FATIGUE TESTING OF A GFRP GLIDER

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Presented at the XXII OSTIV Congress, Uvalde, Texas, U.S.A. (1991)

1. INTRODUCTION

Most gliders operating in Australia and in many other countries, have been designed to airworthiness requirements (Refs. 1 and 2) which, regarding fatigue life, merely state: "the structure shall be designed to avoid points of stress concentration where variable stresses above the fatigue limit are likely to occur in normal service."

Sailplanes manufactured in Germany using glass fiber reinforced construction (GFRP) are required to satisfy a standard (Ref. 3) which has enabled them to be certified for a service life of 3,000 hours. The life has now been extended to 6,000 hours subject to satisfactory inspections at 1,000 hour intervals beyond 3,000 hours. This life extension has been justified using data from fatigue tests on box beam structures. However, this method of justification has been open to question as the construction and test did not closely resemble sailplane conditions. In addition, the use of a scatter factor of three on test life, commonly applied for metal structures, must be viewed with some caution. Recently, some gliders have received an extension to 9,000 hours, but still require inspections every 1,000 hours after 3,000 hours.

A full scale fatigue test has been reported for a braced all-composite wing (Ref. 4), but the spar caps for this structure were carbon/viny-lester pultrusions. The authors reported that after 25,000 equivalent flying hours, the structure did not exhibit any changes. The published report did not explicitly state whether or not damage and repairs were evaluated in the test.

All other reported tests have stopped at either 9,000 or 18,000 hours. Helicopter manufacturers undoubtedly have fatigue data on GFRP constructions since many rotor blades and hubs use this composite as the main structural material. However, the data are not available.

The utilization of sailplanes in Australia is high compared with other countries, and statistics show that many will reach their safe life before the type is superseded (Ref. 5). In 1981 the decision was taken to embark on a program aimed at substantiating the fatigue life of GFRP sailplanes beyond the current 3,000 hour limit. The investigation was initially funded by the participating partners: Royal Melbourne Institute of Technology (RMIT), Australian Civil Aviation Authority (CAA), and Gliding Federation of Australia (GFA). The Aeronautical Research Laboratory (ARL) has provided considerable technical assistance at various stages of the program.

2. FIBER GLASS FATIGUE INVESTIGATION

The investigation was designed to establish by analysis and substantiating tests the fatigue certification of fiberglass gliders to an optimum economic service life (Refs. 6 and 7), and included the following:

(a) flight investigations using the instrumented RMIT Janus glider for strain measurements, and a number of other types of gliders for g load exceedences during typical operations in Australia,

(b) static and fatigue testing of fiberglass specimens to provide basic fatigue data,

(c) development of a finite element mathematical model of the wing to enable the Janus results to be applied to other types of gliders, and

(d) full scale fatigue testing of both a new and a repaired Janus wing, which is the subject of this report.

3. FATIGUE TEST PROGRAM

3.1 Data Acquisition

The RMIT instrumented Janus 'B' sailplane was used to establish a correlation between acceleration at the aircraft center of gravity and wing bending strains, and also to obtain load sequence data. The sailplane was flown by experienced competition pilots during seven competitive events for 164 hours, and by student pilots during training exercises. Flight strains were also determined for aerobatic maneuvers.

3.2 Data Reduction

The strain data from the aircraft were found to correlate with the c.g. acceleration, but were 25% less than the design calculations predicted. These are depicted in Figure 1 being converted to stress using E = 35 GPa. The spanwise strain distribution reduced in magnitude towards the tip more rapidly than predicted, possibly because of a variation of lift distribution and/or conservative manufacturing techniques. Fatigue meter data obtained from other gliders flying in Australia have been collated to deduce an overall g spectrum for Australian conditions (the "Dorning Spectrum"). Those for the Janus and IS-28 gliders, along with the Franzmeyer and Thielemann test spectra are shown in Figure 2.

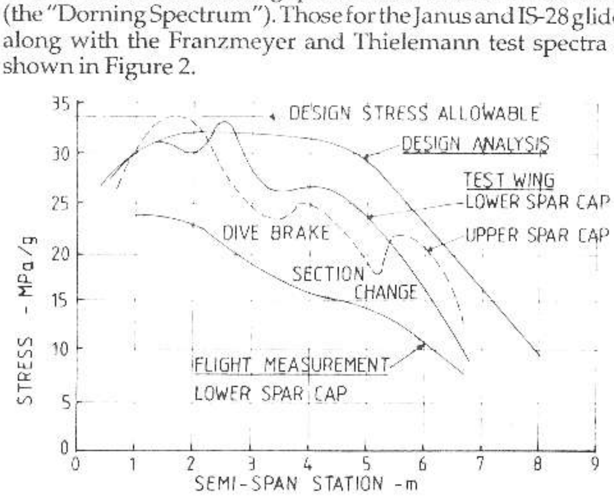

FIGURE 1. Comparison of stress/g in main spar booms.

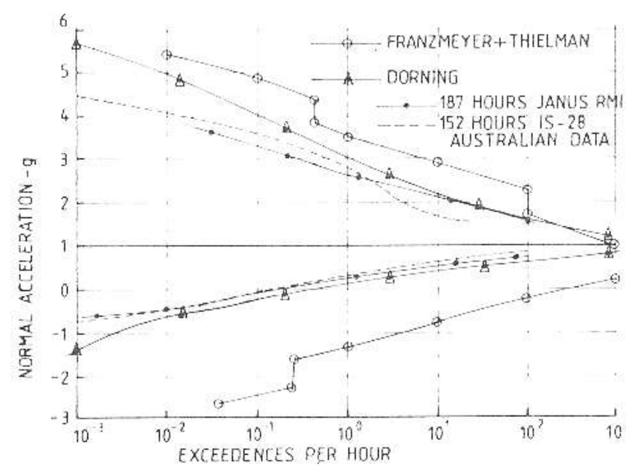

FIGURE 2. Comparison of glider load spectra.

3.3 Fatigue Test spectrum

It was originally intended to create a flight-by-flight sequence of test loads representing a typical set of flying missions, including landing and taxiing. Instead, a simple six load range (i.e. 12 load level block) program, randomized to represent the Dorning spectrum, has been adopted. This load program represents 15 flights totaling 294 hours, and contains 29,434 turning points, which are automatically applied. Each flight begins and ends with a +0.5g load. Maximum loads applied automatically are 5.3g and -1.2g. The extreme loads of +6g and -2.4g which occur with a frequency of 1 in 6,000 hours are applied manually at the end of each 6,000 hours.

4. Test Article

The GFA supplied a crash-damaged but repaired Janus 'B' starboard wing incorporating all types of repairs. The repair schemes used were typical of workshop repairs and "field repairs." Some minor damage was left to be monitored for growth and later repair during the test program.

The RMIT purchased a new left wing from the Schempp-Hirth factory in Germany. This wing was fitted with electrical resistance strain gauges (ersg's) to the internal structure during manufacture. The wing was in the "as removed from the mould" condition except for minor trimming operations. No control surfaces or systems, apart from hinges, are fitted to either wing. A total of 300 ersg's have been installed on both wings with the majority being on the port wing. Eight temperature sensors were also embedded in the structure.

5. Fatigue Test Rig Design

The test article is fastened to a center section mounting structure which is attached to two portal frames. Loads are applied symmetrically to the wing set by a whiffle tree system which loads each wing at eight loading stations across the semi-span. A single double-acting jack, controlled by a closed-loop hydraulic servo system, provides the actuation forces to each side of the whiffle tree by means of cables. both upwards and downwards loads may be applied. The weight of the rig has been adjusted to account for all except the manually applied load of -2.4g. Weights have been attached to provide the torque distribution. The wing is restrained by a dummy fuselage which is free to roll at the main spar, but fixed at the rear spar. Protection devices are incorporated in the rig to lock the jack in the event of a malfunction; the jack is then manually unloaded.

A computerized procedure has been employed to estimate the aerodynamic load distribution at various air speeds and flap angles. The wing weight distribution has been estimated in spanwise and chordwise directions. A single spanwise load distribution has been shown to approximate closely the loading for all the relevant flight conditions. The loading corresponds to a flight speed of 160 km/hr with four degrees flap at an all-up aircraft mass of 620 kg. The load per g corresponds to the ultimate design stress of 300 MPa. Wing torque is constant for all values of g.

Calibration of the test rig showed that the span stresses distribution differed from both that measured in flight and predicted during design. The design calculations showed the stress in the spar booms to be constant for about 40% of the span, with the flight stresses being generally lower, especially toward the tip. The maximum strains measured in the test rig were similar to those measured in flight i.e. less than the design values. A decision was taken to alter the loading distribution and to increase the loads to give the design stress of 300 MPa at the ultimate load of 9g, which required an increase of 25.6% in the original design test load per g. This produces 950 micro strain per g in the rovings of the upper spar cap.

6. THE JANUS 'B' WING STRUCTURE

The wing is mid-plane mounted, having four degrees of dihedral and two degrees of forward sweep at the leading edge. The airfoil section varies from a Wortmann FX-67-K-170 (17% t/c) at the root, to a FX-67-K-150 (15% t/c) at the tip. The wing span is 18.20m and the aspect ratio is 19.95.

The wing box skin and spar web construction is exclusively woven glass fiber/epoxy resin/foam sandwich. The wing has no ribs. At inboard sections, the spar has a box section, while outboard of wing station Y-5470 mm the twin sandwich shear webs change to a single web 'I' beam construction. The spar caps are monolithic uni-directional E-glass/epoxy rovings. The exterior surfaces are finished with a polyester gel coat of typical thickness 0.4mm. Details of materials employed in the wing construction are as follows:

Spar Rovings:
Gevetex E-glass type ES 10-40 x 60 K43.

Box Fabrics:
Interglass E-glass styles 92125 crosstwill (283 g/m^2) and 92415 unidirectional (215 g/m^2) with volan or I-550 finish
Epoxy System:
Shell Epikote 162 resin and BASF Laromin C260 hardener
Sandwich Core:
Continental PVC rigid foam Conticell 60,7.9 m thick (53 g/m^3)
Gel Coat:
Lesonal-Werke white polyester Lack-vorgelat and hardener.

7. INSPECTION AND MONITORING OF THE TEST ARTICLE

Detailed inspections are made every 1,000 hours similar to the GFA Annual Inspection. Inspections in accordance with the manufacturer's schedule are made at the same time. Inspection techniques employed include tapping by coin and hammer, fibroscope internal inspection, visual external inspection, radiographic, acoustic methods and modal analysis. Deflection exceedence counters are monitored to ensure the program disc is performing and to alert the operator to major changes in the wing properties. the bending and torsion stiffness of the wing and the strain/g from selected gauges are regularly monitored.

There are three regions of the wing which merit special attention during inspections. The wing spars spanning the fuselage, root ribs and shear attachments clearly comprise the most highly loaded region. The other two regions contain structural discontinuities, namely the dive brake box assembly and the main spar section change. Apart from existing water ballast holes and some added to facilitate inspections, the remainder of the wing structure is relatively simple and continuous.

8. STATUS OF FATIGUE TEST RESULTS

The full scale fatigue test is continuing, and at the time of writing, over 22,000 equivalent flying hours have been accumulated. there has been no significant damage accumulated on the new wing, although there has been substantial growth of minor damage from unrepaired sections and several "field repairs." The unrepaired damage was retained to simulate undetected damage of the type that could occur in the field, for example, cracking of the inner surface of the wing skin.

There has been no significant change in the measured strain/g values throughout the duration of the test. Some 100 gauges have become unserviceable through problems associated with installation and repair. As yet there have been no gauge fatigue failures in the foil grid of the gauges. Bending and torsional stiffness have also shown no evidence of changes for either wing. Vibration modal analysis has revealed no changes in natural frequency, damping, or mode shape.

The test is not environmentally controlled, which means that the test article has not been subject to the range of temperatures, humidity and ultraviolet light to which a sailplane is normally exposed. Hence there has been no cracking or crazing of the gel coat. However, at 19,000 hours some small cracks appeared in the upper surface of the "new" port wing.

8.1 Repaired Wing — Delamination of Upper Skin from Unrepaired Damage

Cracking of the inner surface of the upper skin on the salvaged wing was left unrepaired to simulate undetected damage. For this type of structure, the term "cracking" is used to describe a crack in the resin, even when the fibers at the crack are intact. The damage was not detectable from the exterior, either visually in the form of a crack, or as a delamination emanating from the crack detectable by coin tapping on the upper surface. It was located just outboard of the dive brake box (station 3735 to 3970).

The crack initiated delamination between the inner and outer surfaces, and grew rapidly in the first 600 hours. The test was stopped for repairs at 613 hours. Repairs consisted of the removal of the entire damaged skin (235 mm spanwise x 295 mm chordwise), and replacement with a new skin segment. The strain in the upper spar boom at the damaged section is determined to be 770 microstrain/g.

8.2 Repaired Wing — Delamination of Lower Skin Near the Rear Spar

A small area of skin delamination extending to the rear spar resulted in skin separation from the rear spar. This also occurred in the first 600 hours near the dive brake box (station 3500 to 3970), and was similarly repaired using insert panels. It is suspected (but unconfirmed) that there was genuine undetected damage resulting from the crash, which propagated when the previously mentioned failure developed.

8.3 Repaired Wing — Unrepaired spanwise Delamination on Lower Surface

A narrow section of delamination extending from station 3200 to 4200 just forward of the main spar was caused by the crash. The defect was easily detectable by coin tapping. Throughout the test there has been only marginal growth at the defect's outer spanwise end, indicating that the tension surface is not susceptible to delamination.

8.4 Repaired Wing — Deliberate Damage to Upper Spar Boom

During the crash no major damage was inflicted on the main spar, so deliberate damage was introduced to similar impact with a steel fence post, damage which could occur during outlandings. A sharp-edged steel tool was used to generate a notch shape depression on the spar cap. The impact was primarily on the upper forward edge of the spar cap. At the forward edge the notch was full depth of the spar cap, reducing to minimal depth at the rear edge. The type of damage inflicted ensured that a substantial proportion of the uni-directional fibers in the spar cap were severed. Repairs were carried out by removing a wedge-shaped segment of material on either side of the notch (total length of repair 1,200 mm), and replacement with new fibers and resin. This repair has been closely monitored throughout the test, and there has been no evidence of fatigue damage occurring at the repair.

8.5 Repaired Wing — Movement of Bearing in Root Rib

Flight loads are transmitted through the wing center section by a fork and tongue arrangement which is designed to enable quick assembly. A spigot on the rear fork of the port (new) wing transmits shear loads to a bearing located in the root rib of the starboard (repaired) wing. The shear load transmitted into the rib through this bearing is calculated to be 7.1 kN/g.

Radial cracks extending through the whitened region were detected at 11,138 hours. At 13,040 hours, delamination occurred in the same region, and at 13,862 hours the test was stopped and a repair made to the damaged root rib. The damaged material on the root rib was removed by grinding, and replaced with glass cloth as instructed by the manufacturer.

At 15,098 hours, cracking was again detected around the bearing housing. It withstood a 5g loading, but there was obvious movement between the housing and the rib necessitating further repairs. The bearing housing was removed, and rib material was removed by grinding on the inner and outer surfaces. The repair consisted of layers of cloth interspersed with rovings to transfer loads from the bearing into the rib. At 18,000 hours, cracking and crazing around the front bearing housing had reached an advanced stage. The bearing was removed and a similar repair scheme effected.

8.6 Repaired Wing — Delamination of Lower Skin Near Aileron Fitting

Separation of the outer surface from the foam core occurred at station 6500 at an aileron push rod guide which is attached to the inner surface of the lower skin. It is suspected (but unconfirmed) that the damage initiated during the crash due to unusual loads in the aileron push rod. The growth of delamination necessitated repairs after about 500 hours. A simple repair technique of injecting resin was employed, and no further delamination has occurred.

8.7 Repaired Wing — New Crack on Upper Skin

This new chordwise crack was detected on the inner surface of the upper skin similar to the earlier crack at station 2200 at 5,572 hours into the test. At the time of detection, it was 130 mm in length, and it was left unrepaired to monitor its propagation rate. After 15,500 hours its length was 196 mm, a crack growth rate of approximately 6.6 mm per 1,000 hours, but at 22,000 hours this had slowed to about 2 mm per 1,000 hours.

8.8 Repaired Wing — Propagation of Existing Gel Coat Cracks

Considerable stress-induced gel coat cracking occurred on the salvaged wing during the crash, but there has been no propagation of these cracks throughout the test.

8.9 New Wing — Cracking of the Gel Coat on the Lower Surface

A crack in the gel coat on the tension surface of the skin over the lower spar boom was detected at station 4650 where the strain is 714 microstrain/g. The crack, detected at 11,138 hours, is in the chordwise direction, and measured 46 mm in length (the spar cap is 50 mm wide at this point). It did not grow in length, and there was no evidence that the crack had extended beyond the depth of the gel coat (nominally 0.4 mm) when it was removed at 18,000 hours. The crack is believed to be stress-related rather than environment-related.

8.10 New Wing — Inadvertent Damage to Upper Spar Cap and Skin

The test is not environmentally controlled, which means that the test article has not been subject to the range of temperatures, humidity and ultraviolet light to which a sailplane is normally exposed. A controlled environment test box was installed on a segment of the new wing in an attempt to reproduce weathering of the gel coat. shortly after installation of the chamber, at 18,743 hours, the humidity and temperature environment significantly exceeded specification, causing a buckling instability of the skin and compression failure of the upper spar cap. The lower spar cap was undamaged. Repairs to the upper spar cap were conducted according to standard procedures, involving removal and replacement of a wedge-shaped segment of material either side of the failure. Failed segments of wing skin were also removed and replaced. A load deflection measurement following repair showed that the bending and torsional stiffnesses had not altered from the undamaged state.

9. CONCLUSIONS

Over 22,000 simulated flying hours have been accumulated on the full scale fatigue test article. There has been some growth of damage from sections which were deliberately left unrepaired, and some minor new damage, but no catastrophic major failures have occurred in either the new or repaired wing. The following conclusions may be drawn from the test to date:

9.1 Cracking precedes delamination,

9.2 The propagation rate of delamination is slow on the tension surface.

9.3 All standard repair techniques employed to date have been validated.

9.4 Field repairs show greater susceptibility to delamination.

9.5 The current GFA maximum inspection interval of 1,000 hours has been confirmed.

The full scale fatigue is planned to continue until either a catastrophic failure occurs, or a total of 36,000 equivalent flying hours have been accumulated. The current rate of testing indicates this goal may be achieved in 1992.

10. REFERENCES

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