FURTHER FATIGUE TESTING OF A GLASS FIBER REINFORCED PLASTIC GLIDER WING

by C.A.Patching and L.A.Wood, The Sir Lawrence Wackett Centre for Aerospace Design Technology, Royal Melbourne Institute of Technology, Melbourne, Australia.

Presented to the XXV OSTIV Congress, St Auban, France, July, 1997.

Summary

Fatigue testing of a Janus Glass Fiber Reinforced Plastic (GFRP) wing was terminated after a total of 35,482 hours of simulated Australian service conditions. A load spectrum was developed from measurements made using a number of gliders from which a randomized set of load cycles was derived and applied during the test. The test article consisted of a new port wing, and a partly repaired crash-damaged starboard wing. Repairs rectifying failures and major structural damage were carried out at various stages of the test. Results have confirmed the efficiency of repairs and indicated no significant change in structural integrity except in areas of concentrated load transfer.

1. Introduction

The results obtained during the fatigue testing of a complete Janus glider wing for 22,000 simulated flying hours were reported in Ref. 1. Testing was continued for a further 14,000 hours, and during this period there was a major failure in the starboard root rib. Actual testing of the wing commenced in May, 1986 and was terminated in November, 1993 when, following a failure of the testing rig, there was insufficient finance available to effect repairs and continue the program. Full details of the test article together with all damage found and repaired is contained in Reference. The acquisition of data for service loadings, conduct of the fatigue test, repairs made before and during testing, and vibration measurements have been reported in Ref. 3. This report summarizes the growth of damage left in the previously damaged right wing, test induced damage, performance of repairs, inspection techniques, and comments on continued airworthiness.

2. Test Loading Sequence

The description of the loading sequence given in Ref. 1 has been revised to read as follows. A simple 6 load range i.e.. a block of 12 load levels, based on the Dorning spectrum (see Ref. 1) was used for commissioning the test rig. The Dorning spectrum had been truncated at the upper level where loads occur once in 6000 hours and the lower level at 50 loads per hour. The block simulated 294 flying hours and contained 29,404 turning points at each load level. For the actual fatigue test these turning points were randomized and grouped into 14 blocks (termed flights). The end of each flight was marked by the insertion of a 1g. load which was held long enough for the computer to print out one line of data. An adjustment was made to ensure that two of the flights contained sufficient high loads to represent an aerobatic mission and included a 5.3g load. The number of turning points at each load level excluding the 1g. level are given in Table 1. These were shared equally between the load levels, and were automatically applied using a computer. The design limit loads of +6.0g and -2.5g were applied manually at the end of each 6000 hours.

Turn Point Level g		Number
Min.	Max.	
0.5	2.3	27,520
0.2	2.9	1,560
0.0	3.5	236
- 0.3	4.1	64
- 0.7	4.7	20
- 1.2	5.3	4
	Total	29,404

Table 1. Number of turning points in a 294 hour block.

3. STRAIN MEASUREMENTS

The test article was fitted with a total of 332 electric resistance strain gauges (ersg's) with the majority on the port wing. Eight temperature gauges were also fitted in the structure which showed little difference from the ambient value. Output from the ersg's was used for three purposes: a) Design of the loading system b) Determination of strain distribution in the structure and c) Monitoring of strain/g as the test progressed. The use of strain data for item (a) was described in Ref. 1.

Non-linear behavior was observed from gauges in the 0° direction (spanwise) in rosette gauges, indicative of gauges monitoring a matrix dominant area. Otherwise a linear load-strain relationship was maintained. The ±45° gauges in the rosette arrangement on the spar shear web showed equal strain magnitudes but of opposite signs. Those on the wing skin did not exhibit the same characteristic because the wing skin did not experience pure shear, being largely affected by the bending loads. During a strain survey, readings were collected from 332 gauges at 17 load levels, between 0g,4.5g,-1.2g and returning to 0g. The data was then processed to check the quality of data and to eliminate spurious data. The strain/g was then determined for each load increment and the mean value used for further analysis. Figure 1 shows typical values for gauges in a spanwise direction on both upper and lower surfaces at seven chordwise locations. A periodic examination of the strain gauge data indicated that there was no significant change in the measured strain per g. Many gauge failures (over 100) occurred through problems associated with their installation and repairs made to the wing. Fatigue failure of gauges was identified by change in the gauge resistance at zero load. Gauges located in high strain areas performed well beyond the manufacturer's predicted life.

4. PRE-TEST DAMAGE

The starboard wing had been extensively damaged in the accident and repairs involved the removal of large areas of skin from both the upper and lower surfaces adjacent to the root rib to facilitate both inspection and to be able to inflict further damage. Inspection holes were also cut in the upper and lower surfaces. Holes in the upper surface were filled prior to test and those on the lower surface fitted with reinforcing rings. Not all of the damage was repaired. Some damage continued to grow necessitating repairs early in the test as reported in Reference 1. A large easily-detected narrow section of delamination in the lower surface extending from Sta.3250 to 5300 just forward of the main spar was monitored. There was only marginal growth outboard from this defect, and the strain in this region was in the order of 730 microstrain/g. A chordwise crack 130mm long was left in the inner skin of the wing upper surface at Sta. 2200. This crack grew at the rate of

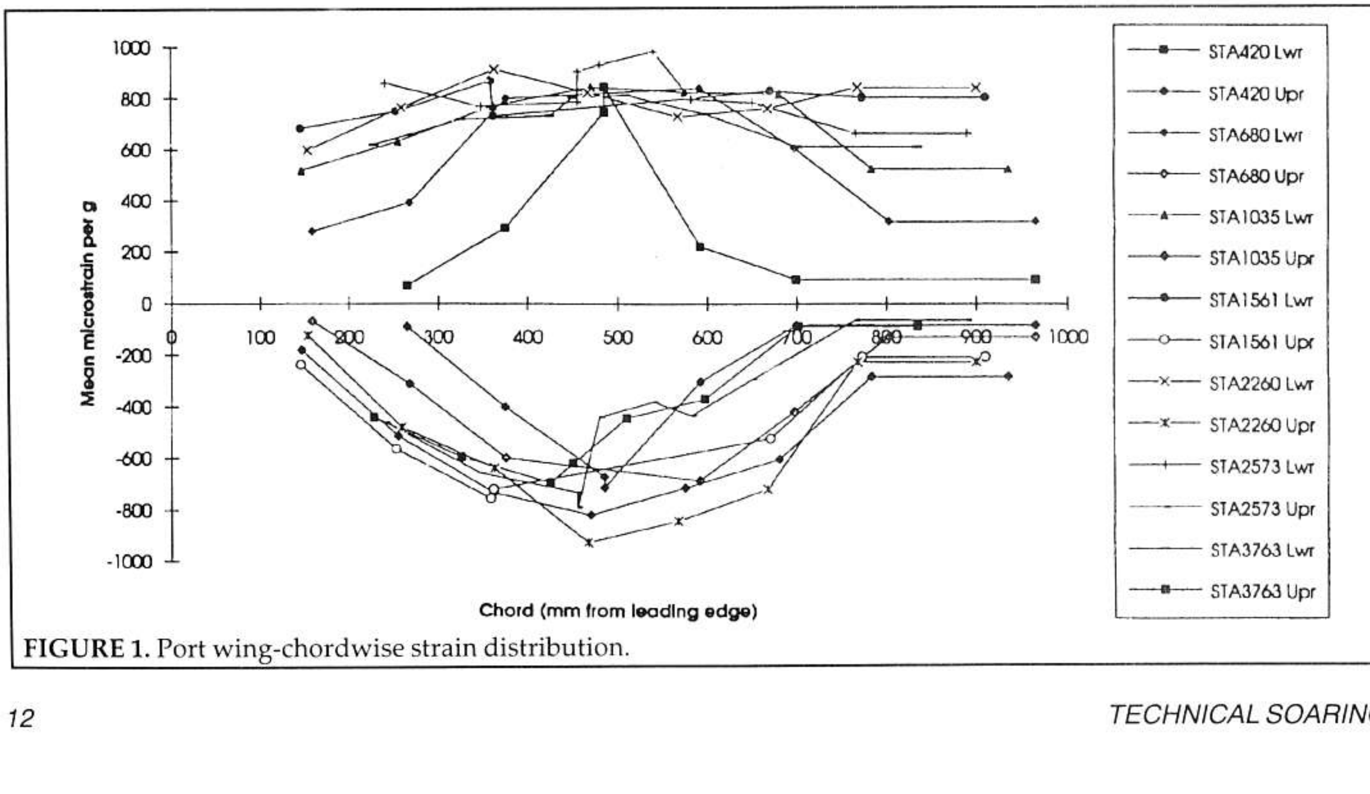

FIGURE 1. Port wing-chordwise strain distribution.

about 6mm per 1000 hours, mainly in a forward direction, until 16,000 hours when it stopped growing. However, growth commenced again at 35,227 hours in an aft direction, but this was probably induced by the external crack at Sta. 2500, see below. The strain/g varied from 900 microstrain at the forward end down to 200 microstrain at the aft end. An area of delamination between the upper skin and the spar boom left unrepaired at Sta.7000 did not grow sufficiently during the test to warrant repair. The skin was also showing obvious signs of delamination both fore and aft of the spar at this location.

5. DELIBERATE DAMAGE

As described in Reference 1 there had been deliberate damage done to the wing upper spar boom 600mm outboard of the root rib, and repaired using a splice angle in the rovings of 40 to 1. Holes had also been punctured in the main spar shear web and repaired using a standard repair technique. The installation of 14 strain gauges on the main spar rovings required removal of the skin on both the upper and lower surfaces of the port wing. These areas of skin were replaced using splice angles in the order of 10 to 1.

6. INADVERTENT DAMAGE

As reported in Reference 1 the wing was accidentally overheated on the upper surface in the region of Sta.4650. This resulted in a buckling instability of the skin and compression failure of the upper spar boom. During scarfing a number of voids approximately 2mm x 50mm were revealed. There was no evidence of growth since manufacture. Following replacement of the starboard root rib at 28,346.6 hours there was an increase in clearance between the rib and the dummy fuselage at the shear pins. The clearance at the front pins was a total of 5.5mm, and at the rear 10.5mm instead of 4mm. During cycling the wing was able to move thus producing a variation of clearances at each pin. The resulting fore and aft motion coupled with the up and down motion from the wing bending unscrewed the port front pin from the root rib. The pin was removed at 29,464 hours, and after cleaning was replaced using Loctite on the thread. A 5mm spacer was fitted on the starboard rear fuselage pin at 29,504 hours, between the fuselage and the shoulder of the pin. All four pins were tight in the ribs when the test stopped.

7. FATIGUE TEST DAMAGE

A full description of all the damage that appeared during the test is given in Reference 2 and only damage considered to be important is briefly described below. The terms minor and major have been used to classify damage as it affected the fatigue test, and minor may be considered as being major by some operators.

7.1 Minor Skin Cracking

A crack 30mm long was found in the starboard upper outer skin at 4,122 hours, running chordwise at Sta.2,500 in an aft direction. This was regarded as being caused by the installation of a strain gauge on the spar boom. The crack grew more or less equally in both directions at a total rate of about 4mm/1000 hours until 9,825 hours, after which growth was only in an aft direction. The rate of growth was only 1.2mm/1000 hours until 31,000 hours, when it increased, reaching a rate of 35mm for the hour prior to the test stopping. At this stage it had a total length of 150mm. The crack that had been left in the internal skin as described in Section 4 had started to grow again and had extended 5mm in the last 1,000 hours of testing. From Figure 1 it can be seen that the spanwise strain was almost constant over the length of the crack. Had testing been continued it is possible that these two cracks would have lead to failure of the foam sandwich, with a need to repair.

7.2 Minor Cracking in rear spar

At 32,150 hours cracks were found at both lower corners of the cut-out for the aileron push rod in the shear web of the starboard rear spar. When testing ceased at 35,482 hours there was no indication of the cracks having propagated into the lower flange of the rear spar. The new port wing did not contain this cut out.

7.3 Minor Delamination of upper skin

An area of delamination was detected between the starboard upper skin and the main spar boom at 35,482 hours. The delamination extended from Sta.4,480 to 4,590 and was about 8mm in width. Figure 1 shows that the strain in the upper spar boom was approximately 600 microstrain per g.

7.4 Major Root Rib failures

The first failures in the starboard root rib at the two spigot bearings were reported in Reference 1. During manufacture a filled resin was used in the lay-up near the bearing housings, producing a whitened region. For the first repair, damaged material was removed from the outer surface and replaced with glass cloth as advised by the manufacturer (Repair Scheme No. 1). The next two repairs consisted of layers of cloth interspersed with rovings to assist transfer of load from the bearing into the rib (Repair Scheme No. 2). A revision of the calculated shear load transmitted into the rib through each bearing showed that it was 8.5kN/g, and not 7.1kN/g as previously reported in Reference 1. The bearing housings had an outer diameter of 40mm and a width of 18mm, giving a bearing stress of 10.6daN/mm^2 at 9g ultimate load. The Schempp-Hirth design value at ultimate load was 8.6daN/mm^2. At 23,379 hours, with a repair life of 8,281 hours, an area of micro cracking was found in the inboard face of the rib at the rear bearing. The area was 11mm long with a maximum width of 4mm at the 5 o'clock position. This area did not increase in size or intensity. At the same time, the front housing with a repair life of 4,636 hours exhibited a larger whitened area, 14mm long and 2mm wide, extending from the 9 to 12 o'clock position. This area continued to grow and extended to the 6 o'clock position with a maximum width of 6mm by 28,150 hours (at its repair life of 9,407 hours). However, the intensity had not increased at these whitened areas. At 28,346.6 hours there was a catastrophic failure at the starboard root rib upon application of a 5.3g load. Since the repair at 15,098 hours, a 6.0g load had been applied (21,000 hours), followed by a total of 50 applications of 5.3g. Failure occurred simultaneously at both the front and rear spar spigot bearings and appeared to be a combined mode of compression, bearing and tear-out. Both bearings had separated from their housings and remained tightly fitted on the spar spigots, while the housings were forced outboard through the rib into the wing box. The damaged areas from the root rib along with the bearing housings were subjected to examination at the Aeronautical and Maritime Research Laboratory (AMRL) by Hill and Pell (4).

Their significant findings can be summarized as follows:

1. There was strong evidence that both bearings had been moving in their housings prior to the collapse. The front housing was worn and the bearing had become tilted in the housing.
2. Hardness and metallographic examinations by Hill and Pell (5) revealed that, although not required, the rear housing had been heat-treated while the front replacement housing had not.
3. The bond between each housing and rib had failed as evidenced by the presence of a 'fretting' compound at the interface.
4. The amount of delamination in the rib material was minimal and the fracture at each housing had followed a line from each surface of the rib at about 45° to the axis of the housing as shown in Figure 2 around the lower half of the housing.
5. Both the original rib composite and repair material were of uniform hardness, indicating a satisfactory heat treatment (tempering) following the repair.
6. The detection of delamination and its growth during testing, together with other evidence including rubbing damage to the fracture surfaces, all indicated that the failure mode was fatigue. A possible reason for the limited damage growth was that the rovings introduced in Repair Scheme No. 2 had constrained the propagation of delamination.

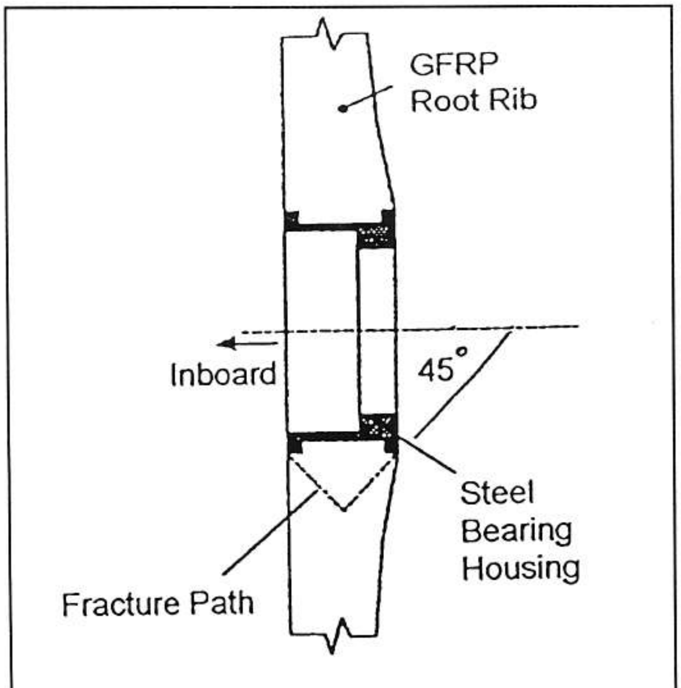

FIGURE 2. Fracture path in the starboard root rib.

Rib construction status	Front bearing Initial†	Front bearing Final‡	Rear Bearing Initial	Rear Bearing Final
	(simulated flying hours)			
Original Manufacture	Not available	18743	11138	13862
Repair Scheme 1	-	-	-	1236
Repair Scheme 2	4636	9603	828	13248
1992 Replacement	-	7135+	6881	7135+

+testing incomplete

†First visual sign of cracking/crazing.

‡Structural component needed repair and the test was stopped.

Table 2. Root rib test lives at the starboard wing half.

The front bearing when tilted introduced a more severe loading condition in the root rib, subsequently it was likely that it failed first, closely followed by the rear bearing as the load was transferred. The fatigue test lives for the first reported detection of failure in the root rib at each of the bearings and when repairs were made are given in Table 2.

The wing was repaired by completely removing the damaged rib and replacing it with one manufactured during 1992 by Schempp-Hirth. The wing was taken from the test rig for this repair. A separate investigation was conducted into the fatigue life of the root rib material when subjected to bearing loads. The preliminary results of these tests given in Reference 6, indicated that bearing failure due to fatigue load occurred after a similar number of simulated flying hours to those which caused failure in the full scale fatigue test.

8. Gel coat cracking

There was no propagation of the cracks induced in the gel coat by the crash, nor in the single stress induced crack detected at 11,138 hours in the lower surface of the port wing at Sta.4650. This crack had not shown any growth after 7,645 hours of testing when it was removed. Cracks appeared in the gel coat applied following repair of the port wing upper surface and these also did not show any growth after being detected. The absence of ultraviolet light and moisture in the laboratory environment has had a significant effect on the behavior of cracks in the gel coat.

9. Stiffness Measurements

The bending and torsional stiffnesses were measured periodically at each wing tip and showed no significant change during the test.

10. Vibration Mode Measurement

A total of seven vibration model analyses were performed at roughly 6,000 hour intervals. The first before any fatigue load cycling commenced and the last at 28,346 hours. The measurements made are contained in Reference 3, along with results from a ground vibration test on the flight test Janus glider. All the test wing measurements were performed using a single shaker with random excitation. The anti-symmetric modes, initially measured at 1.42Hz and 6.48Hz displayed a trend of reducing frequency. The symmetric modes at 2.33Hz and 8.16Hz showed very little change of frequency and damping.

11. Spar Spigots

The steel spar spigots were inspected three times during the test by AMRL staff using an ultra sonic technique, since failure of a spigot would lead to a catastrophic failure of the wing.

There were no indications of any cracking in either the front or rear spigots when testing stopped at 35,482 hours. However, the design of the Janus spigots does not include the stress concentrations or welding features present in some other designs that have been found cracked both during test and in service.

12. Discussion of Results

The Janus wing was selected as an appropriate test article because it contained structural features representative of a large proportion of the GFRP glider population. Furthermore, the design stress levels were reasonably high. The actual Janus flight tests showed the strain/g to be lower than that predicted by design. However, the test values of strain/g, load magnitude and spanwise load distribution, while based on design values, are all more severe. The applied test load was required to be 25.6% larger than the design value in order to induce the design value of 300 MPa in the upper spar boom rovings at the ultimate load of 9g. A value of 35 GPa for Young's modulus was used to calculate stress in the spar boom roving material.

12.1 Test Failures

Prior to the test a number of areas had been identified as warranting attention during inspections. However, the only significant failures occurred at points of high load transfer from steel fittings to the GFRP material. The starboard root rib failed four times in the area around the bearings for the spar spigots that transmit the wing bending moment. In each instance failure was preceded by crazing and cracking. The first repair at the rear bearing did not remove enough of the damaged material thus requiring further repair after 1,236 hours. The changed repair scheme produced test lives comparable with those of the original rib. However, the mode of failure changed from crazing and delamination to cracking from the junction of the steel and GFRP at the outer edges of the housing, progressing into the rib from both faces. This resulted in a smaller area of visible damage on the surface of the rib. Examination of the failures indicated that passing a strong light through the rib would have revealed a larger area of damage.

12.2 Performance of repairs

The repairs were deliberately done by persons simulating skill and experience ranging from high to low in order to not only follow what happens in practice, but to also investigate the efficiency of various repair schemes. Circumstances dictated working conditions which could be classed as minimal to adequate for the repair of composite structures. Many of the repairs embodied non-standard procedures for various reasons, such as not being described in existing repair manuals and space limitations restricting the length of scarf joints. Scarf angles as steep as 1 in 10 were used in a number of cases. The repair following collapse from overheating involved removal of only those areas with visible damage.

During the test, apart from the first root rib repair, there were no failures detected in any of the other repairs done prior to or during the test.

12.3 Growth of damage

Apart from the growth of some obvious damage that had been left in the wing, which was repaired at 600 hours, there was either no growth or insufficient growth to warrant repair of other prior damage. Manufacturing voids found in the spar boom rovings during the repair at 18,783 hours did not exhibit any signs of growth. Fatigue cracks found in both inner and outer skins during the test were monitored and had not been repaired when testing ceased at 35,482 hours. However, the growth rate, if any, for the delamination of skin from the upper spar boom at Sta. 4500 could not be determined (See Section 7.3).

12.4 Modal analysis

The vibration model analysis showed no significant changes in relation to the natural frequency, damping and mode shapes of the wing. The anti-symmetric modes did display a trend of reducing frequency, but these included rotation of the dummy fuselage at a central pin, however, there was no rotation in the case of the symmetric modes. Hence wear or changes in stiction would influence the anti-symmetric modes. The consistency of this data along with the strain and stiffness measurements, indicated that neither the repairs nor the damage present had any significant effect.

12.5 Inspection techniques

All progressive damage in composite material caused by the fatigue loading was found using simple non-destructive inspection techniques. The transparency, smoothness and simplicity of the GFRP wing structure enabled the efficient use of strong light and tapping as inspection tools. However, for the reliable inspection of certain areas the creation of inspection access holes was essential.

13. Conclusions

The fatigue testing of the Janus wing for 35,482 hours has produced information on a number of items pertinent to the continuing airworthiness of GFRP glider structures.

1. The fatigue life of glass rovings in the spar boom, at a strain level of 1000 microstrain/g, is in excess of 36,000 hours of the applied load spectrum.
2. The root rib of the test wing failed before the wing spar had shown any sign of damage. The data indicate that the fatigue life to failure of the rib, with a bearing stress of 10.6 daN/mm^2 at 9g ultimate load, would be in the order of 18,000 hours. However, visible signs of crazing and cracking preceded failure.
3. Results from using various repair techniques on the rib indicated that lay-ups using cloths without rovings were more damage tolerant than those used with rovings.
4. Standard repair techniques were validated.
5. Visual inspection proved to be an effective means of detecting and monitoring structural damage in the composite structure. The damage modes in composites were observed to be different to those in metals.
6. When considering that vibration data measurements

in the field are not done with the aid of sophisticated instrumentation, and that only the fundamental wing bending mode (first symmetric bending) is measured, it can be concluded that there is very little prospect of vibration data providing a unique indication of structural damage. This conclusion supports the widely held view in vibration analysis that frequency is very insensitive to structural stiffness changes.

7. The GFRP structure proved to be damage tolerant with both slow crack growth and slow rate of delamination. The rate of delamination was much slower in the tension (lower) surface than in the compression (upper) surface.

8. The current inspection interval (in most countries) of 1,000 hours has been confirmed as being adequate.

14. Acknowledgments

This research project starting from its initial phase on flight test to its final phase on the documentation of the project, involved a large team effort for a period of 14 years. Furthermore, the project was conducted with limited financial support. The continuation of the program was only made possible by contributions from the following organizations whose input was as follows:

Civil Aviation Authority (CASA), Airworthiness Branch: finance, inspection, equipment, management and technical support.

Aeronautical and Maritime Research Laboratory (AMRL): equipment and technical support.

Gliding Federation of Australia (GFA): finance, management and technical support.

Royal Melbourne institute of Technology (RMIT): finance, flight test glider, laboratory facilities, management and technical support.

15. References

1. Patching, C.A, and Wood L.A, 'Fatigue testing of a GFRP Wing' *Technical Soaring* Vol. 15 No.4 October, 1991.
2. Patching, C.A, Fatigue testing of the Janus glider wing Technical Report TR 95/01, RMIT March, 1995.
3. Patching, C.A, and Loh, A., 'GFRP Glider project-Final report' Technical Report TR 96/04, RMIT November, 1996.
4. Hill, T.G, and Pell, R.A, 'GFRP wing fatigue test-Post failure analysis' AMRL Defect Assessment and Failure Analysis Report No. M105-2/g2, August, 1992.
5. Hill, T.G, and Pell, R.A, Janus wing-spigot bearings AMRL Defect Assessment and Failure Analysis Report No. M105/91, March, 1992.
6. Loh, A, Herszberg, I, and Hill, T.G, 'Fatigue of glass fibre reinforced plastic subject to bearing loads' First Australian Congress on Applied Mechanics, February, 1996.