PATCH REPAIR OF CRACKS IN THE UPPER LONGERON OF AN F-16 AIRCRAFT OF THE ROYAL NETHERLANDS AIR FORCE (RNLAF)

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Abstract

Cracks were detected in the flange of the upper right longeron 16B1111-21, FS 186.90 – FS 247.89 of an F-16 aircraft of the RNLAF. The National Aerospace Laboratory (NLR) was asked by the RNLAF to consider the possibility of a bonded patch repair. The repair had to last 400 flight hours, at which point the specific F-16 aircraft would be retired from service. Titanium 6A14V sheet was used for repair of the 2 mm thick 2024-T62 aluminium longeron flange. A symmetric bonded repair could be done with a room temperature curing acrylate based adhesive.The feasibility of the proposed repair geometry was determined by performing spectrum fatigue tests.

After repair of the F-16 aircraft, periodic inspections were done to check on debonding and fatigue crack propagation. With the retirement from service of the specific F-16 aircraft the project was successfully completed.

1 Introduction

During a phase inspection of an RNLAF F-16, cracks were detected in the upper right longeron 16B1111-21, FS 186.90 – FS 247.89. The cracks occurred in the longeron flange for fixing the access panel 2408, at the location of the third fastener hole from the right of the panel, see figure 1. There were two cracks: one in longitudinal direction (outside the hole) and one transverse crack through the hole, see figure 2. The longitudinal crack length was about 70 mm and the transverse crack about 22 mm.

Due to the limited available space for repair a conventional repair could not be applied to the cracked longeron, and Lockheed Martin advised removal and replacement of the concerned longeron. Removal and replacement of the longeron would require at least 600 man hours. Further, delivery time for a new longeron was uncertain. Hence the possibility of a bonded patch repair was considered. The repair had to last 400 flight hours, since the specific F-16 aircraft would then be retired.

This paper reports on the analysis of the cracks in the longeron, the bonded patch repair, supporting spectrum fatigue tests, and the patch inspection results during the remaining life of the F-16 aircraft.

2 Damage analysis of the cracks in longeron 16B1111-21

Figure 2 shows the main cracks that were found in the upper attachment flange. In the top view it is seen that the flange is somewhat deformed and the upper surface shows signs of rubbing. Further, it was observed that the nut riveted to the flange lower side had an oblique orientation. These discrepancies indicated a bad fit of the panel on the longeron flange.

The specific access panel had been previously repaired with a relatively large stiffener that ended at the fastener hole where the cracks were found (Fig. 3). It is most likely that the access panel plus this large stiffener had introduced large static bending stresses in the flange owing to the bad fit.

Operational aircraft loads in combination with assembly-induced bending stresses most probably caused the longitudinal crack first. Subsequently, the transverse crack initiated from the fastener hole owing to dynamic tensile stresses in the longeron.

This conjectured sequence of damage propagation could not, however, be verified from the fracture surfaces, since no material could be cut out of the longeron if it were to be repaired.

3 Longeron repair

3.1 Conventional repair

According to the report of Air Force Base Twenthe [1], a conventional mechanical repair could not be applied to the cracked longeron, mainly because of the limited available space for repair. This was also the opinion of Lockheed Martin [2]: "*Due to the severity of the cracking in a part that is a critical load path, a repair is not feasible*". They advised removal and replacement of the concerned longeron.

For removal and replacement of the longeron at least 600 man hours were required [1]. Since the specific aircraft would be retired from service after about 400 flight hours, replacement of the longeron would be an expensive solution. Therefore other possibilities to enable a safe remaining operational life were considered. The NLR investigated the feasibility of a bonded patch repair.

3.2 Bonded patch repair

For a bonded patch repair to be feasible, it would have to have the following advantages:

- Repair could be executed at short notice, limiting the downtime for operational usage of the concerned F-16 aircraft.
- The costs would be very small compared to the costs for removal and replacement of the longeron.
- No additional holes would need to be drilled. Such holes cause a further weakening of the structure which is an important barrier to a mechanical repair.

In determing the feasibility of a bonded patch repair for the cracked longeron, the following aspects were considered:

- Design of a bonded patch repair that restores the strength of the cracked structural component.
- Testing the design concept in laboratory tests accounting for the operational loads in the structural component.
- Methods and procedures for inspection of the repair during the operational life.

3.3 Repair design

The patch repair concentrated on restoration of the longeron strength in the longitudinal direction. However, a bonded patch does not remove the longitudinal crack. In order to prevent possible further growth of the longitudinal crack, stop drilling at the tips of the longitudinal crack was recommended.

Important items for the repair design are: patch material, patch dimensions and type of adhesive.

- Titanium 6Al4V annealed sheet with a thickness of 0.5 mm was selected for the patch. The thinner the patch, the lower the mis-match upon re-assembling the access panel 2408. Further, Titanium 6Al4V has outstanding strength, stiffness and corrosion properties.

- For dimensioning the patch repair the following material properties were used:

Longeron	$\sigma_{.2}$(MPa)	$\sigma_{.ult}$(MPa)	E(GPa)
2024-T62	400	500	70
Patch			
Ti-6Al-4V	800	1000	110

  and the following design rules:

  i) Stiffness × cross-section of flange and patch have to be similar.
  ii) Strength of flange and patch must be nearly equal.
  iii) The adhesive bond area must be large enough to transfer the ultimate load in the aluminium flange.
  iv) A symmetric load introduction from the longeron into the patch.

Figure 4 shows the proposed repair concept with a Titanium patch on both sides of the longeron flange. For a 25* mm wide and 125 mm long patch this results in the following analysis:

Stiffness check $E_{Al} \times A_{flange} = 70 \times 2 \times 22 = 3080$ kN $E_{Ti} \times A_{Ti} = 110 \times 25 \times 2 \times 0.5 = 2750$ kN
Strength check Strength Al flange $F_{max} = 22 \times 2 \times 500 = 22$ kN Strength Ti patch $F_{max} = 25 \times 2 \times 0.5 \times 1000 = 25$ kN
Adhesive shear stress at F_{max} of Al flange Assume effective adhesive length to be 45 mm Shear strength $\tau_{adhesive} = \dfrac{22000}{45 \times 25 \times 2} = 10$ MPa

- Adhesive Agomet F 310 (Degussa) was selected for bonding. This is a methyl methacrylate based adhesive that cures at room temperature. The maximum shear strength is 30 MPa. The adhesive is not so critical for the surface pre-treatment and has good gap filling properties. Accounting for knockdown factors owing to temperature and ageing, an adhesive shear load of 10 MPa (from the shear stress analysis) was considered acceptable.

4 Testing of the repair concept

In order to assess the performance of the patch repair, limited fatigue tests were performed on notched 25 mm wide aluminium strips with and without a patch repair, figure 5. Two open hole specimens and two patch repair specimens were prepared. A fastener of 6 mm in diameter was installed in the central hole of the repaired coupon specimen. The fastener was torqued at 3 Nm, simulating the panel attachment to the longeron flange. However, load transfer occurs exclusively through the bonded Titanium patches.

For fatigue testing a forward fuselage bending moment sequence should be used. However, no directly usable sequence for the fatigue testing machine was available. Measurements on F-16 aircraft of the RNLAF showed a more or less similar shape of the forward fuselage and the wing root bending spectrum. Therefore the LW-VAL F-16 load sequence was selected, this being representative for the wing root bending moment for a severe EPAF** usage, figure 6. For ease of testing the negative stress levels in the sequence were set to zero. To obtain a reasonable testing time for the specimens with the central hole, the fatigue tests were executed at 130 % of the LW-VAL load sequence. This means that the maximum gross stress in the specimens was 268.5 MPa. According to GD-16 PR8150 [4] the maximum spectrum stress for longeron 16B1111 is 196.77 MPa (28.11 ksi).

The test programme consisted of:

Open hole specimens	
Specimen 1	Static open hole tension test
Specimen 2	Open hole spectrum fatigue test to failure
Repair specimens	
Specimen 3	Fatigue testing for 15000 flights + residual strength
Specimen 4	Fatigue testing for 3000 flights + residual strength

Before the residual strength tests the patch repair specimens were inspected for debonding with the Fokker Bondtester.

Figure 7 gives the test results. The static failure load of the undamaged open hole specimen was 19.9 kN, specimen 1. Fatigue testing at 130 % LW-VAL resulted in a life of 14300 flights, specimen 2. Specimen 3 demonstrated that the repair survived 15000 flights after which a residual strength of 23.1 kN was obtained. NDI before residual strength testing showed local delamination at the edges of the titanium patches. Specimen 4 did not show signs of debonding after testing for 3000 flights. The residual strength was 25.3 kN. Figure 8 shows the failed repair specimens. Both specimens showed tensile failure of the

* Mean patch width.

** EPAF European F-16 users.

Titanium patches despite local fatigue induced delamination for specimen 3.

Although no fatigue tests were performed on bonded repairs after ageing, the repair concept seemed to be feasible for an additional service life of 400 flights.

5 Inspection of the bonded patch repair

In March 1997 the cracked longeron was repaired. Before the Titanium patches were applied the deformed attachment flange was straightened to guarantee good contact between patch and flange. The surface pre-treatment consisted of abrasive blasting (Al_2O_3) of the Titanium patches and sanding of the aluminium flange. After bonding, the repair region was provided with the F-16 primer, FMS-3035, figure 9. A spare access panel was fixed to the structure using liquid shimming to compensate for the patch thickness.

An inspection procedure for the bonded repair was established for the remaining operational life of the F-16 aircraft. The inspection concentrated on:

- Visual inspection for paint cracking at the edges of the patch (indicating debonding initiation).
- Fokker Bondtester inspection for debonding of the patches.
- Eddy Current inspection for crack propagation from the drilled-off hole.

For checking on debonding the Fokker Bondtester Model 70 was used with transducer 1414 and coupling fluid. The patch surface was divided into areas of about 1 cm^2, figure 10. The possible Fokker Bondtester responses are indicated in the same figure and were used as calibration signals during the periodic inspections.

For crack detection the Nortec-19e^{II} Eddy Current scope was used with a Nortec PR/1 kHz – 100 kHz/A pencil probe. The test frequency was 15 kHz. A calibration standard was made to monitor possible crack propagation from the stop-drilled holes. The reference signals are shown in figure 11.

Inspections after the bonded repair were performed after 25, 90, 190, 300 and 400 flight hours. No signs of paint cracking at the edges of the patches, debonding or crack propagation were found, and in 2000 the concerned F-16 aircraft was retired from service.

6 Concluding remarks

In the present investigation cracks in the upper right longeron of an F-16 aircraft of the RNLAF were investigated. The main cause for cracking was the occurrence of assembly stresses due to an improperly repaired access panel 2408. Since a conventional mechanical repair could not be done, the OEM advice was removal and replacement of the longeron. However, because the aircraft would be retired from service after 400 flights hours, a cost effective bonded patch repair was evaluated and applied to the cracked longeron. Inspections during the remaining operational life did not show any damage propagation in the repair. Thus the present investigation has shown that adhesively bonded patch repairs can be very useful and cost-effective, especially when the repair has to be sustained only during a limited service life.

References

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[3] Nederveen A. and Spiekhout D.J. *Fatigue lives of single side cracked hole specimens tested with GD made F-16 flight load spectra,* NLR CR 89104 C.

[4] – GD-16PR8150, F-16 A/B Block 1/5/10/15. *Durability, damage tolerance and service life analyses,* CDRL Items 2306 I and 2306 J.

Acknowledgement

The Scientific Research Division of the Directorate of Materiel, Royal Netherlands Air Force, is greatly acknowledged for permission to publish the performed investigation.

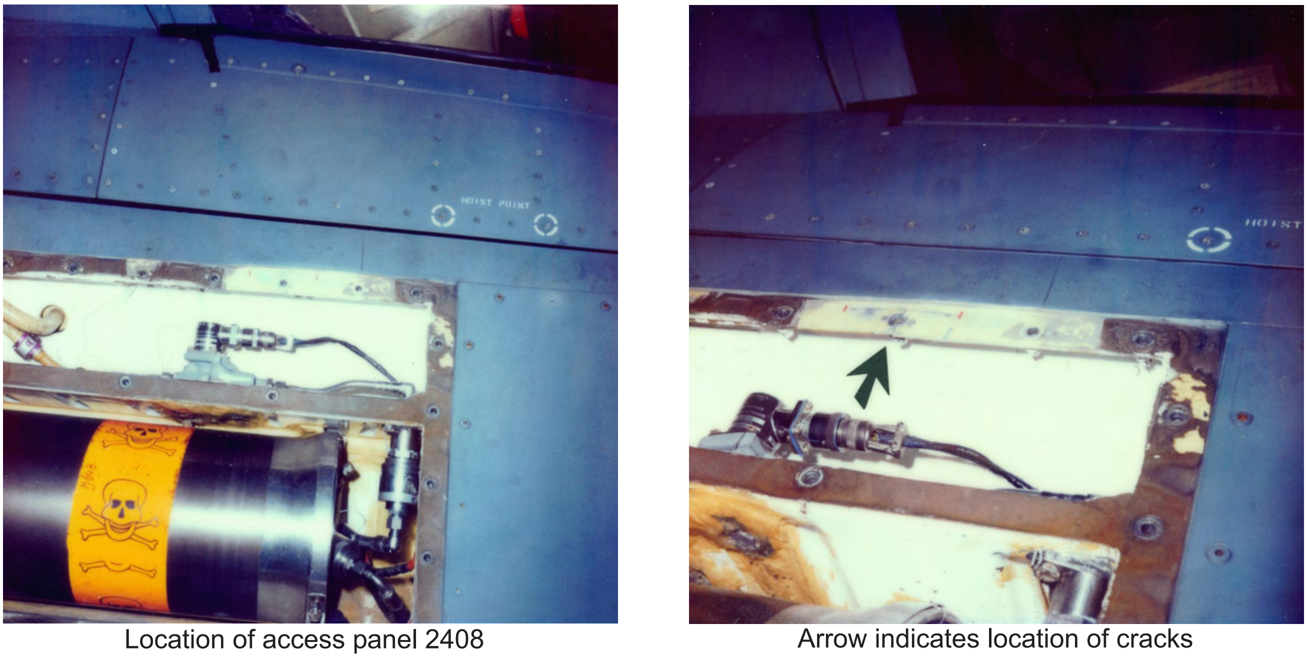


Fig. 1 Cracks in longeron 16B1111 at third fastener hole from the right (see arrow)

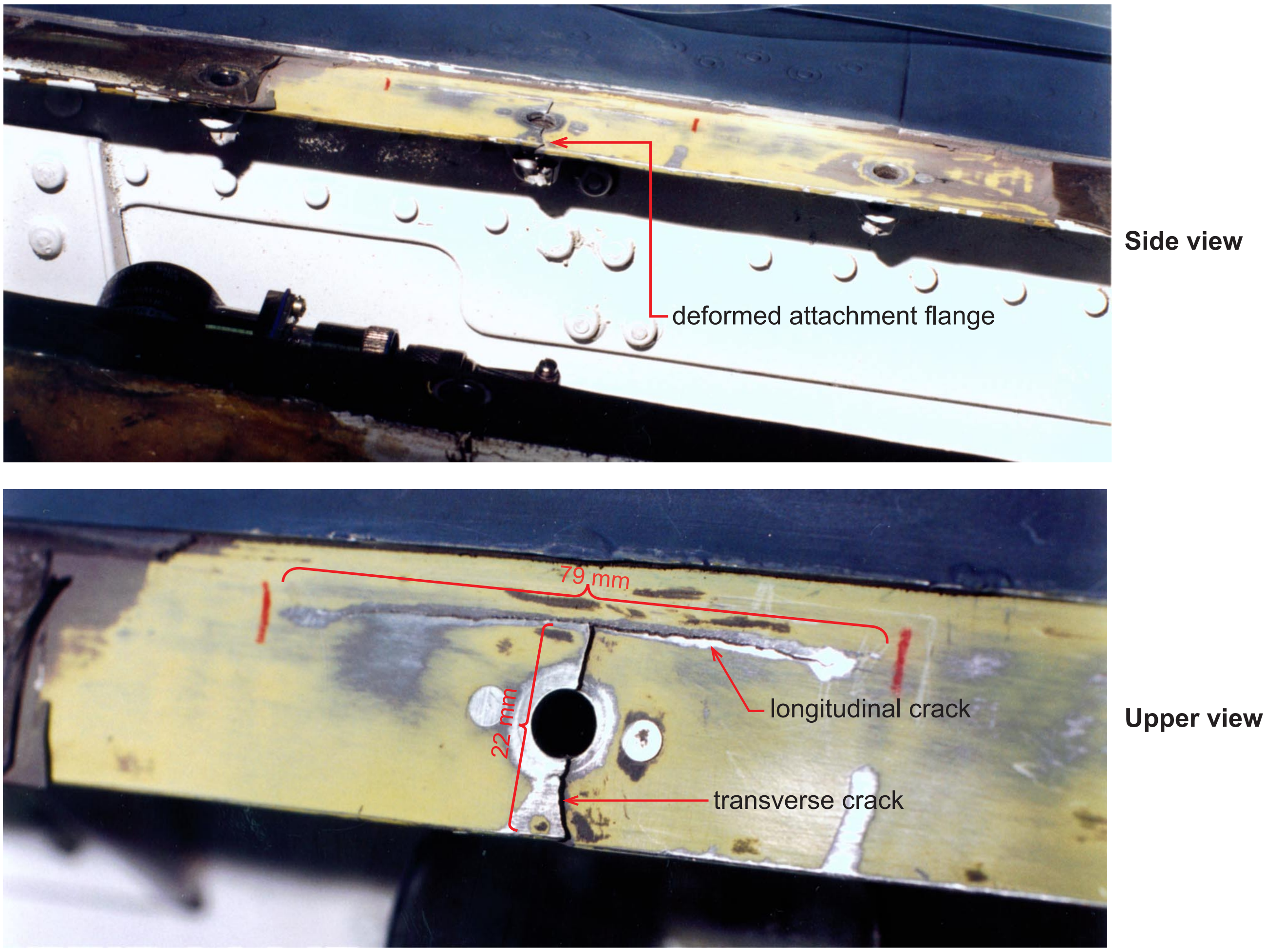


Fig. 2 Details of cracks in longeron

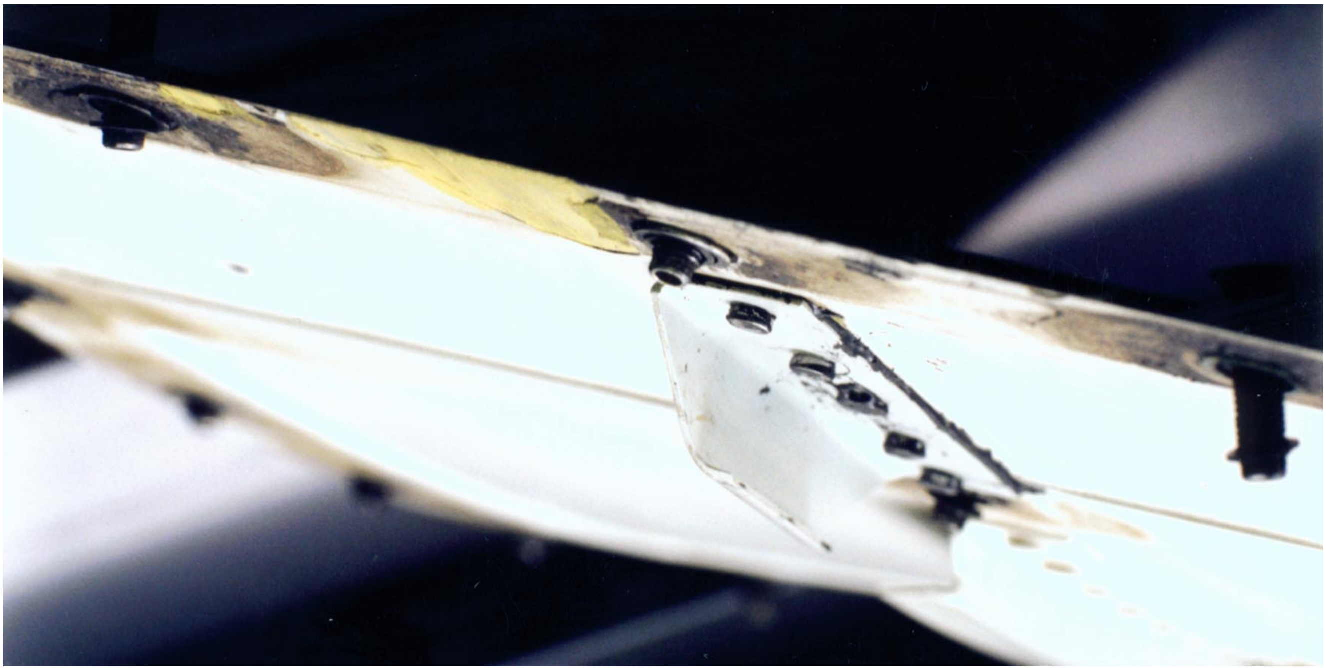


Fig. 3 Lower side of access panel 2402 with a stiffener repair

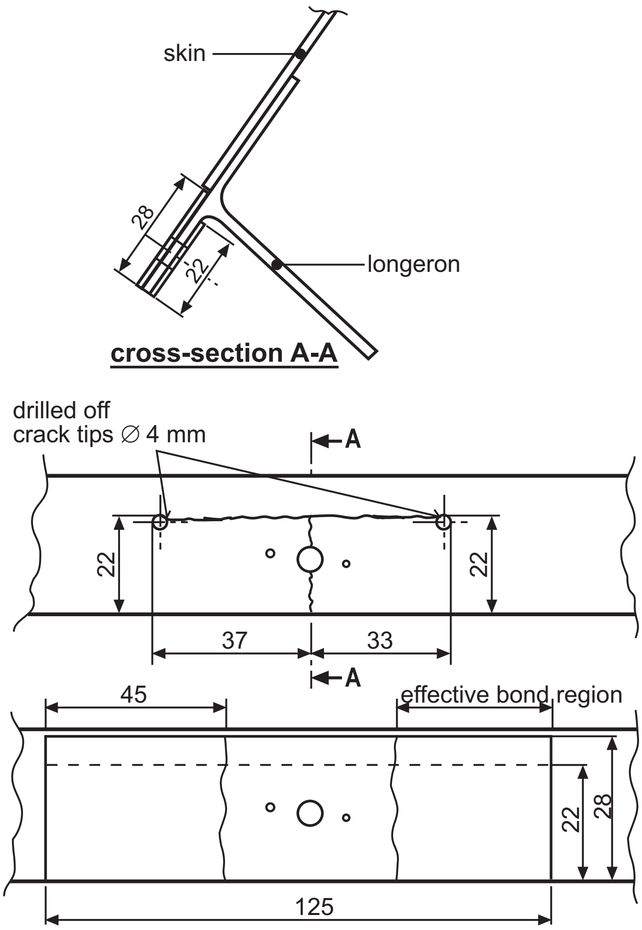


Fig. 4 Repair concept for cracked longeron

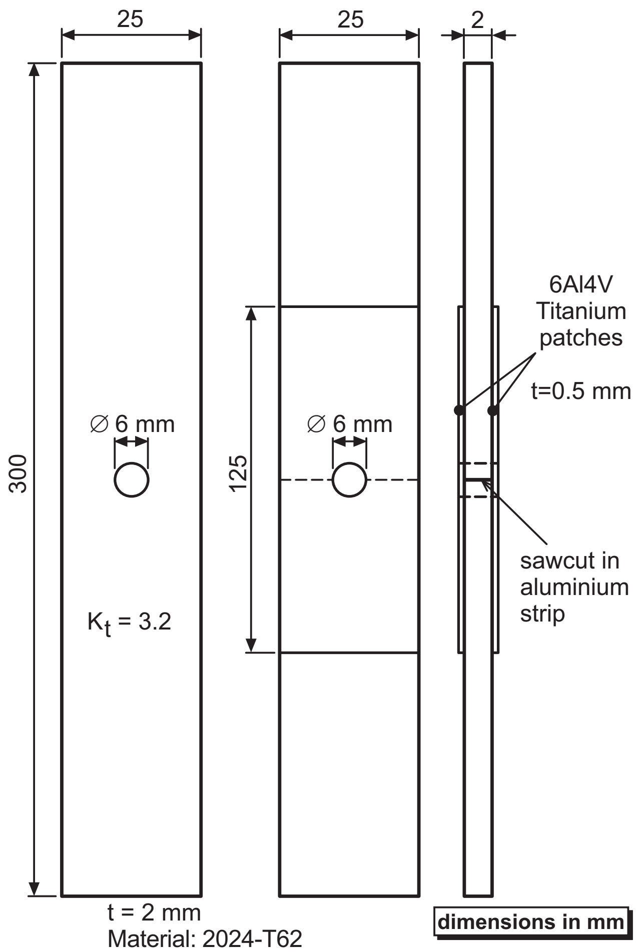


Fig. 5 Fatigue specimens for flight simulation fatigue testing

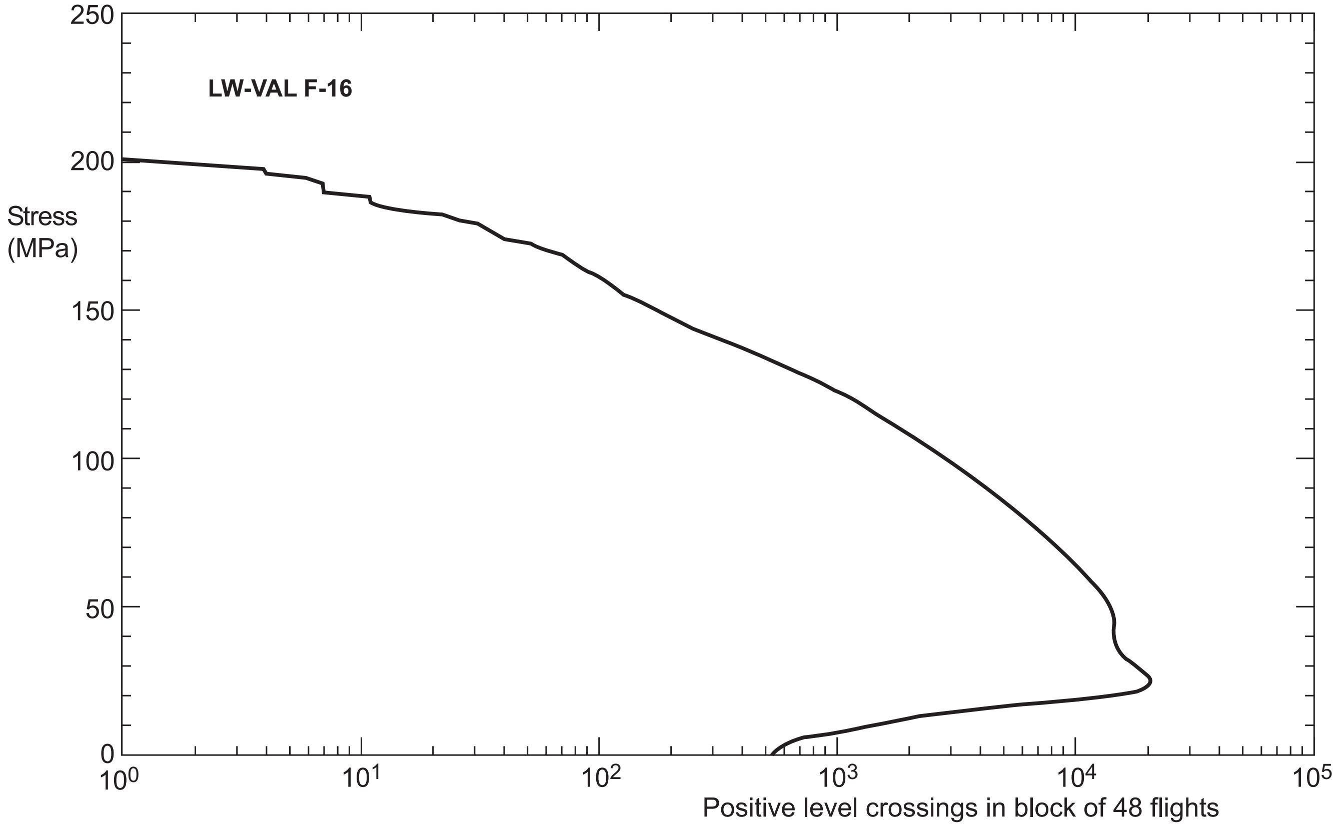


Fig. 6 Stress and cumulative level crossings of 100%-LW-VAL F-16

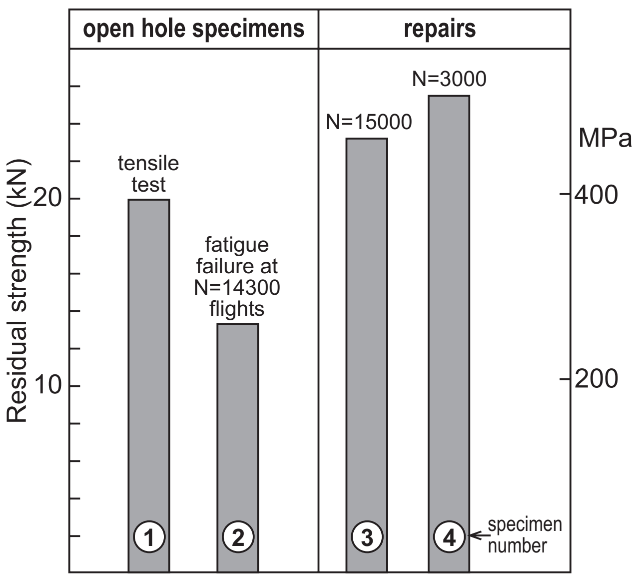


Fig. 7 Test results of experimental programme

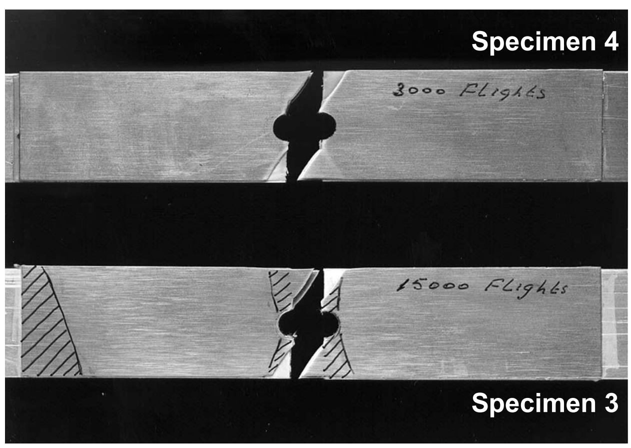
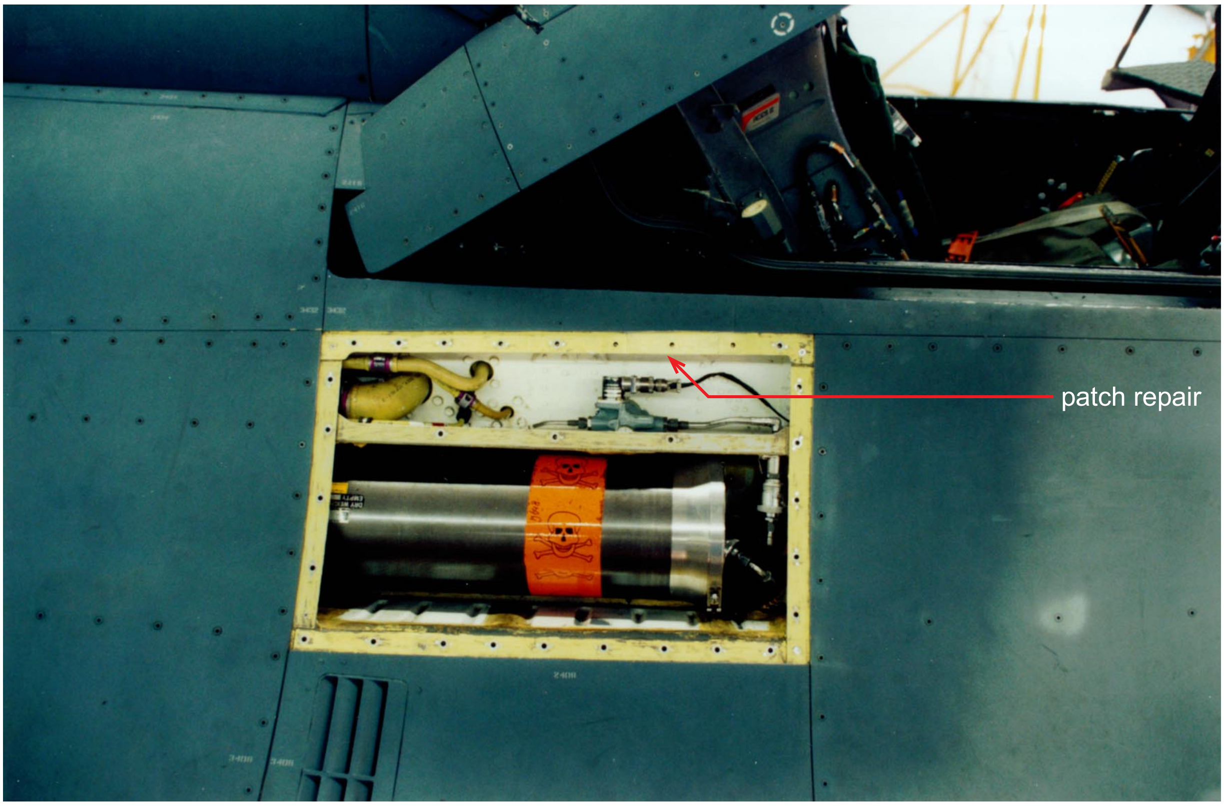
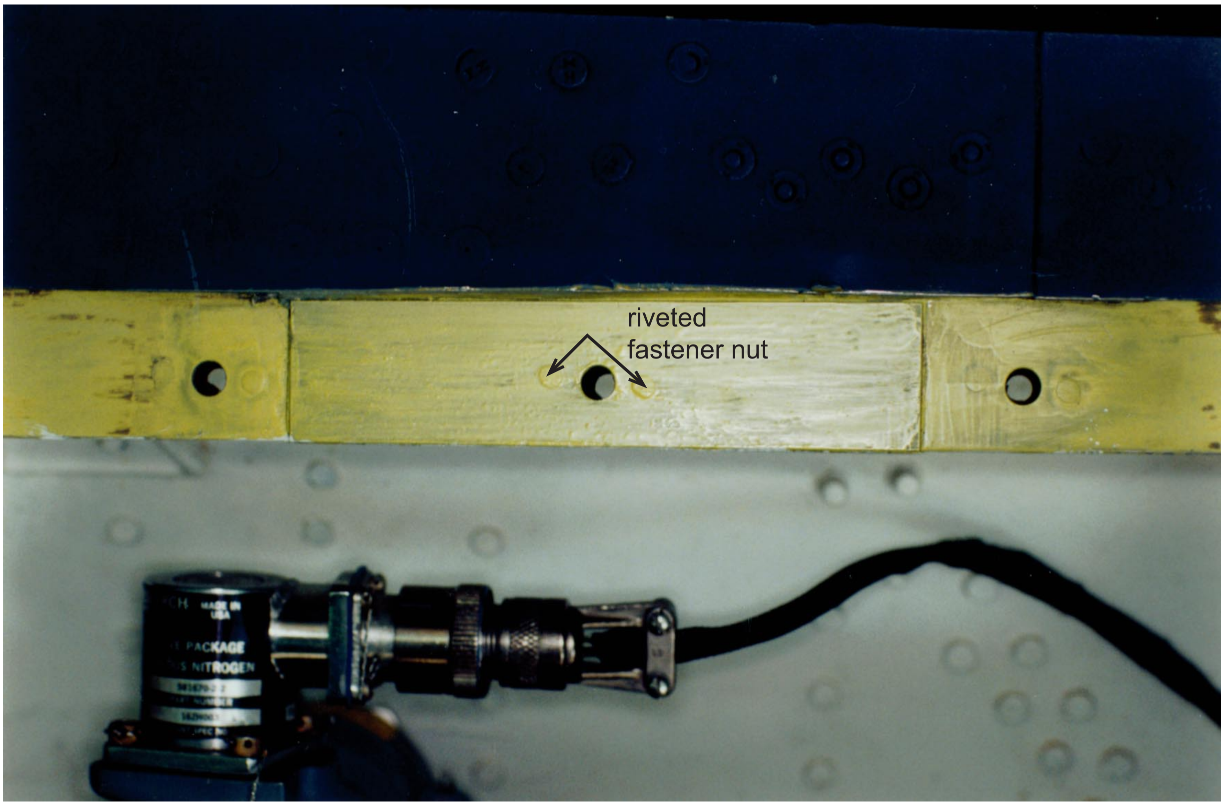


Fig. 8 Residual strength tested specimens after fatigue testing for 3000 and 15000 flight hours.(delamination after fatigue testing was only found for the specimen tested for 15000 flights)

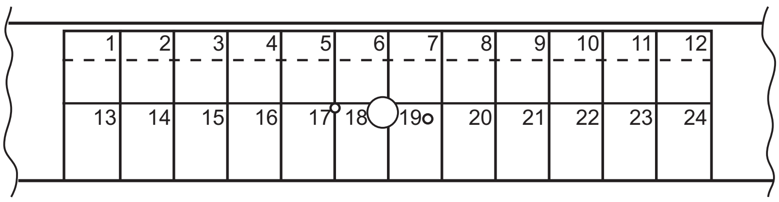


Location of the repaired attachment flange

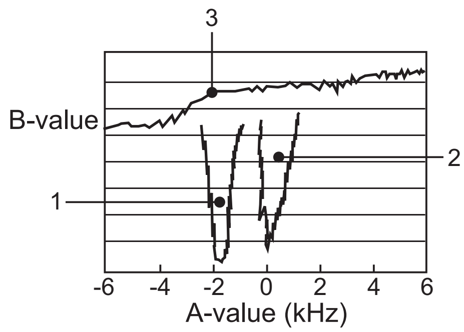


Detail of bonded patch repair

Fig. 9 Repaired longeron



Division of patch surface in discrete regions
for the Fokker Bondtester inspections



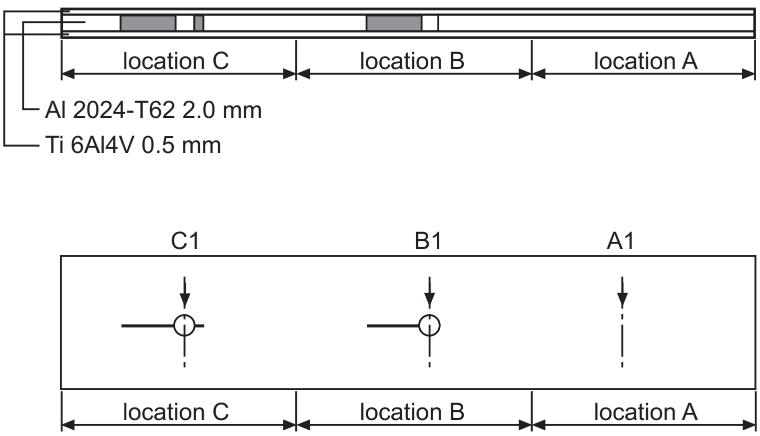
Response of Fokker Bondtester:

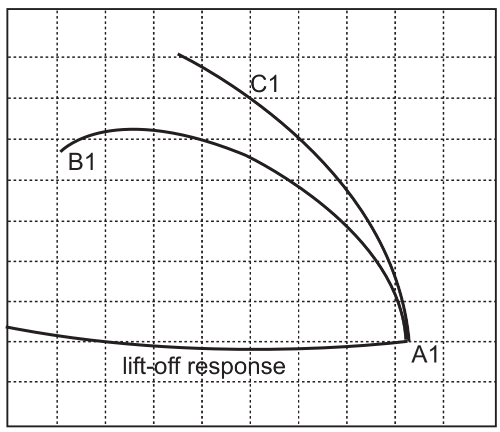
No. 1 : Probe in air

No. 2 : Probe on a debonded patch area

No. 3 : Probe on a patch area with good adhesive bonding

Fig. 10 Inspection set-up for Fokker Bondtester inspection





Location A: Defect free area

Location B: 4 mm diameter hole + 10 mm EDM notch

Location C: 2 mm EDM notch + 4 mm diameter hole + 10 mm EDM notch

Response A1: Scan over defect-free substrate

Response B1: Scan over hole with a one sided EDM notch

Response C1: Scan over hole with a two sided EDM notch

Fig. 11 Calibration standard for Eddy Current detection of crack growth from the drilled-off hole at the tip of the longitudinal crack