit load, measuring strains and deflections, after inflicting large, visible impact damage in the tailplane shell;

4. static tests at room temperature, 54°C and 72°C up to ultimate load, measuring strains and deflections, after repairing the impact damage;

5. fatigue tests at room temperature, for a further two lives (30000 hours);

6. static tests at room temperature and 72°C to ultimate load, measuring strains and deflections;

7. static tests at room temperature and 72°C to ultimate load, measuring strains and deflections, after conditioning the tailplane at 45°C and 85% relative humidity to reach an equilibrium moisture content level; and finally;

8. static test at room temperature to failure to check whether the mode is fail-safe.

Impact Defects

An important feature of the testing of the tailplane is the demonstration of damage tolerance to defects caused by impact. Low velocity impact, for example, dropped tools, and high velocity impact caused by, for example, small objects on the runway, have to be considered in addition to normal damage like skin punctures. Barely visible impact damage is a problem in structures made from carbon fibre composites. In order to check the resistance of the tailplane to such damage, visible and barely visible impact damage was inflicted at several positions. A drop tool was designed for this purpose, of a specific mass to impart a known level of impact energy after release from a given height. Most of these impact loads were applied to the skins above or below the front and rear spars, in regions where there would be compression loads.

To prove the tolerance of the tailplane to relatively large visible defects and the effectiveness of the repair scheme, it was planned that a hole of approximately 25 mm in diameter be punched into the starboard section of the lower skin, just forward of the front spar, after one fatigue life, and this would then be repaired after static tests. A hammer would be used to inflict the damage and a scarf bonded patch of carbon fibre reinforced epoxy would be used to repair the damage.

Details of such a repair scheme can be found in Reference 5. The scarf bonded patch technique is particularly suitable if the damage size is limited to a hole of approximately 150 mm diameter, if the repair must be made in place and there is access to the damage from one side only. The patch, laid up and cured in the tailplane mould, would be bonded into the tailplane after being machined to the correct dimensions. This type of repair preserves the aerodynamic shape of the tailplane and the joint can have up to 80 percent of the strength of the original material.

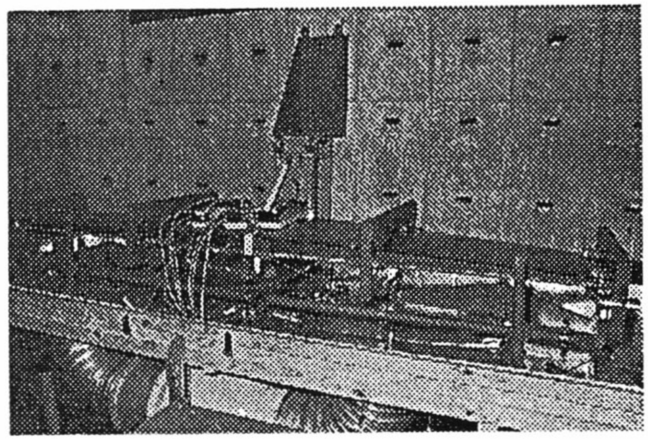

Figure 5 : Tailplane in Test Rig

Test Equipment

The static and dynamic loads were applied to the tailplane by means of a servo-controlled hydraulic jack of 40 kN capacity through a whiffle tree arrangement. The loads were measured using a load cell in series with the jack. Strain gauge rosettes were bonded in various positions on the skins, spars and ribs of the tailplane so that the principal strains could be determined. LVDT's were mounted in several places on the tailplane to check deflections.

The signals from the LVDT's and the strain gauge rosettes were amplified, monitored and recorded with the aid of three HBM UPM 60 multipoint measuring units. The signals for the load spectrum were generated by computer and fed into the servo-control system of the hydraulic jack.

The test set-up is shown in Figures 4 and 5, and in Figure 6, a schematic diagram of the whiffle tree arrangement is given.

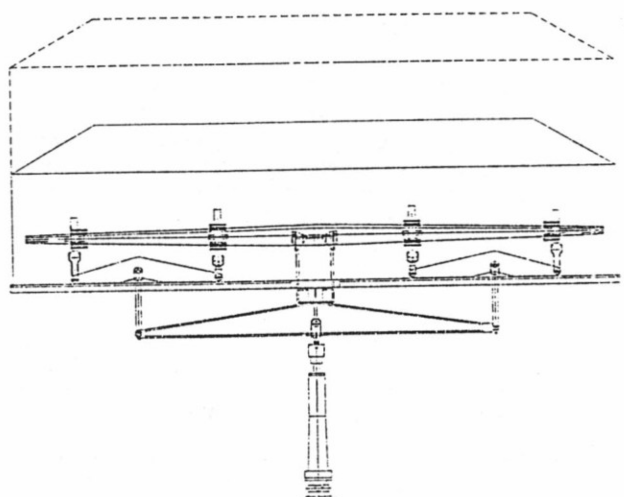

Figure 6 : Schematic Representation of Whiffle Tree

An environmental chamber for the elevated temperature tests and for the moisture conditioning was specially constructed by Atlas Aviation. It consisted of a steel frame with large perspex windows, and could be lowered on to or lifted from the wooden platform under the tailplane, shown in Figures 4 and 5, comparatively easily.

Figure 4 : General View of Test Set-Up

Results

Because the test programme is still in its early stages, the results obtained so far are limited. Only one series of static tests and a fatigue test to one lifetime have been completed so far.

Values for tip deflection at limit load are given in Table 4. These deflections are quite small for a tailplane of 4,3 m span. Elevating the temperature did not markedly increase the deflection.

In Figure 7, load-deflection curves are plotted for the static tests carried out at room temperature, 54°C and 72°C. The curves are essentially linear up to limit load.

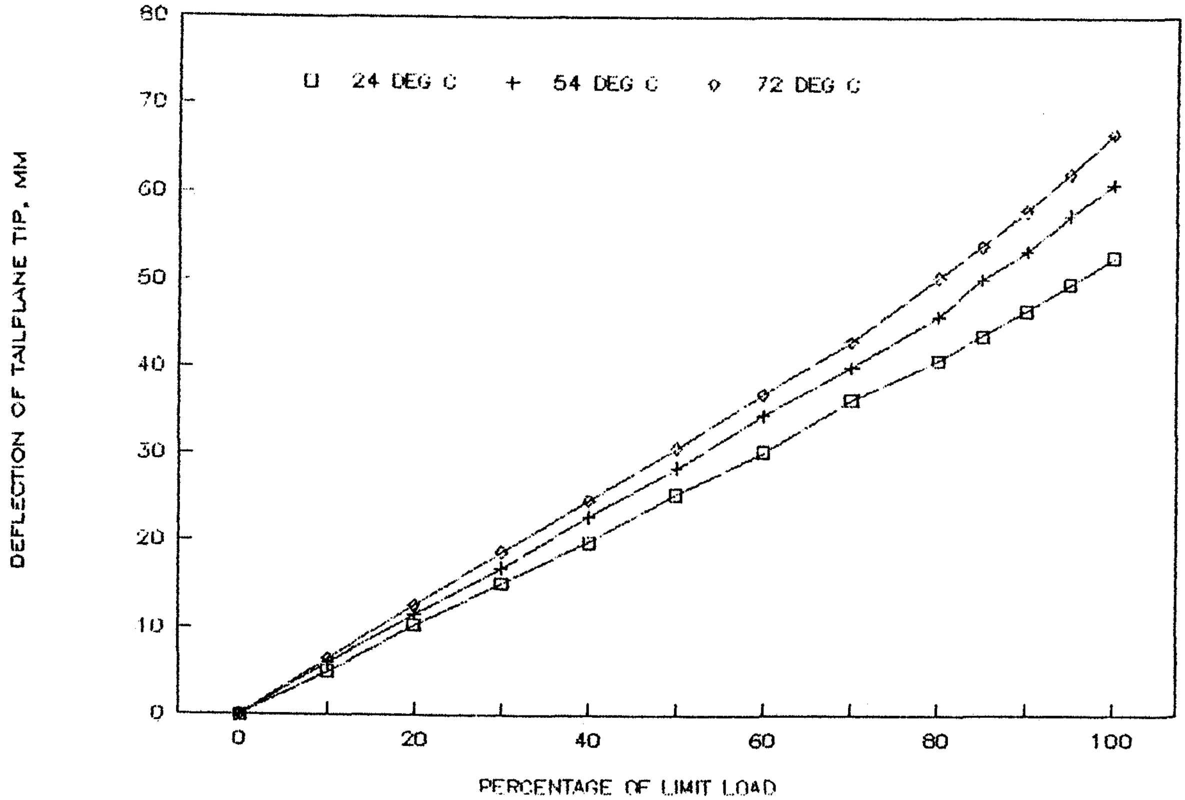

Figure 7 : Load-Deflection Curves

Table 4 : Deflection of Tailplane Tip at Limit Load

Test Temperature, °C	Tip Deflection at Limit Load, mm
24	52,65
54	60,91
72	66,98

Table 5 contains strain values obtained at limit load from selected strain gauge rosettes. Strain values below 400 microstrain were not included in the Table. The highest values of strain were obtained from those gauges applied to the spars. A high shear strain value was also obtained from the rosette R_1, which had been bonded on to the surface of upper skin, roughly in the middle. The principal strains were well below the design value of 4000 microstrain, and the shear strains were also quite low.

The positions of the strain gauge rosettes on the tailplane are shown in Figure 8.

Table 5 : Values of Strain at Limit Load

	Strain at Limit Load, microstrain		
Strain Gauge Rosette	24°C	54°C	72°C
L_1 (single)	2547	2764	2921
L_2 (single)	-985	-1115	-1198
L_3 (single)	2138	2396	2585
L_4 (single)	2204	2621	2962
L_5 (single)	1723	2012	2257
R_1 (0/45/90)	3973 (shear)	3862 (shear)	4364 (shear)
R_2 (0/45/90)	505 (shear)	492 (shear)	435 (shear)
R_3 (0/45/90)	697 (shear)	765 (shear)	783 (shear)
R_{10} (0/45/90)	1314 (shear)	1490 (shear)	1652 (shear)
R_{14} (0/45/90)	1696 (shear	1777 (shear)	1887 (shear)
R_{17} (0/90)	1210 (90)	1172(90)	1213(90)
R_{19} (0/90)	1080 (90)	979(90)	1159(90)

Non-Destructive Testing

Two techniques, X-radiography and transient thermography, were used in an attempt to monitor the propagation of the defects. Although identification of the defects was moderately successful, the resolution was not good enough to measure dimensions accurately. Propagation of the defects could be assessed only qualitatively.

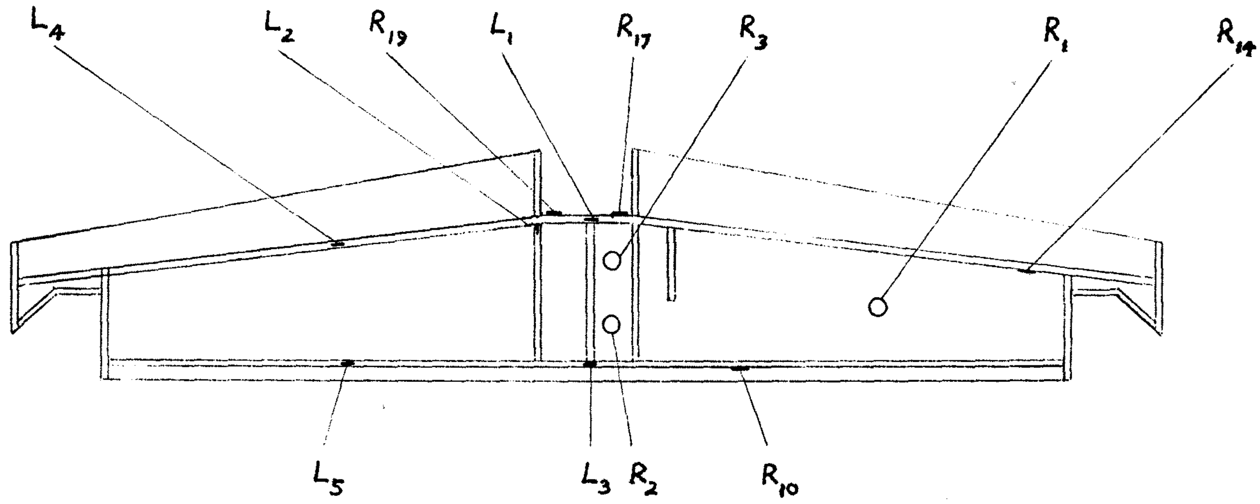

Figure 8 : Positions of Strain Gauge Rosettes on the Tailplane

Conclusions

From the tests conducted to date the tailplane has withstood the static and fatigue loads applied to it. The principal strains measured have been below 4000 microstrain and the deflections have not been excessive. From the limited non-destructive evaluation carried out so far it appears that the propagation of the built-in defects and the impact damage is not significant. It is still too early to predict whether the tailplane will survive the full testing programme but the benefits gained from tackling the project have been considerable.

References

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