XIV NASAS: Fatigue, Fracture and Ageing Structures; Eds.: CM Manjunatha, VR Ranganath , RK Paretkar & DR Peshwe
30-31 January 2006, Visvesvaraya National Institute of Technology, Nagpur, pp. 384-392.

Damage Growth Studies on Composite Flap Structures Under Fatigue Loading

V Sesha Kumar*, HV Ramachandra, B Varughese, KM Gaddikeri,
Ramesh Sundaram and M.Subba Rao
Advanced Composites Division, National Aerospace Laboratories, Bangalore 560 017

Abstract

The rib-skin construction using carbon fiber composites has been an attractive design option for aircraft control surfaces like aileron, elevator, rudder, flap etc. The major concern in such structures is the debonding between skin and rib flange which can occur due to low velocity impacts like tool drops, run way debris etc. Such debonds which occur in service are barely visible and may not get detected till the next inspection schedule. The integrity of the structure in the intervening period is of great concern to the designers. In the present work, the structural integrity of a composite flap structure having multiple debonds at the rib skin interfaces under fatigue loading has been addressed. The flap is subjected to cyclic loading at design limit load for 110000 cycles using a whiffle tree mechanism. The strains have been monitored at different locations to understand the behaviour of structure during the test using strain gauges and Fiber Bragg Grating (FBG) sensors. Ultrasonic A-scan was used to monitor the defect growth after each block of 1000 cycles. The growth of debonds was not significant during the fatigue testing. The strain levels did not change appreciably throughout the test period indicating the damage tolerance capacity of the flap structure. The low growth of debonds was attributed to the low level of strains in the structure since the flap design is driven by stiffness considerations.

Keywords: Composites, Skin-stiffener debond, Cyclic loading, Limit load, Whiffle tree mechanism, Strain gauges, Fiber Bragg Gratings (FBG), Ultrasonic A-scan, Damage tolerance

1. Introduction

Fibre reinforced composites have become an attractive choice in aerospace structures due to their high specific strength, high specific stiffness and corrosion resistance. The ability of composites to integrate large number of smaller parts to a single part through cocuring and cobonding technologies has been widely exploited. The resulting box structures having multi rib/multi spar constructions are structurally superior. The principal advantages of this technology are the elimination of stress concentrations due to holes, reduced assembly time and associated costs. The application of these materials in primary structures has been limited by the lack of experience about damage tolerance limits of the structure. Damage tolerance[1] is defined as the ability of a structure to tolerate reasonable level of damage or defect that may be encountered during manufacturing or while in service, which does not result in a catastrophic failure prior to its detection by inspection.

The bonding between the flange of the rib and the skin is crucial for the load transfer. The debonding between skin and stiffener can occur in manufacturing due to improper tooling and in service due to impacts. The debonds in manufacturing do not pose problems as they are easily detected during ultrasonic inspection and decision is made through the design disposition. However, the components in service are vulnerable to both accidental handling damages such as tool drops and foreign object damages such as bird hit, hail stones and runway debris. The debonds occurring due to low velocity impact are classified as barely visible impact damage (BVID). As such debonds may not get detected till the next inspection schedule, the integrity of the structure in the intervening period is of great concern to the designers.

* Corresponding Author, Tel: 91-80-25086401, Fax: 91-80-25267352, Email: rameshs@css.nal.res.in

Studies on the effect of skin-stiffener debond in a panel and its growth has been carried out by continuous analysis, fracture mechanics techniques and finite element analysis. Wang et.al.[2] performed a continuous analysis for determining interfacial stresses and strain energy release rates at the interface of skin and stiffener. An analytical tool based on FEM was used by Yap[3] et.al. to study the damage initiation and this was compared with experimental results on a panel. The analytical tool was validated using test data obtained from a large stiffened panel that contained a debond. The critical parameters were established based on the onset of crack growth using fracture mechanics. Parametric studies viz., debond size, debond location, multiple debonds etc., were conducted using the analytical tool. Krueger et.al.[4] developed three procedures to determine strain energy release rates in composite skin/stringer specimens for various combinations of uniaxial and biaxial loading conditions. Greenhalgh et.al.[5] carried out testing and failure analysis of skin-stringer panels having defects caused by manufacturing and impact located at two places viz., between the stringers and at the foot of stringers. It was concluded that the foot of stringers were more sensitive to the defects caused by impact compared to the defects caused in manufacturing. Between two stringers, the manufacturing defects were more sensitive compared to defects due to impact.

The failure process in composite components is very complex owing to the anisotropy and non-uniform distribution of stresses. Most of the studies are at a laminate level and the results cannot be extrapolated to a structure with multiple load paths. In a delamination investigation with multiple stiffeners, Wiggenraad[6] concludes that the delamination growth in stiffened panels is often influenced by structural features such as the presence and spacing of ribs. The reliable way of demonstrating the damage tolerance of a composite structure with defects is through a comprehensive structural testing program. Federal Aviation Regulations (FAR) stipulate that the damage tolerant capability of a composite structure should be demonstrated through analysis and tests. Though rigorous analytical techniques are available now, they can only partially replace the need for structural testing. The demonstration of structural integrity by full scale testing has become an important requirement towards the certification of aircraft structures.

The studies on full scale testing of structures with rib-skin debonds are not available in open literature. In an earlier study[7] composite flap structures that are developed for NAL's SARAS aircraft were considered. The outboard flap had multiple debonds between rib and skin interfaces. The full scale static testing of the flap structures with defects was carried out up to design ultimate load. The structure withstood the load without any failure and debonds did not grow at design ultimate load. The load carrying capability of the flap structure for sustenance of a single flight was demonstrated experimentally. In order to generate the experimental data on the effect of fatigue on the debonds between the rib and the skin on the structural integrity, the same out-board flap has been subjected to cyclic loading up to limit load.

2. Test Article

The outboard flap of SARAS aircraft is a cambered aerofoil manufactured using carbon/ epoxy prepreg manufactured by Ms.Hexcel Composites. The flap structure is divided into a nose box and an aft box. The aft box is a cobonded construction with 8 chordwise I-section ribs bonded to the skins using Redux 319A film adhesive. The root rib, tip rib and the spar are mechanically fastened to the aft box. The nose ribs are mechanically fastened to the nose skin. An exploded view of the flap is shown in Figure 1. After curing, debonds between the top skin and rib interface were noticed at many places. Most of the debonds were of half width of the bonded flange area and lie on one side of the rib web as shown in Figure 2. The bottom skin and rib interface did not have any debonds.

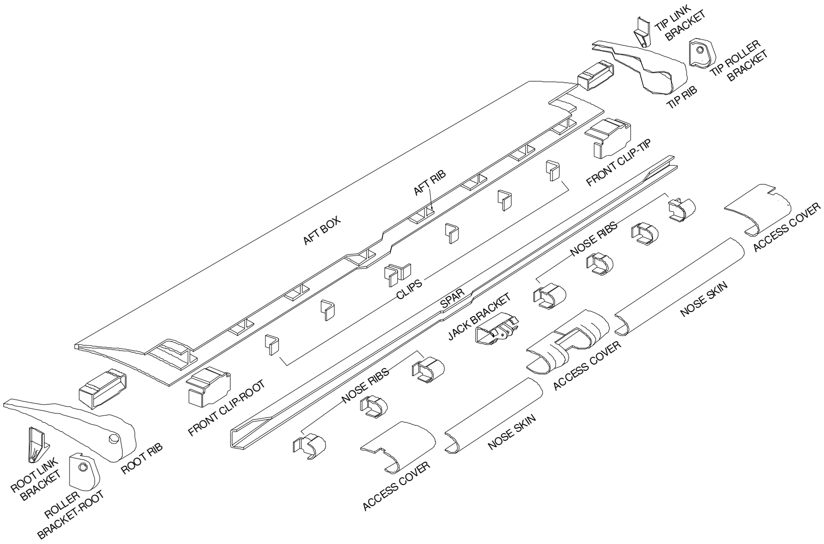


Fig.1. Exploded view of outboard flap

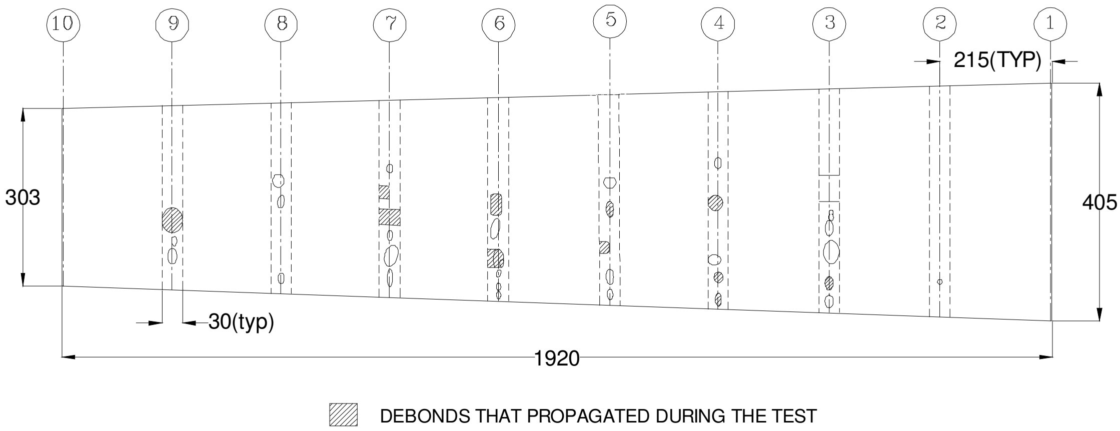


Fig. 2. Debonds in the rib-top skin interface of aft box of outboard flap before test.

3. Aerodynamic Load

The test loads have been evaluated from the pressure distribution patterns prescribed by FAR 25.345 and 25.457. The critical loading case is for the transitory position (FAR 25.457) of the flaps at 30°deflected case. The pressure distribution acting on the flap is divided into horizontal (chord wise) and vertical components. The vertical load on the top and bottom skins are pulling (suction) and pushing (pressure) loads respectively. The magnitudes of suction, pressure and chord wise loads are 394Kg, 213 Kg and 106 Kg respectively.

Since the simulation of varying pressure distribution over the entire area of the exposed surface of the flap is extremely difficult, statically equivalent lumped loads are applied. The pressure and chord wise loads are distributed as concentrated loads at each rib location. The suction load is distributed as lumped loads to act between the ribs so that the rib skin interface is subjected to tensile loads. This also facilitates in avoiding fouling between suction and pressure loading members. The suction and pressure loads at each rib were further divided into two loads such that their resultant lies on the center of pressure. Figure 3 shows the pressure distribution in top and bottom skins. Figure 4 shows all the lumped load locations on top and bottom skins of the flap.

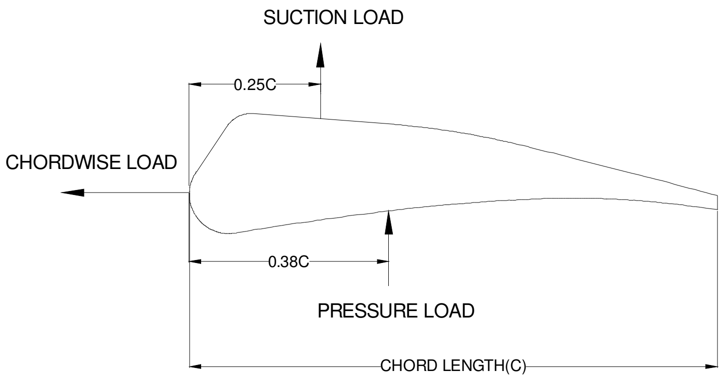


Fig. 3. Resultant pressure distribution on flap.

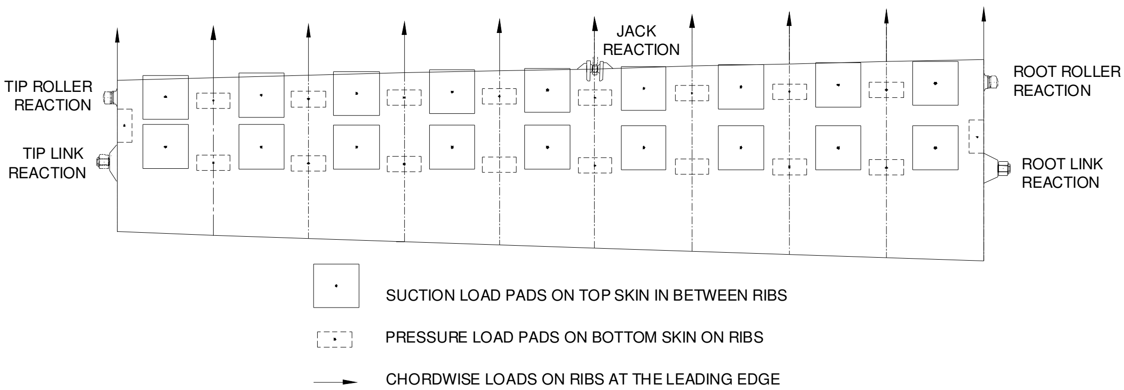


Fig. 4. Flaps with lumped loads and reactions.

4.1. Loading Mechanism

The total aerodynamic load was redistributed as 46 concentrated loads viz., suction and pressure loads are applied at 18 points and chordwise load at 10 points. A special whiffle tree loading mechanism as shown in Figure 5 and 6 was designed to reduce all the 46 loads to a single actuator thereby avoiding the problems that are encountered in using multiple actuators. This eliminated the need for three actuators which otherwise would have been be needed to apply suction, pressure and chord wise loads through independent whiffle tree systems. In such a system, all the loads must be applied simultaneously and synchronization of three actuators would become very difficult in cyclic loading without any phase lag between the actuators.

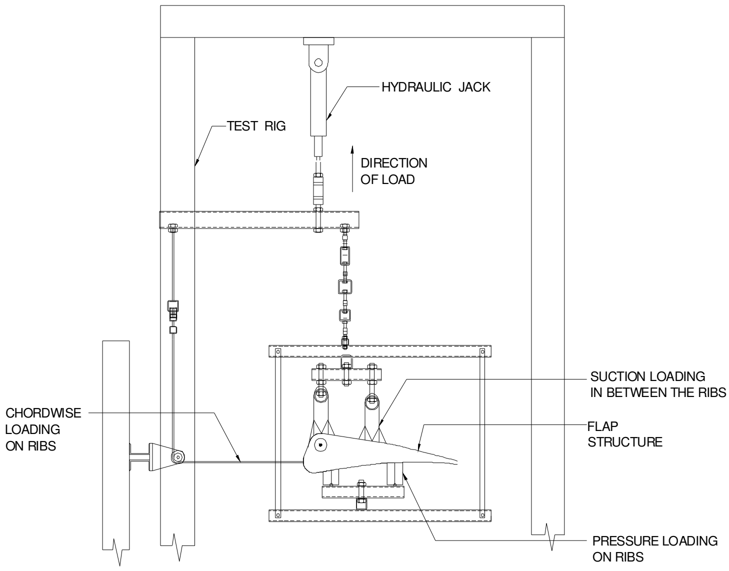


Fig.5. Fatigue test set up – End view

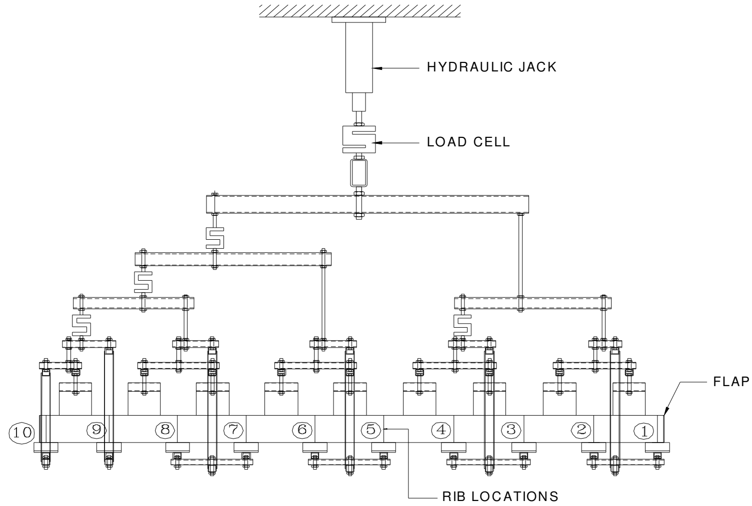


Fig.6. Fatigue test set up – Elevation

The arms of the whiffle tree were designed such that the pressure load at one rib was connected to the adjacent suction load. The chord wise loads in the horizontal direction were changed to normal loads through pulleys. Finally, all the loads were carefully connected to the actuator by avoiding fouling between the whiffle tree loading members. Extreme care was taken to distribute the load precisely as the magnitude of load at each load location is different. Load cells were placed at required locations in the whiffle tree to ensure that the load distribution is as required. The pressure loads were applied using the wooden pads with rubber lining. The suction and chordwise loads were applied using special canvas pads bonded to the flap surface.

5. Simulation of Attachment Conditions

The Saras flaps have two rollers (at 9% of local flap chord) at its ends, which roll in the tracks mounted on the wings. There are two links (at 50% of local flap chord) attached at both ends of flaps to wing. The flaps are operated by a screw jack located in the mid span. The entire mechanism helps the flaps to achieve the required translation and rotation. The flaps can be operated from 0° to 40°. The aerodynamic load that acts on the flap is reacted at all five points (2 roller points, 2 link points and 1 jack point) in specific directions depending on the flap position. The direction of these reactions depends on position of the flap. Test was conducted for 30°deployment case. The flap along with all the boundary conditions was rotated by 30° so that its attitude was horizontal which is convenient for testing. The rollers on the flaps were constrained so that the movement of flap was avoided. The load cells were placed in the link rods to monitor link reactions. Figure 7 shows the simulation of attachments in the test set up.

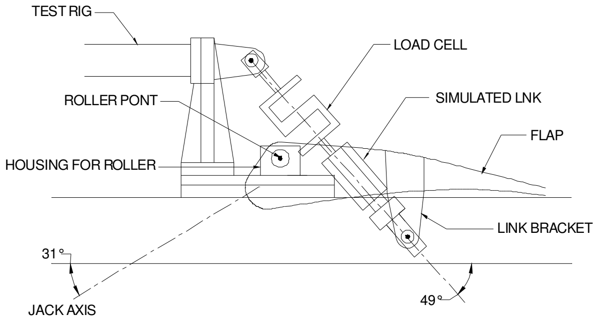


Fig. 7. Simulation of attachment condition on flap.

6. Positions of Sensors

The resistance strain gauges were mounted on the debonded region to study the variation of the strains. Few gauges were also mounted on the healthy regions very close to the debonds to observe the change in strain when the debond grows. A 100 channel data logger system was used to monitor the strains in the structure. One dial gauge was placed at the point of maximum deflection of the flap. The acoustic emission sensors were placed to capture the real time data of acoustic activity in case of damage growth or initiation. The top surface was also instrumented with FBG sensors. FBG sensors are one of the potential candidates for monitoring and mapping strains due to their advantages of lightweight, small size, high sensitivity and immunity to electromagnetic interference (EMI). FBGs have multiplexing capability along a single fibre with one input/output lead and ability to be embedded within composite structures. The distribution of all the sensors on the flap top surface is shown in Fig. 8.

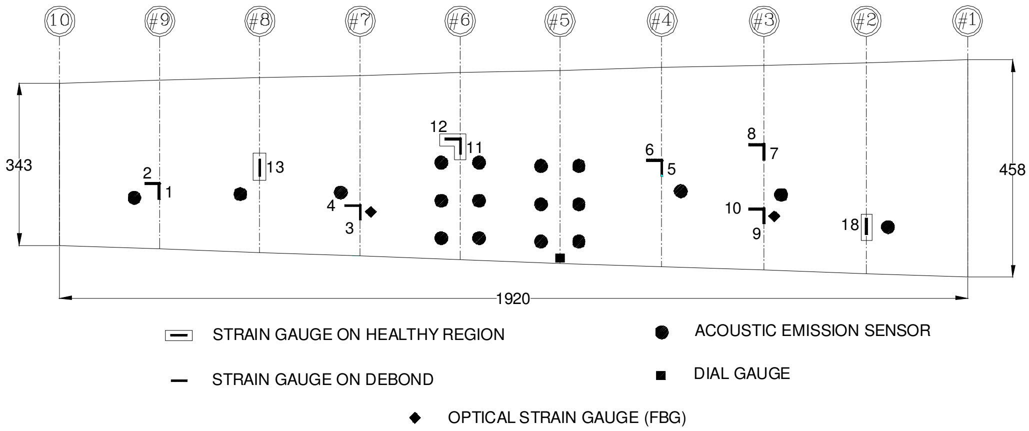


Fig. 8. Sensor positions on top skin of outboard flap.

7. Testing

The cyclic limit load was applied on the flap using the whiffle tree loading mechanism and a hydraulic jack. A load cell was used to monitor the applied load. The component was loaded at a very low frequency (0.15Hz.) because of number of loads, complex loading system and power pack limitations. Fatigue load equal to design limit load 713 Kg (the sum of all the three components) was applied on the component. The minimum load was fixed at 200 Kg so that the loading pads do not get disturbed during the cycling. The strains and deflections were monitored under static load after every 1000 cycles. The acoustic emission and FBG data was monitored continuously during the load cycling. The debond growth was assessed using ultrasonic A-scan after every 1000 load cycles without removing the component from the test rig. 110000 fatigue cycles have so far been completed.

8. Results and Discussions

The debonds in the skin and rib interface did not show any growth till 25000 load cycles after which their growth initiated slowly at various locations as indicated by the ultrasonic A-Scan between few cycle blocks. However it was found that the growth of debonds at a given location was found to be intermittent and not a continuous phenomenon. Different debonds had grown by different magnitudes in different cycle blocks. This may be due to the variation in the quality of bond, which is assumed to be uniform. Figure 9 shows the strain plot of gauges 1 and 5 which were mounted on the debonds. Significant changes in the strain levels could not be seen though these debonds had grown slightly during the test. Figure 10 shows the strain plot of strain gauges 13 and 18 which were mounted on a good region closer to the debonds. The strains in these gauges did not change in spite of some growth. The low growth of debonds was attributed to the low level of strains in the structure since the flap design is driven by stiffness considerations.

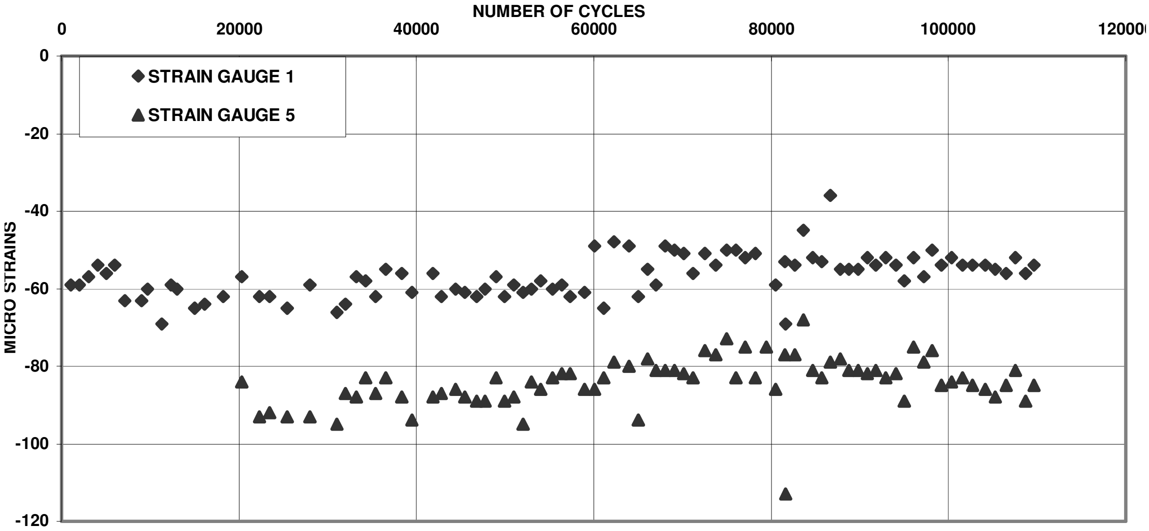


Fig.9. Strain data on debonded regions

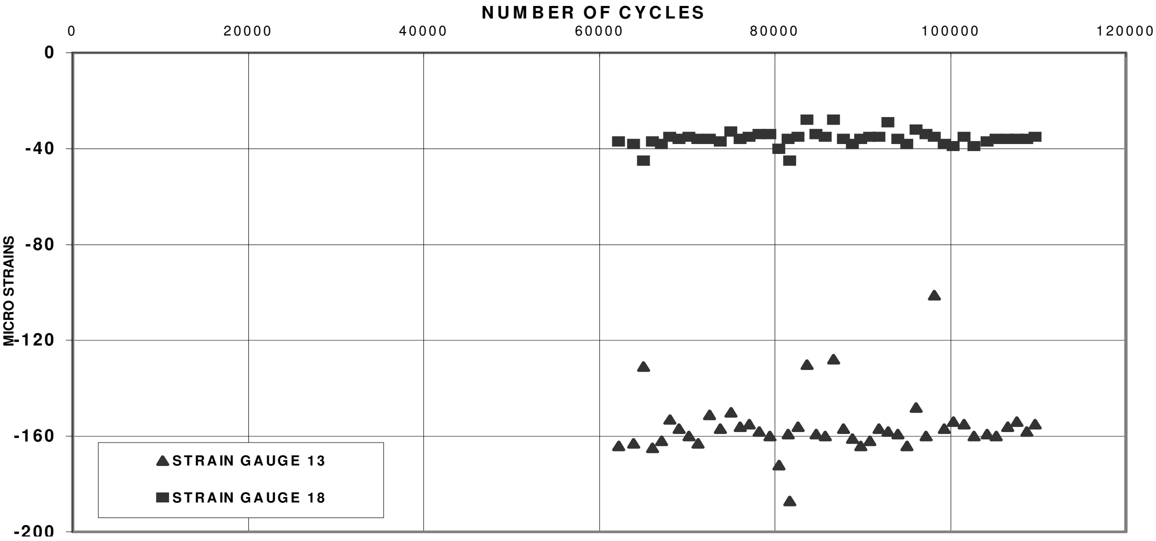
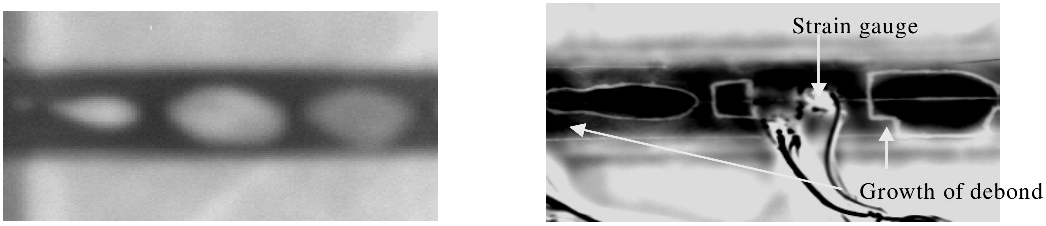


Fig.10. Strain data on healthy regions

It was also observed that the FBG interrogating system was able to capture the strains during cycling, which is not possible with conventional strain gauges. The strains measured statically compared well to the strains measured from the strain gauges effectively validating the measurement of the FBG sensors. The FBG sensors were able to pick up the small changes in strain due to the growth of debonds and subsequent stress relaxation in the skin. However, this is a very local phenomenon and does not affect the overall structural integrity.

As there was no appreciable change in the strains, it was concluded that the structural integrity of the flap is intact after 110000 cycles of limit load. The small amount of growth in debonds was not enough to cause the loss of stiffness of the structure which is vindicated by the strain data. This was also corroborated by the dial gauge data to monitor the maximum deflection of the structure. The deflection of 10mm under limit load at the start of the cycling was increased by 1mm at the end of the cycling. The increase in deflection is not significant to conclude that there was a stiffness loss in the structure.

The infrared thermography images were taken at each rib before the start and end of fatigue cycling. Figure 11 shows thermography images for the rib 3 before and after the test. The thermography report at each rib was in good agreement with the ultrasonic A-scan report. The acoustic emission sensors continuously monitored the test and the results are presented in a companion paper[8].



Before the test After the test

Fig.11. Infrared thermography of rib no. 3

9. Conclusions

Full scale fatigue testing has been carried out on the flap structure with skin to stiffener debonds for the damage tolerance studies. The resistance strain gauges, FBGs, ultrasonic A-Scan, infrared thermography and acoustic emission sensors were used to assess the debond growth during the test. The flap structure was subjected to cyclic load at design limit load for 110000 cycles using an innovative whiffle tree mechanism. There was no growth of debond noticed till the end of 25000 cycles. The changes in strains during the damage growth were not significant as the growth of debonds was very small at many locations.

The strains measured by FBG sensors statically compared well to the strains measured from the strain gauges. The strain gauges and FBG sensors gave information about the stiffness of the structure. From the test, it was found that the growth in any debond was discrete and intermittent which can be attributed to the quality of the bond. FBG sensors were able to capture small growths in debond. However, these small changes are local and do not affect the structural integrity of the flap.

The images from thermography helped to assess the growth of debonds and helped to correlate the manual ultrasonic A-scan report. Acoustic emission monitoring proved to be an extremely useful tool for online monitoring of the health of the structure. From the results of test so far, it can be concluded that the debonds in the skin and rib interfaces of a structure subjected to low level of strains can sustain a number of limit load cycles. However, the number of cycles up to failure should be established before fixing the inspection intervals.

Acknowledgements

The authors acknowledge the fabrication, static testing and NDE teams in ACD, NAL for their support in conducting the test. The authors thank Mr.Nitesh Gupta for carrying out the FBG measurements. The authors thank Director, NAL for his interest and support in this work.

References

1. H.W. Bergmann, *Fatigue behaviour and damage tolerance of composites*, Lecture Notes, National Aerospace Laboratories, Bangalore, 1992.
2. J.T.S.Wang and J.T.Huang, "Skin/stiffener interface delamination using continuous analysis", *Composite Structures,* 30, pp. 319-328, 1995.
3. J. W. H. Yap, M. L. Scott, R. S. Thomson and D. Hachenberg, "The analysis of skin-to-stiffener debonding in composite aerospace structures," *Composite Structures,* 57, pp. 425-435,2002.
4. R.Krueger, P.J.Minguet and T.K.O'Brien, "A method for calculating strain energy release rates in preliminary design of composite skin/stringer debonding under multi axial loading," NASA/TM-1999-209365, 1999.
5. E.Greenhalgh, C.Meeks, A.Clarke and J. Thatcher, "The effects of defects on the performance of post-buckled CFRP stringer stiffened panels", *Composites,* Part A 34, pp. 623-633, 2003.

6. J.F.M. Wiggenraad, R. Aoki, M. Gadke, E. Greenhalgh, D. Hachenberg, K. Wolf and R. Bübl, “Damage propagation in composite structural elements – analysis and experiments on structures”, *Composite Structures,* 36, pp. 173-186, 1996.
7. Byji Varughese, H.Y.Subramanya, B.Ramaniah, N.Nagarajappa, Kotresh M. Gaddikeri, G.M.Kamath, Ramesh Sundaram and M.Subba Rao, “Damage tolerance on carbon fibre composite flap structures having defects at rib-skin interfaces”, Proceedings of ICASI-2004 International Conference on Advances in Structural Integrity, ICASI/DT-C53, 2004.
8. S.Sanjeev Kumar, B.Ramaniah, H.N.Sudheendra, G.M.Kamath, and M.Subba Rao, “Application of acoustic emission to study debond growth in co-cured / co-bonded composite structures under fatigue loading” (To appear in Proceedings of XIV NASAS National Seminar on Aerospace Structures, January 2006).