Nationaal Lucht- en Ruimtevaartlaboratorium

National Aerospace Laboratory NLR

Executive summary

GLARE teardowns from the MegaLiner Barrel (MLB) fatigue test

Problem area
The MegaLiner Barrel (MLB) full-scale pressure cabin test was begun in the mid-1990s. As part of the Airbus A380 development programme, the MLB was used to investigate several design solutions, structural materials and joining methods, including the use of GLARE (GLAss REinforced aluminium laminates). The test was done by Airbus Deutschland and was discontinued after 45,402 simulated flights. Stork/Fokker Aerospace then specified a programme of teardown and additional fatigue testing for GLARE structures from several key areas of the MLB. The NLR contracted to carry out this programme.

Description
The present report surveys the teardowns of a window area, a beam above the passenger door, and some stringer couplings. The teardowns began with Non-Destructive Inspection (NDI), and were followed by fractographic investigation of NDI-indicated cracks in the window and door beam locations. The main objectives were to verify the NDI techniques and capabilities, determine the fatigue initiation and crack growth behaviour, and provide data to check fatigue crack growth models for GLARE. The results demonstrated very good NDI teardown capabilities and high fatigue damage tolerance by the GLARE structures.

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Summary

The MegaLiner Barrel (MLB) pressure cabin fatigue test was part of the Airbus A380 development programme. The NLR carried out teardowns of GLARE (GLAss REinforced aluminium laminate) structures from three key locations of the MLB: a window area, a beam above the passenger door, and some stringer couplings. The teardowns began with Non-Destructive Inspection (NDI), and were followed by fractographic investigation of the longest NDI-indicated cracks in the window and door beam locations. The main objectives were to verify the NDI techniques and capabilities, determine the fatigue initiation and crack growth behaviour, and provide data to check fatigue crack growth models for GLARE. The overall results demonstrated very good NDI teardown capabilities and high fatigue damage tolerance by the GLARE structures. The window area cracks grew under variable amplitude loading, while the door beam cracks grew under almost constant amplitude loading. The window area cracks were too small to check model predictions, but a significantly longer door beam crack had a constant growth rate, which agrees with model predictions.

Contents

GLARE TEARDOWNS FROM THE MEGALINER BARREL (MLB) FATIGUE TEST

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INTRODUCTION

The MegaLiner Barrel (MLB) full-scale pressure cabin test was begun in the mid-1990s to study the fatigue behaviour of a double-deck transport aircraft configuration. As part of the Airbus A380 development programme, the MLB was used to investigate several design solutions, structural materials and joining methods, including the use of GLARE (GLAss REinforced aluminium laminates). The applied fatigue loads were set high enough to obtain fatigue damage. The test was done by Airbus Deutschland in Hamburg, Germany, and was discontinued after 45,402 simulated flights. Stork/Fokker Aerospace in Papendrecht, the Netherlands, then specified a teardown programme for GLARE panels and components from several key areas of the MLB. The NLR carried out this programme under contract to the Netherlands Agency for Aerospace Programmes (NIVR).

The present paper surveys the teardowns of a window area, a beam above the passenger door, and some stringer couplings. The teardowns began with Non-Destructive Inspection (NDI), and were followed by fractographic investigation of NDI-indicated cracks in the window and door beam locations. The general objectives were:

(1) Verification of teardown NDI techniques and capabilities.
(2) Establish the patterns of cracking in the GLARE aluminium layers.
(3) Determine the fatigue initiation locations and likely causes.
(4) Estimation of fatigue “initiation” lives and crack growth behaviour in the GLARE aluminium layers.
(5) Provision of data to check fatigue crack growth models for GLARE.

THE MLB

Figure 1 shows the MLB in the test hall in Hamburg, and the general loading conditions applied in the test rig. These were pressurization cycles combined with longitudinal and transverse bending and ground loads. Figure 2 shows a schematic of the MLB with key-codes of its construction and the GLARE locations investigated by the NLR. F4 is the window area, F6 is the stringer coupling area, and F7 is the passenger door beam area. The colour-shading codes refer to the GLARE and aluminium alloy skin materials. The GLARE, 2024 and 2524 panels were assembled with mechanical fasteners. The 6013 and 6056 panels were welded.

MLB FATIGUE LOAD SPECTRUM/HISTORY

The MLB fatigue load spectrum was defined by Wagner [1]. The spectrum was based on a 6.25 hour mission and included basic ground and flight loads, with incremental loads for taxiing, rotation, landing, vertical and lateral gusts, and coordinated turns. There were eight basic flight types, ranging from severe turbulence (A) to calm air (H). Flight type A was combined with one ground load condition, type B with two ground load conditions, and types C – H with three ground load conditions, resulting in a total of 21 flight types. These occurred with differing frequencies in a block of 2150 flights, which was repeated until the end of testing. Table I gives the positions of the severest flight types A – C in each flight block.

Table I Severest flights in the MLB fatigue load history

A	B	C
1379	58, 127, 196, 1094	139, 148, 366, 494, 671, 956, 1026, 1392, 1549, 1785, 1854, 1928

THE F4 WINDOW AREA

Construction

Figure 3 shows the F4 GLARE window area just after its removal from the MLB. There is a circumferential butt joint (C64, see figure 2) just to the right of the removed area. The basic structure of the window area was a GLARE 3-7/6-0.3/0.4 countersunk skin fastened to die forged 7175-T73 aluminium window frames by 4.76 mm press fit Hi-Loks. The GLARE code means seven 2024-T3 aluminium layers 0.3 mm or 0.4 mm thick and interleaved with six glass fibre layers 0.25 mm thick; the outer two aluminium layers were 0.4 mm thick.

Teardown procedure

Full details of the teardown and NDI are given in Refs. [2–4]. The teardown was done in several stages:

(1) NDI
- Removal of fasteners around windows and eddy current rotor inspection of fastener hole bores.
- Removal of window frames and eddy current pencil probe inspection of fastener holes in the GLARE skin and aluminium window frames on the faying surface sides.
- Disassembly of window frames and eddy current pencil probe inspection of aluminium rebates.

(2) Optical fractography
- NDI-indicated cracked fastener holes in the GLARE skin and window frames forcibly opened.
- Low-magnification fractography to verify and map fatigue cracks in the fastener holes.

(3) Field Emission Gun Scanning Electron Microscope (FEG-SEM) detailed fractography
- Fractography of the largest fatigue cracks in either window frame and the GLARE skin.

NDI results

216 fastener holes were inspected. 45 crack indications were obtained for the GLARE fastener hole bores in the partially disassembled condition, and 41 when completely disassembled. 8 crack indications were obtained for the aluminium window frames. The crack indications were almost equally divided between the windows. Figure 4 shows 21 crack indications for the GLARE skin and 3 crack indications for the aluminium window frame of the C65-C66 window. Both of the windows had most crack indications in the B and D quadrants. These were the quadrants largely subjected to tensile shear loads during the MLB test.

NDI verification

Opening up the GLARE fastener holes with crack indications revealed 4 false calls, which for 216 inspected holes is less than 2 %. Optical fractography measurements of the crack sizes were used in a Probability Of Detection (POD) analysis [5]. This showed that for the 90 % probability + 50 % confidence level the detectable crack length was only 0.25 mm [3–5], which is an excellent teardown NDI capability.

Optical fractography

Mapping the GLARE skin fatigue cracks showed that most were in the fastener hole bores, with only a few in the countersinks [3]. Figure 5 gives examples of the cracking patterns in the aluminium layers. The shapes of smaller cracks indicated that cracking generally began at aluminium layer corners, most probably because of the stress concentrations provided by the corners. However, this does not explain why most cracks were in the bores rather than the countersinks, where there were relatively severe “knife edge” stress concentrations. The explanation, confirmed by fractography of the window frame cracks, is that local secondary bending favoured cracking in the fastener hole bores.

FEG-SEM detailed fractography

This detailed fractography had several objectives:

(1) Check the "readability" of the MLB fatigue load history.
(2) Estimate and compare the fatigue " initiation" lives and crack growth behaviour of the largest "readable" fatigue cracks in either window frame and the GLARE skin.
(3) Provide data to check fatigue crack growth models for GLARE.

Load history "readability": Figure 6 gives an overview and a low-magnification detail of the largest crack in the window frames. Its position is arrowed in figure 4b. Higher magnifications showed crack front markings due to severe simulated flights, and the "readability" was generally very good. Figure 7 is an example of identifying the severest flight types A, B and C.

Fractographic analysis of the largest window frame crack: Figures 8a and 8b show the crack lengths, a and a*, perpendicular to the fastener hole bore, plotted against the number of simulated flights (N) and the crack growth rates, da/dN, where a* is the mean crack length for each growth interval used to calculate da/dN. Both figures show the effects of severe simulated flights, pointed out explicitly in figure 8b. Some effects appeared to be transient, but a significant period of crack growth retardation began at a = 0.6 mm. This could be due to termination of a "short crack effect". Such effects are generally attributed to a lack of fatigue crack closure in cracks smaller than about 0.5 mm [6]. Thus it is likely that once the window frame crack grew beyond about 0.5 mm the peak loads in severe simulated flights caused closure-induced retardation.

Back-extrapolation of the data in figure 8a suggests a fatigue "initiation" life of zero and an initial crack size of about 0.06 mm. In fact, the fatigue crack began at a fretting scar caused by fastener movement against the bore of the hole, see figure 6b. Since fretting rapidly induces fatigue-initiating damage [7] and promotes early crack growth [8], it explains the (effectively) zero "initiation" life and also the suggestion of an initial crack size.

Fractographic analysis of the largest "readable" GLARE skin crack: The position of this crack is arrowed in figure 4a. The crack was 0.91 mm long. Preliminary examination showed that the initial 0.2 mm was obscured by debris and sealant. Figures 9a and 9b show the crack lengths, a and a*, perpendicular to the fastener hole bore, plotted against N and da/dN. The data are limited but sufficient to show a maximum crack growth rate about 50 % of that in the aluminium window frame at similar crack lengths.

Back-extrapolation of the data in figure 9a is unfeasible, owing to the limited data. Hence an estimate of the fatigue "initiation" life was not possible. Also, this crack (and, of course, all the other GLARE skin cracks) was too small to provide a check on fatigue crack growth models for GLARE: see, for example, the models proposed by De Koning [9], Alderliesten and Woerden [10], Beumler [11], Alderliesten and Homan [12] and Alderliesten [13]. However, on the positive side this teardown result demonstrates the high fatigue damage tolerance capability of the GLARE skin.

THE F7 DOOR BEAM AREA

Construction

Figure 10 shows the F7 GLARE door beam area before removal of a rectangular sample to be pulled to failure. The basic structure of the door beam area was a GLARE 3-9/8-0.4 countersunk skin reinforced with seven GLARE doublers to make a total of 34 aluminium layers. The GLARE skin code means nine 2024-T3 aluminium layers 0.4 mm thick and interleaved with eight glass fibre layers 0.25 mm thick. The doublers also consisted of 2024-T3 aluminium layers 0.4 mm thick and interleaved with glass fibre layers 0.25 mm thick.

Teardown procedure

The teardown was done in several stages:

(1) NDI: Removal of fasteners and eddy current rotor inspection of fastener hole bores by Airbus Deutschland.
(2) Sample testing: Removal of the rectangular sample and pulling it to failure.
(3) Optical examination
- Fractography for fastener hole 33.
- Fastener holes 31 – 35.

(4) FEG-SEM examination
- Fractography of the largest fatigue crack in fastener hole 33.
- Fastener holes 31 – 35.

NDI results

The Airbus Deutschland results are in Ref. [14]. The largest crack indication was 7 mm in fastener hole 33.

Rectangular sample testing

The sample indicated in figure 10 was pulled to failure in a 900 kN machine. The arrows in figure 10 point to the failure position, and figure 11 shows this position after failure. The sample failed across fastener holes 12 and 33, with some fibre pullout and delamination. Note the cracks at the nearby fastener holes 11, 32 and 34. These cracks were made visible (opened up) by the testing. The sample breaking load exceeded the Limit Load (LL) requirement despite the prior presence of fastener hole cracks.

Optical examination

Fractography for fastener hole 33: Figure 12 shows the best visible fracture surfaces. The aluminium layer fatigue cracks generally appeared silvery-white, although there were blackish areas near the fastener hole bore and on much of the fatigue fracture surfaces of the bottom aluminium layer. The overall contours of the fatigue cracks indicate a strong influence of local bending. This bending was due to pressurization-induced bulging of the door cutout area during the MLB fatigue test [14]. The largest fatigue crack was in aluminium layer 31 (the first layer of the last doubler), see the right-hand image in figure 12. The crack length was 6.54 mm, which is close to the Airbus Deutschland NDI indication of 7 mm.

Fastener holes 31 − 35: The left-hand image in figure 12 shows considerable black debris and/or damage in the bore of fastener hole 33. Consequently some nearby fastener holes were sectioned perpendicular to the visible or NDI-indicated cracks. All the holes had circumferential scoring (grooves) suggesting poor initial hole quality [15].

FEG-SEM examination

Fractographic analysis of the largest crack in fastener hole 33: The fatigue fracture surface was first checked for “readability”. This was generally excellent: most of the fracture surface, *including the very beginning*, see figure 13, was covered with uniformly spaced fatigue striations. There were occasional larger striations, but the load history was essentially constant amplitude cycling. This must have been due to the predominance of pressurization loads at the F7 location.

As a first approximation the uniform striation spacings were taken to represent the crack growth in each simulated flight. This meant obtaining an a versus da/dN plot first, and then deriving the a versus N plot, i.e. the reverse procedure to that for the F4 window area cracks. Figure 14 shows the derived a versus N plot based on a fatigue “initiation” life of zero. The trend line for the data points shows that the estimated total life is too long by about 18 %. This is most probably because severe simulated flights (occasional larger striations, ignored in the first approximation) accelerated the overall crack growth. Consequently, figure 14 also includes a corrected trend line.

Figure 15 is a compilation of the a versus N results for the F7 door beam and F4 window area cracks. There are two main points to make:

- The data for the door beam crack shows the original a versus da/dN plot and a trend line derived from the corrected plot in figure 14. There is an evident trend of nearly constant crack growth rates. Bearing in mind that the load history was almost constant amplitude cycling, this result is an encouraging validation of the more sophisticated crack growth models for GLARE [9, 10, 13].
- There are significant differences in crack growth rates for the F7 and F4 locations and also differences in the trends. These differences are attributable to differing structural geometries, local load levels and load histories, and also different materials (GLARE compared to a monolithic aluminium alloy).

Fastener holes 31 − 35: Detailed viewing of the cracked fastener hole bores was enabled by the FEG-SEM, owing to its capability of operating at low kV to minimise charging-up the non-conducting glass fibre layers. Figure 16 gives an example of the observed damage in the bores. The following characteristics of cracking were ascertained:

- Fatigue cracking began in the aluminium layers, generally at or near corners. However, the higher magnification micrograph in figure 16a shows a kinked rack that initiated both at a corner and heavy scoring. This proves that (a) the scoring was present during the MLB fatigue test and was not a result of fastener removal before the NDI by Airbus Deutschland, and (b) the initial hole quality was indeed poor.
- As sequential aluminium layers became through-cracked the intermediate glass fibre layers began to protrude into the hole bores.

- More severe cracking of the aluminium layers led to cracking of the interleaved glass fibre layers as well as further protrusion.
- Black debris in the hole bores, see the left-hand image in figure 12, was most probably due to cyclic displacements of the protruding glass fibre layers during the MLB fatigue test. In other words, the debris was a kind of fretting product.

These results suggest it is difficult to drill good quality fastener holes in a thick GLARE laminate. For the F7 door beam area this meant that fatigue crack growth in some fastener holes began as soon as the MLB test commenced. Even so, the longest fatigue crack was less than 7 mm at the end of the test. Since the applied fatigue load history was set at a conservatively high level, this teardown result once again demonstrates the high fatigue damage tolerance capability of GLARE.

THE F6 STRINGER COUPLING AREA

Construction

Figure 17 is a schematic of part of the F6 stringer coupling assembly. The materials used in this assembly were:

(1) 2524-T351 aluminium skin, 5.25 mm thick, panel B1.
(2) GLARE 4A-5/4-0.4 skin, 4.15 mm thick, panel D1. The GLARE code means five 2024-T3 aluminium layers 0.4 mm thick and interleaved with four glass fibre layers 0.25 mm thick.
(3) GLARE 2B-10/9-0.4 butt strap, 6.25 mm thick. The Glare code means ten 2024-T3 aluminium layers 0.4 mm thick and interleaved with nine glass fibre layers 0.25 mm thick.
(4) 7349-T7651 aluminium, 5.25 mm thick, for the forward stringers.
(5) 4 × GLARE 2A-2/1-0.3 bonded by 3 × 0.15 mm adhesive layers, total thickness 3.85 mm, for the aft stringers. The GLARE code means two 2024-T3 aluminium layers 0.3 mm thick and interleaved with one glass fibre layer 0.25 mm thick.
(6) 3 thicknesses, 5.85 mm, 4.85 mm, 2.85 mm, of GLARE 2A-2/1-0.3, decreasing in two steps outwards from the coupling mid-points, bonded by 0.15 mm adhesive layers. As before, the GLARE code means two 2024-T3 aluminium layers 0.3 mm thick and interleaved with one glass fibre layer 0.25 mm thick.

Teardown procedure

The teardown was done for NDI only:

- Removal of fasteners to enable disassembly of the GLARE stringers and butt strap from the skins and the stringer couplings from the stringers, butt strap and skin. The aluminium stringers and skin could not be separated because they had been adhesively bonded as well as mechanically fastened.
- Eddy current rotor inspection of fastener hole bores for the skins, butt strap, stringers and stringer couplings.
- Eddy current pencil probe inspection of faying surfaces (not possible for the aluminium skin/stringer faying surfaces) to verify the eddy current rotor inspection and estimate crack lengths.

Full details of the teardown and NDI are given in Ref. [16]. A summary of the NDI results is given next.

NDI results

Figure 18 classifies the NDI-indicated crack lengths for the components of the F6 stringer coupling assembly, with the exception of the 2524-T351 aluminium skin, for which there were 4 crack indications. There were many crack indications for the GLARE components, but they were all less than 4.5 mm. However, two aluminium stringers contained cracks with indicated lengths of 6.3 mm and 7 mm [16]. These differences are consistent with GLARE's susceptibility to fatigue crack "initiation" in the aluminium layers of the laminates, but its increasing resistance to crack growth, owing to fibre bridging [12].

Support for the above interpretation comes from comparing the NDI-indicated crack lengths for GLARE and aluminium stringer couplings. The maximum indicated crack length for the GLARE stringer couplings was 3.4 mm after 45,402 simulated flights [16]; but aluminium stringer couplings gave fifteen indications with crack lengths 6 – 25 mm after about 20,000 simulated flights, and another twenty-four indications with crack lengths 6 – 16 mm after less than 40,000 simulated flights [14].

CONCLUDING REMARKS

Teardowns of GLARE structures from three key locations of the MLB have demonstrated very good NDI capabilities and high fatigue damage tolerance by the GLARE structural features. The NDI capabilities were verified for the F4 window area by opening up crack-indicated fastener holes in the GLARE skin, measuring the crack sizes, and carrying out a POD analysis.

The high fatigue damage tolerance of the GLARE structural features is inferred from (a) the NDI-indicated and measured crack lengths in GLARE fastener holes from the F4, F6 and F7 locations, (b) their comparison with NDI-indicated crack lengths in aluminium alloy components, and (c) the applied fatigue load history being set at a conservatively high level. The GLARE crack lengths were less than 7 mm after 45,402 simulated flights, when the MLB fatigue test was discontinued. However, some of the aluminium alloy components, notably stringer couplings, had NDI-indicated crack lengths up to 25 mm after "only" about 20,000 simulated flights.

The high fatigue damage tolerance of GLARE is due to fibre bridging, which causes increasing resistance to crack growth in the aluminium layers of GLARE. This was indirectly shown by the largest crack in the F7 door beam sample. This crack experienced almost constant amplitude loading and had a nearly constant growth rate, which agrees with the more sophisticated model predictions that account for fibre bridging and its interaction with delaminations at the interfaces between aluminium and glass fibre layers.

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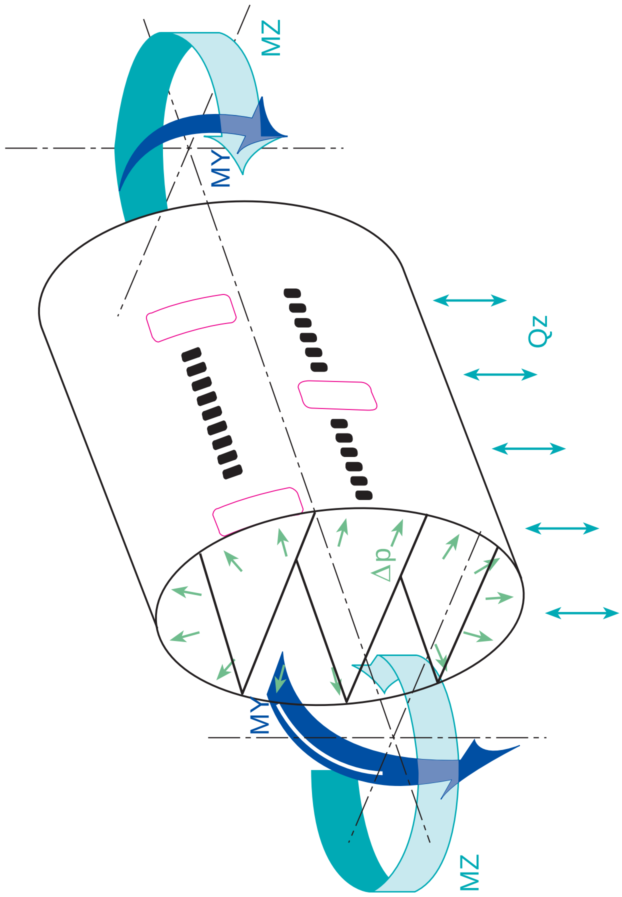


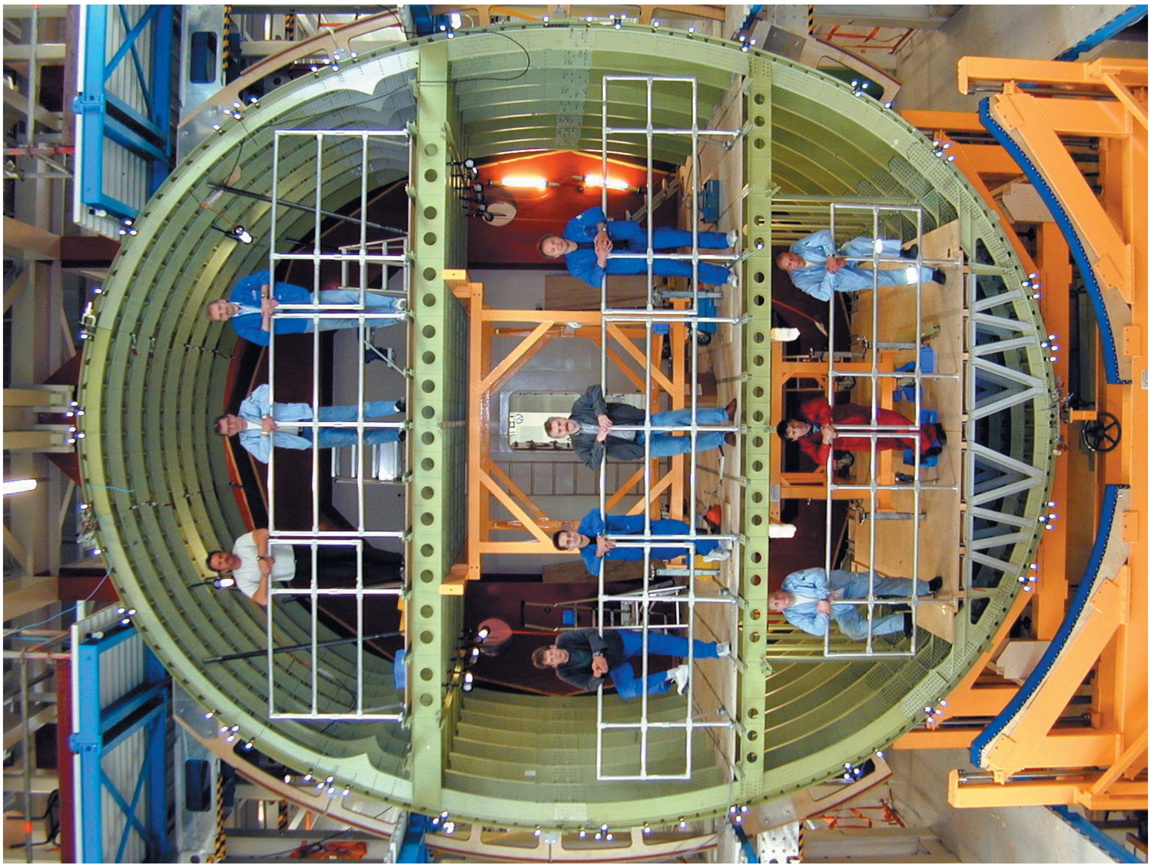


Figure 1 The MLB configuration and general loading conditions

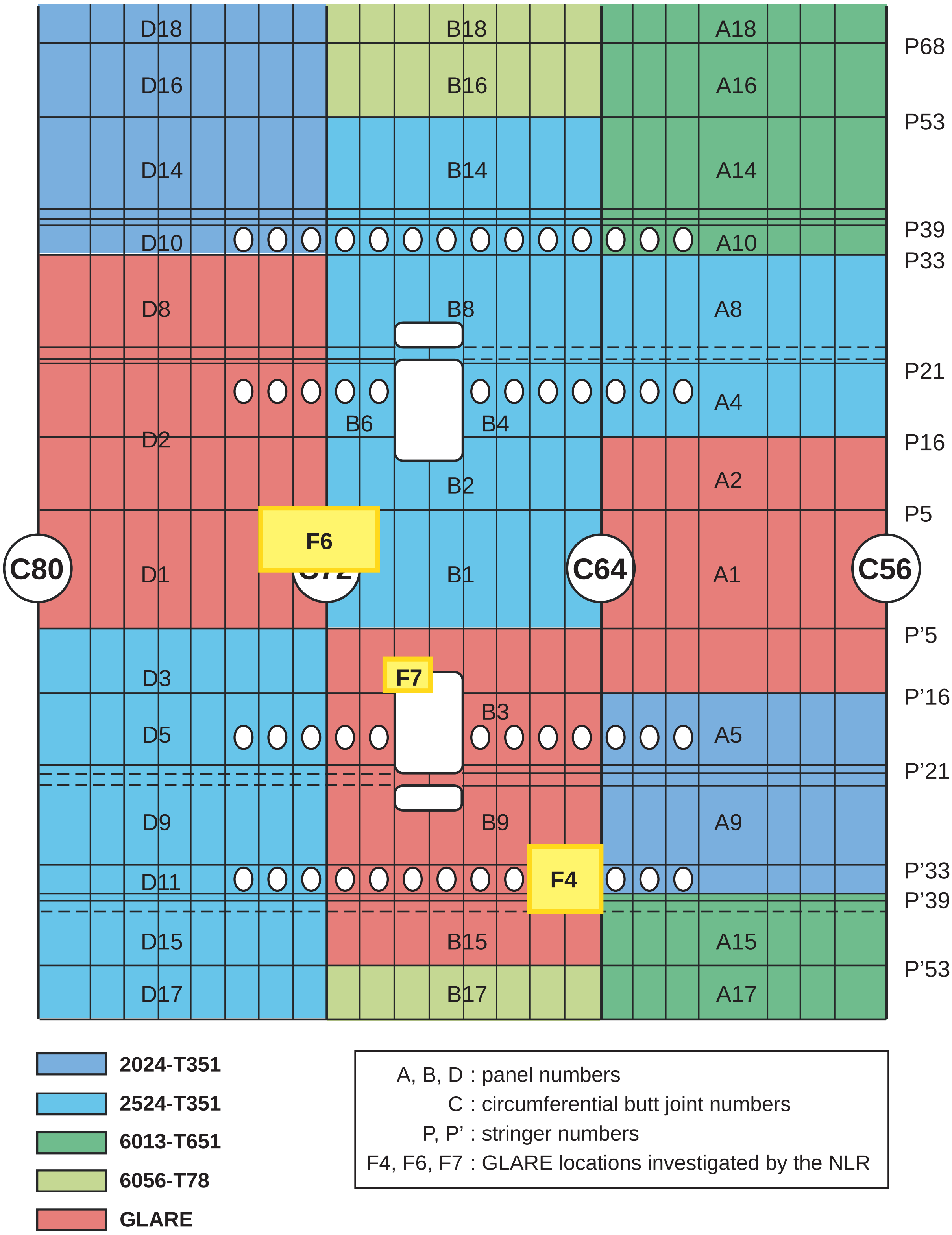


Figure 2 "Opened out" view of the MLB

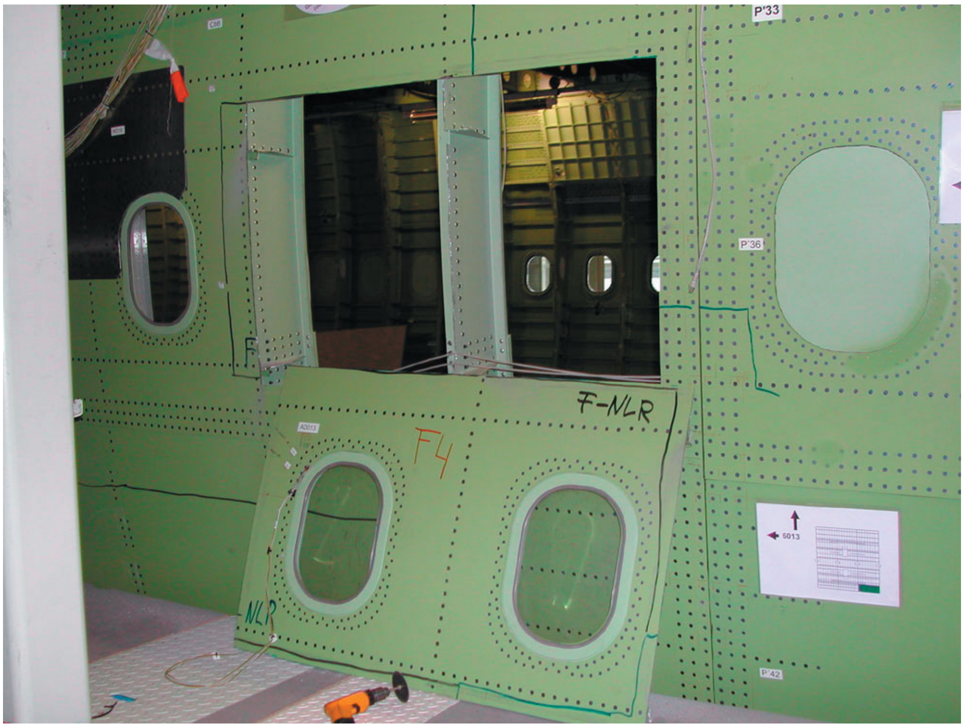


Figure 3   The F4 GLARE window area just after its removal from the MLB

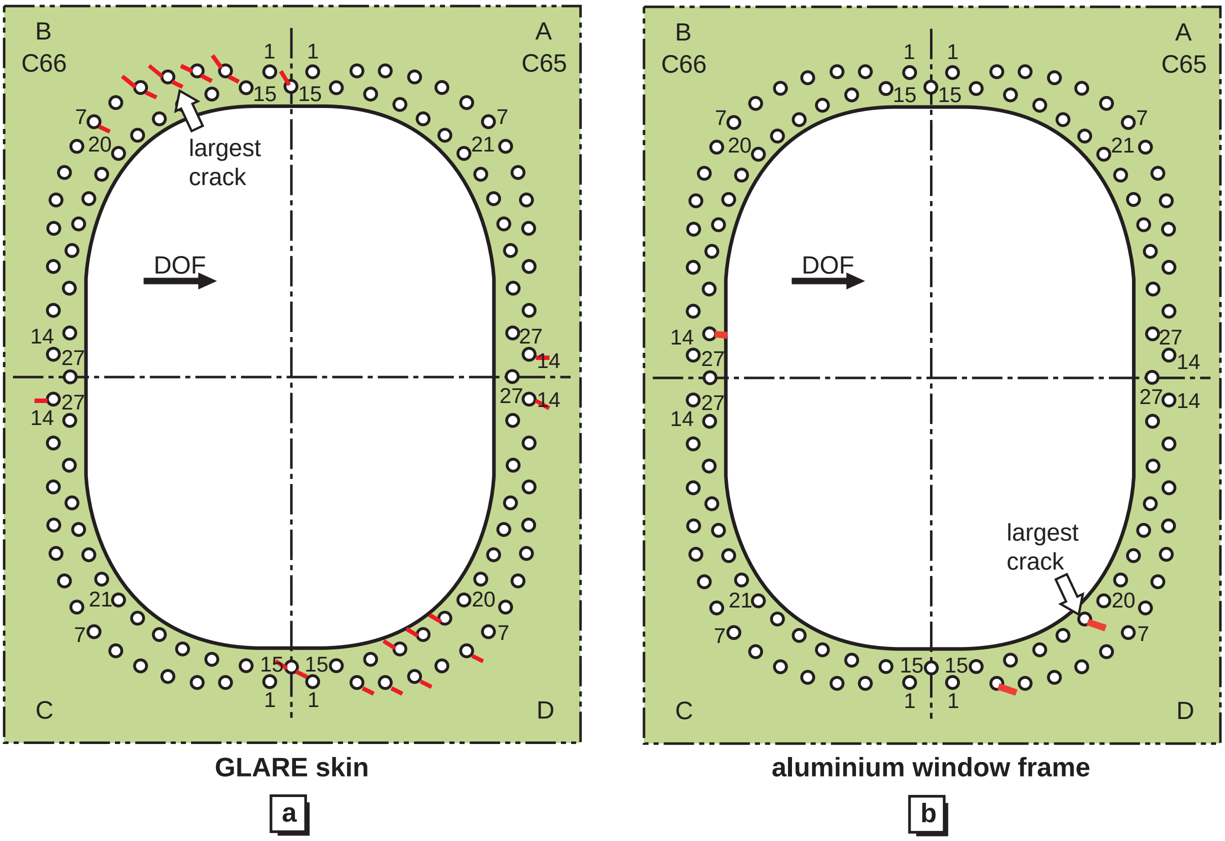


Figure 4   NDI crack indications for the GLARE skin and aluminium window frame fastener hole bores of window C65-C66: DOF = Direction Of (simulated) Flight

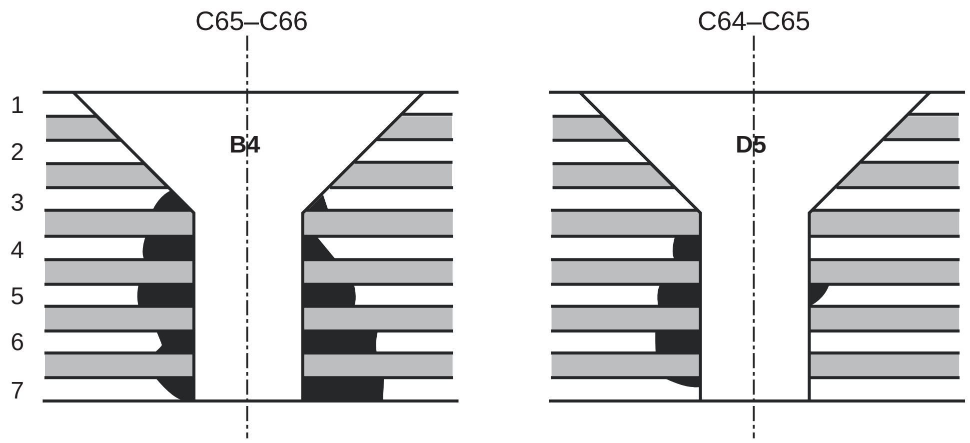


Figure 5 *Examples of fatigue cracking patterns in the GLARE aluminium layers*

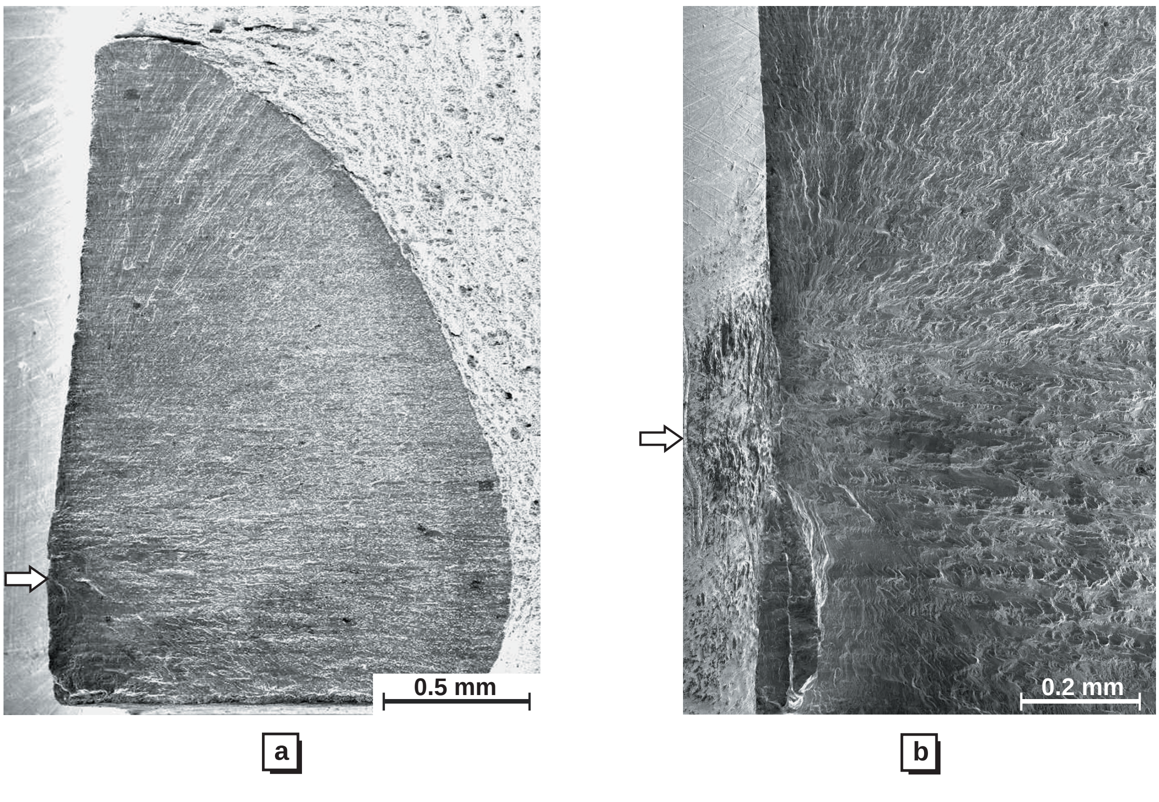


Figure 6 *Largest window frame fatigue crack: (a) overview with fatigue origin arrowed; (b) detail with fretting scar in fastener hole bore arrowed*

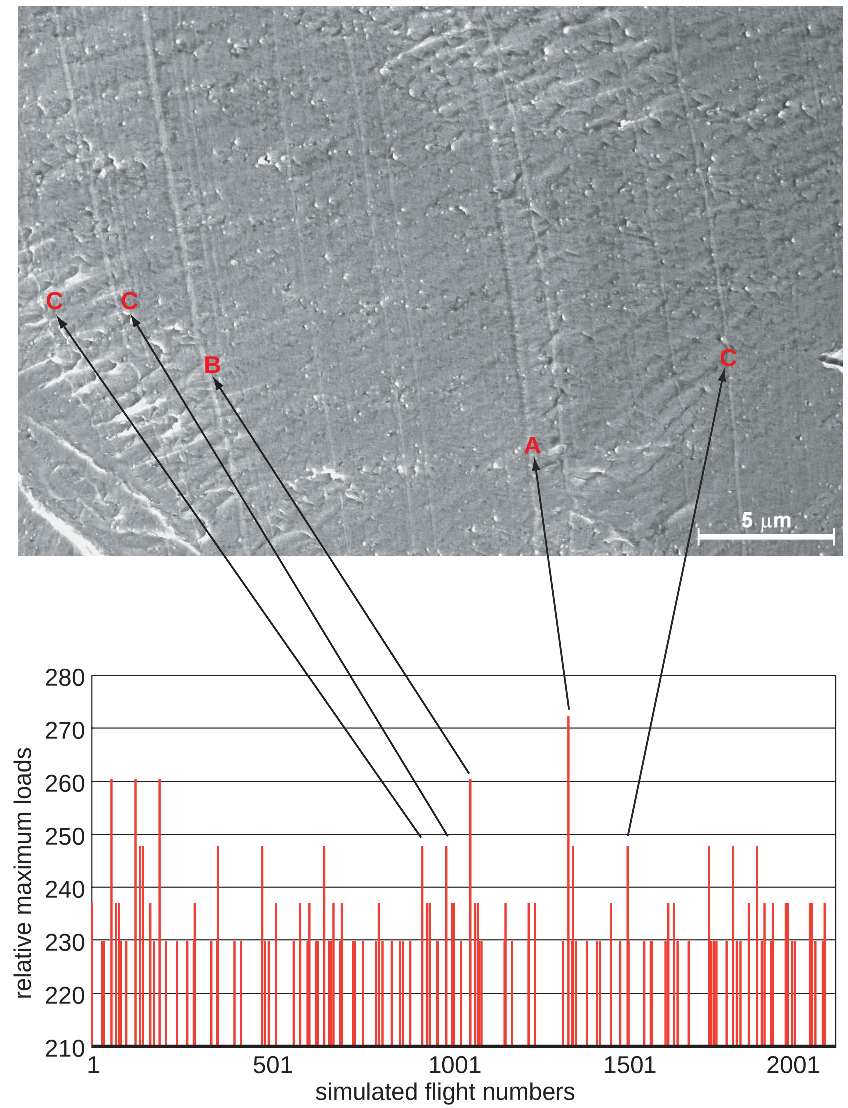


Figure 7 Example of identifying severe simulated flights from crack front markings due to peak loads

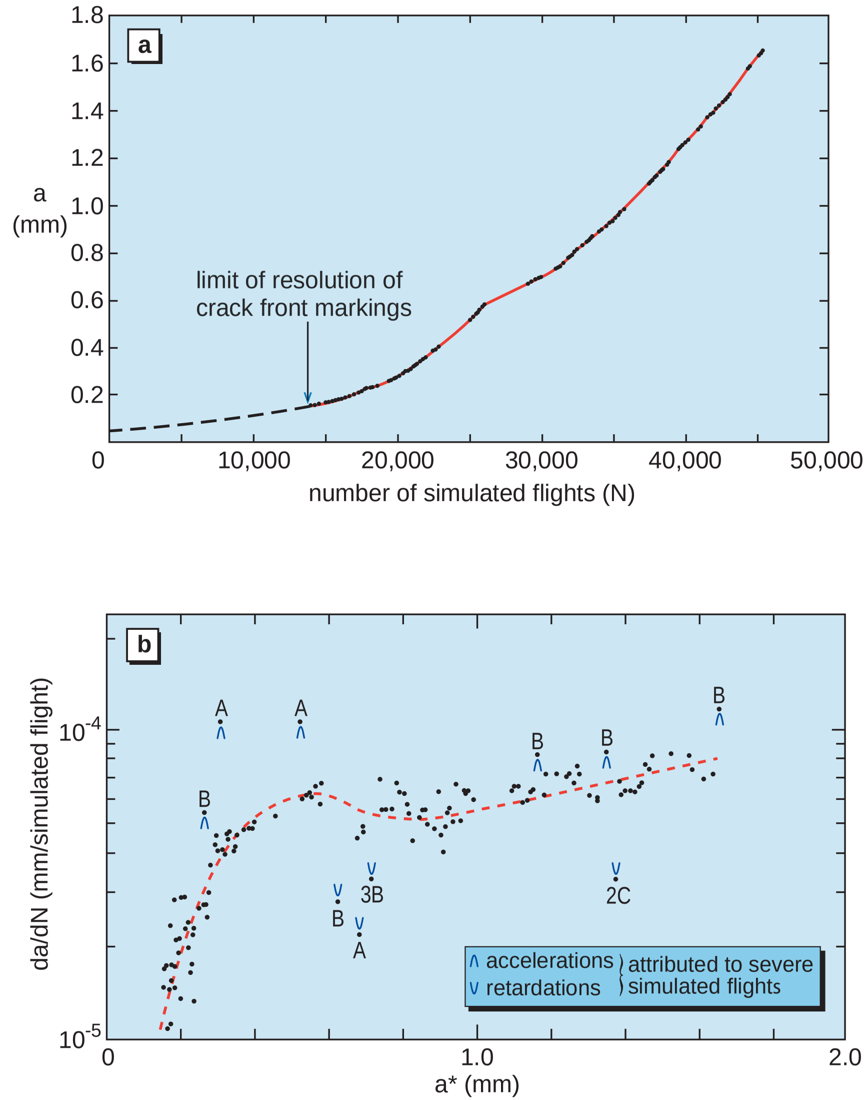


Figure 8 Crack growth curves for the largest window frame crack. The material was die forged aluminium alloy 7175-T73

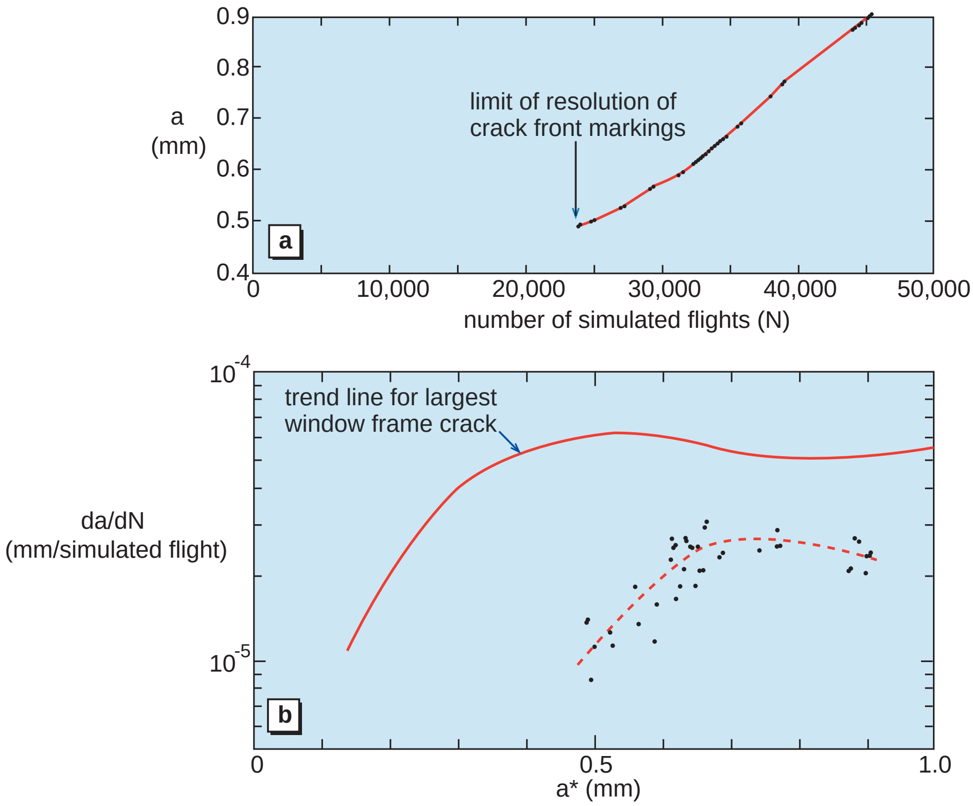


Figure 9 Crack growth curves for the largest “readable” GLARE skin crack

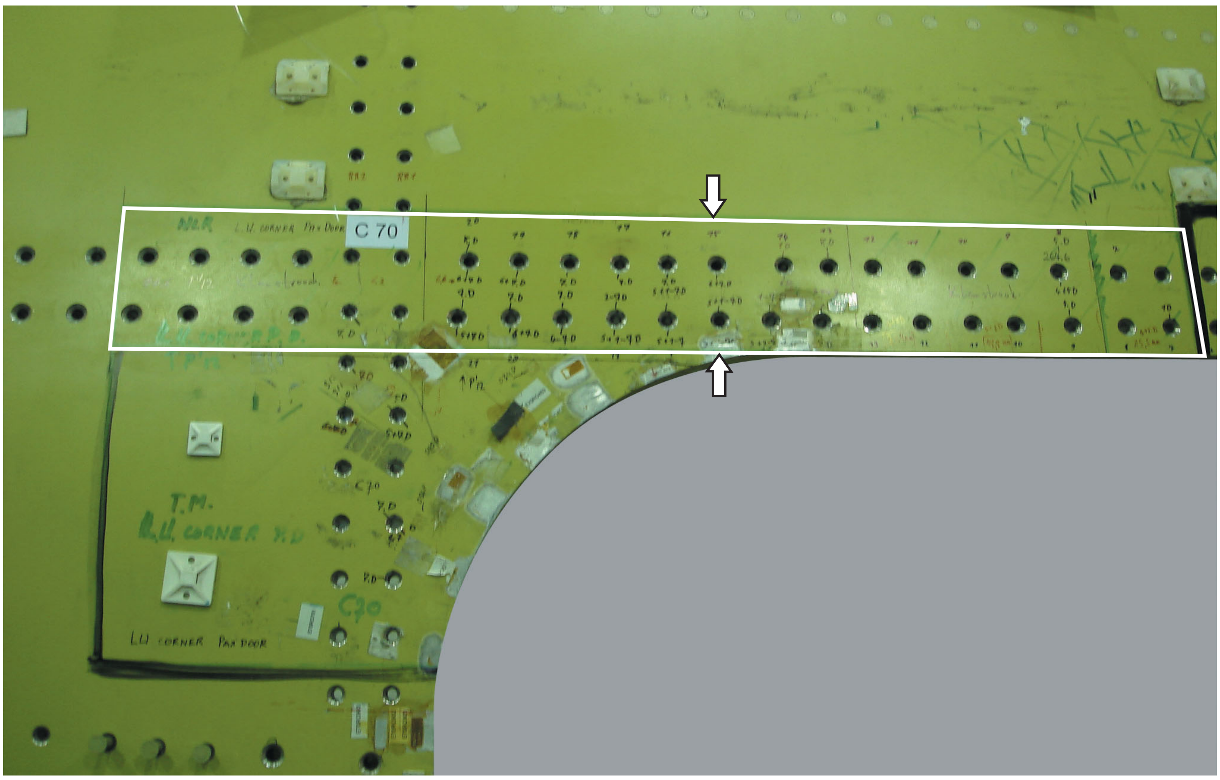


Figure 10 The F7 GLARE door beam area before removal of the rectangular sample indicated by the white border. The arrows point to the subsequent failure position

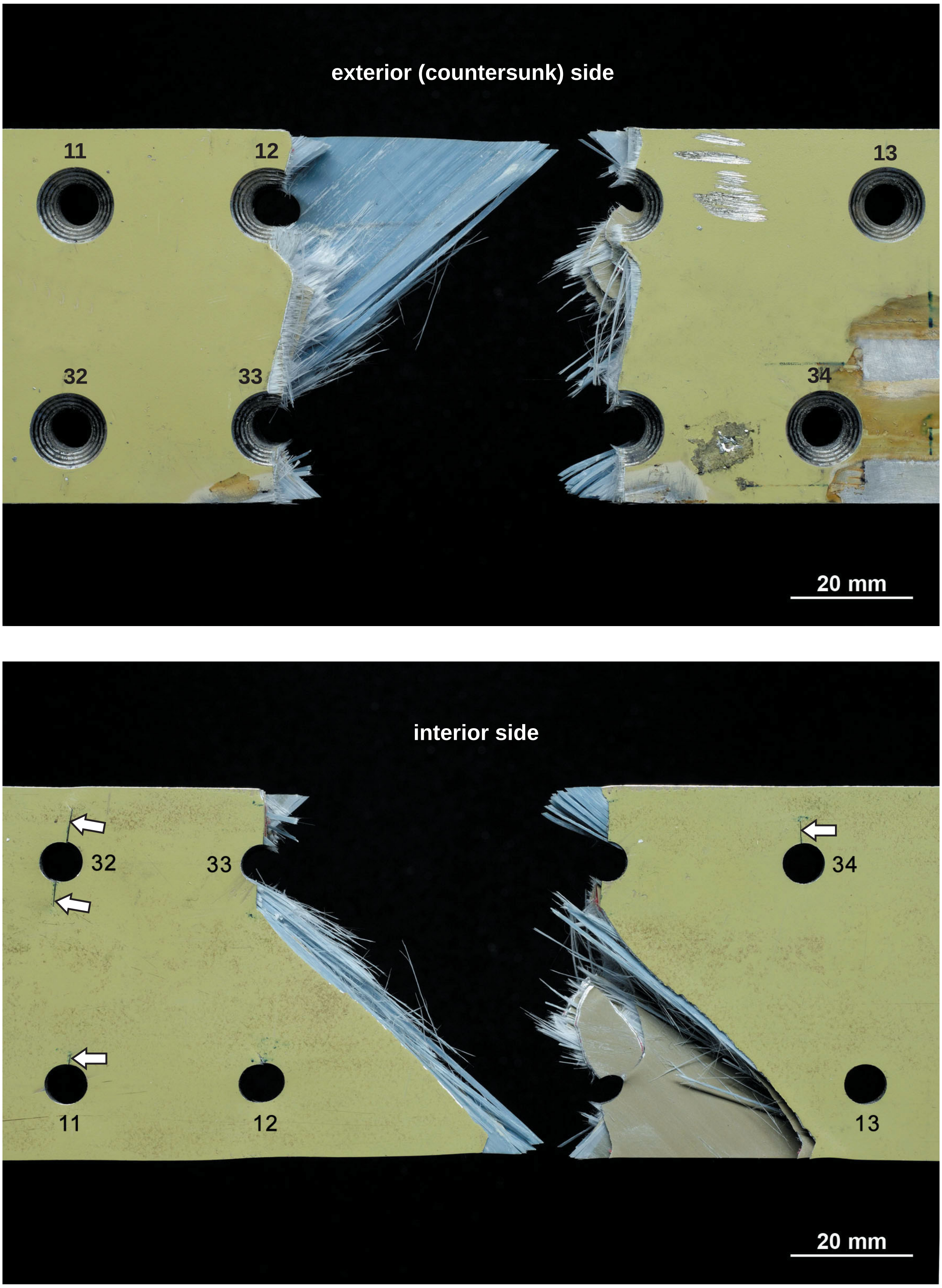


Figure 11 Failure position of the rectangular door beam sample. Arrows in the lower photograph point to cracks at the nearby fastener holes 11, 32 and 34

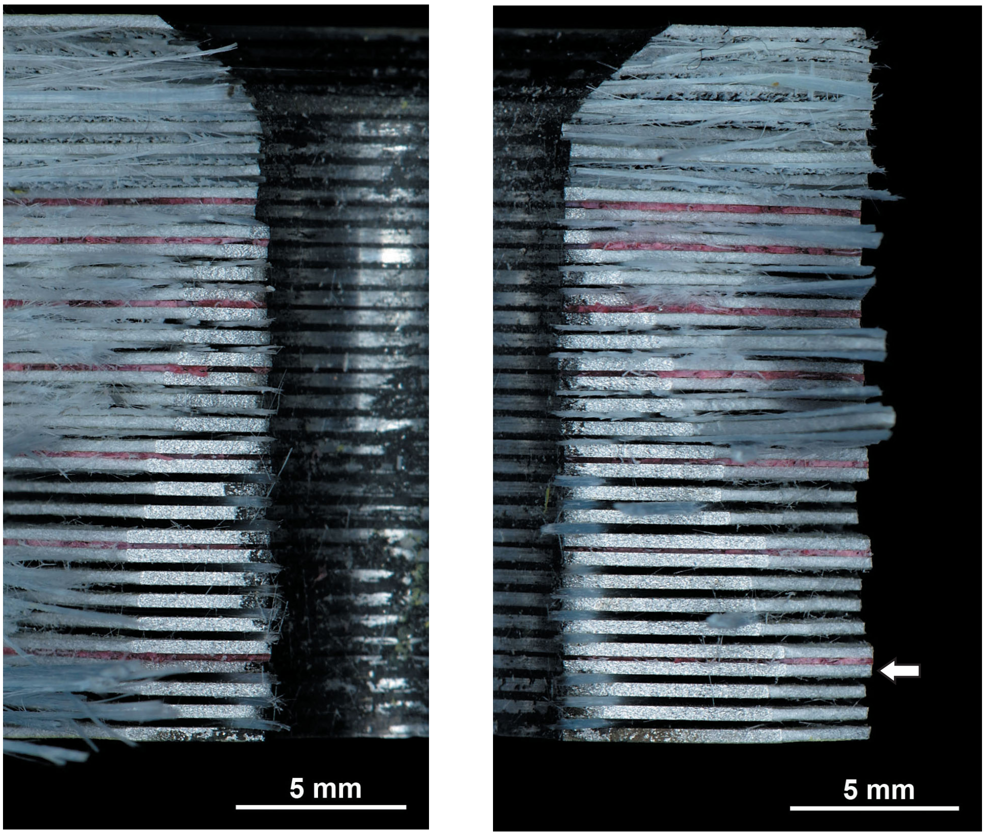


Figure 12 Macrofractographs for fastener hole 33. The arrow points to the largest fatigue crack

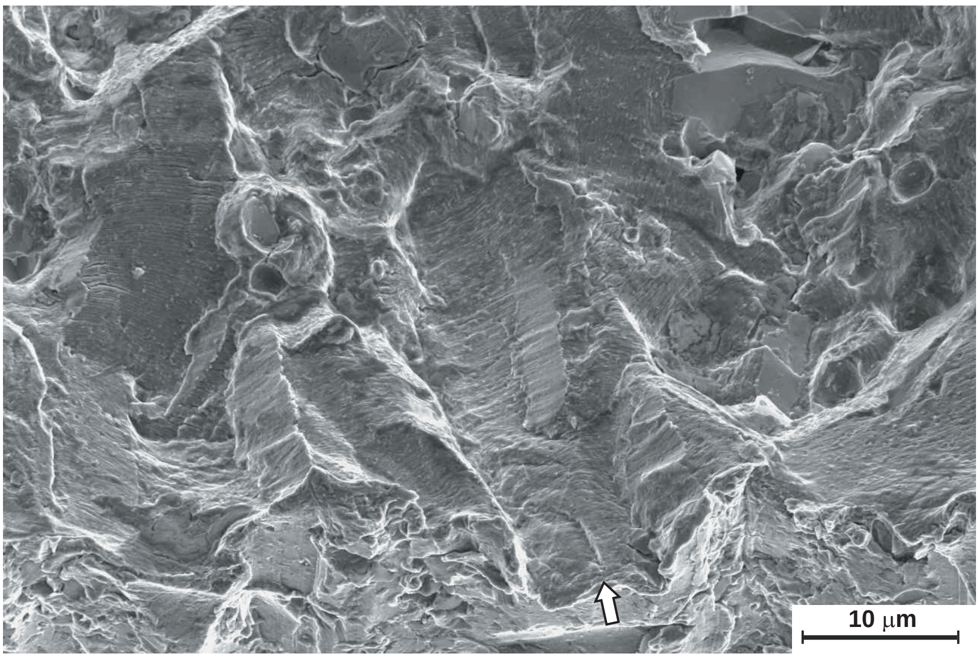


Figure 13 Fatigue crack growth (striations) commencing directly from the bore of fastener hole 33. The arrow points to a fatigue striation "plateau"

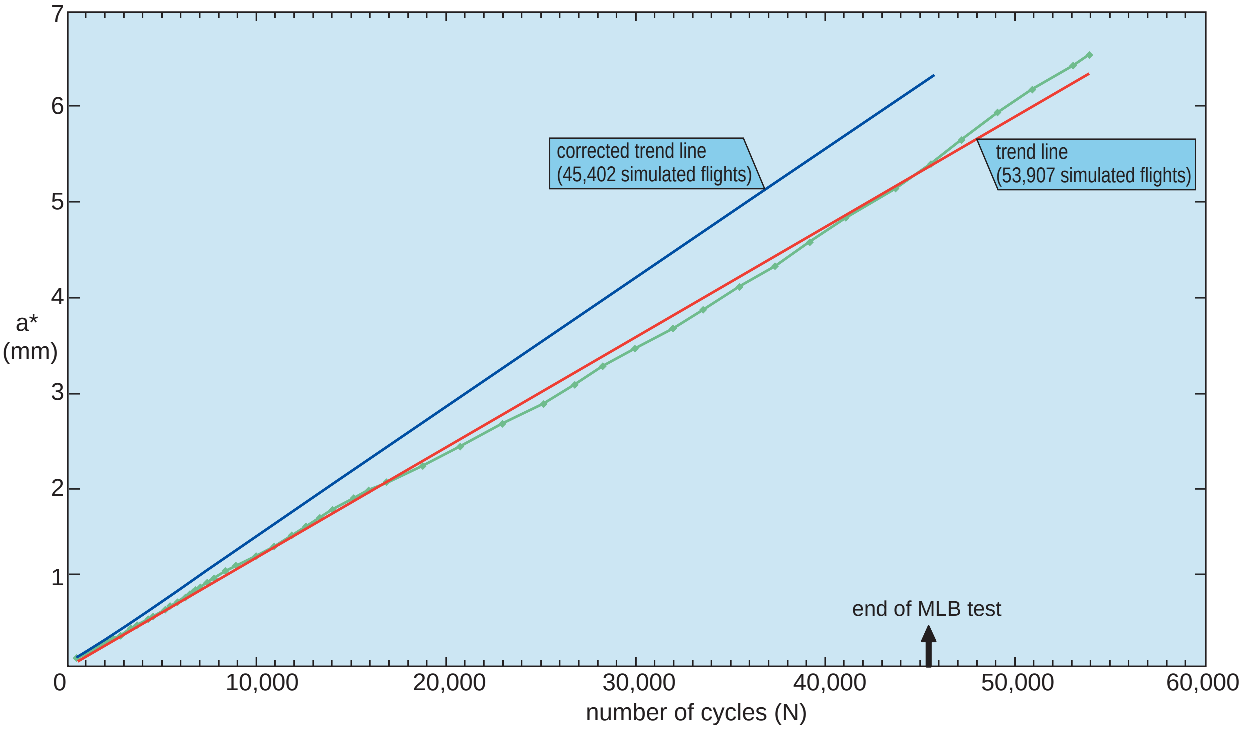


Figure 14 Crack growth plot, a versus N, derived from fatigue striation spacings for the longest crack in the door beam GLARE sample*

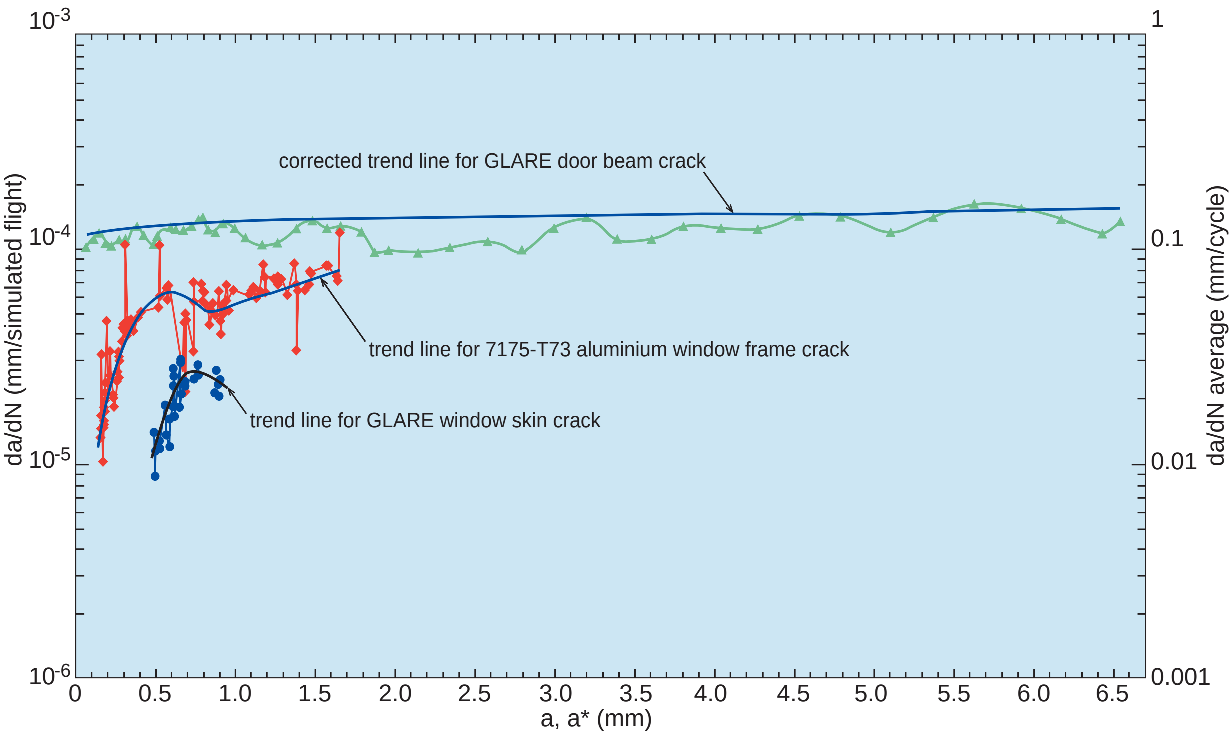


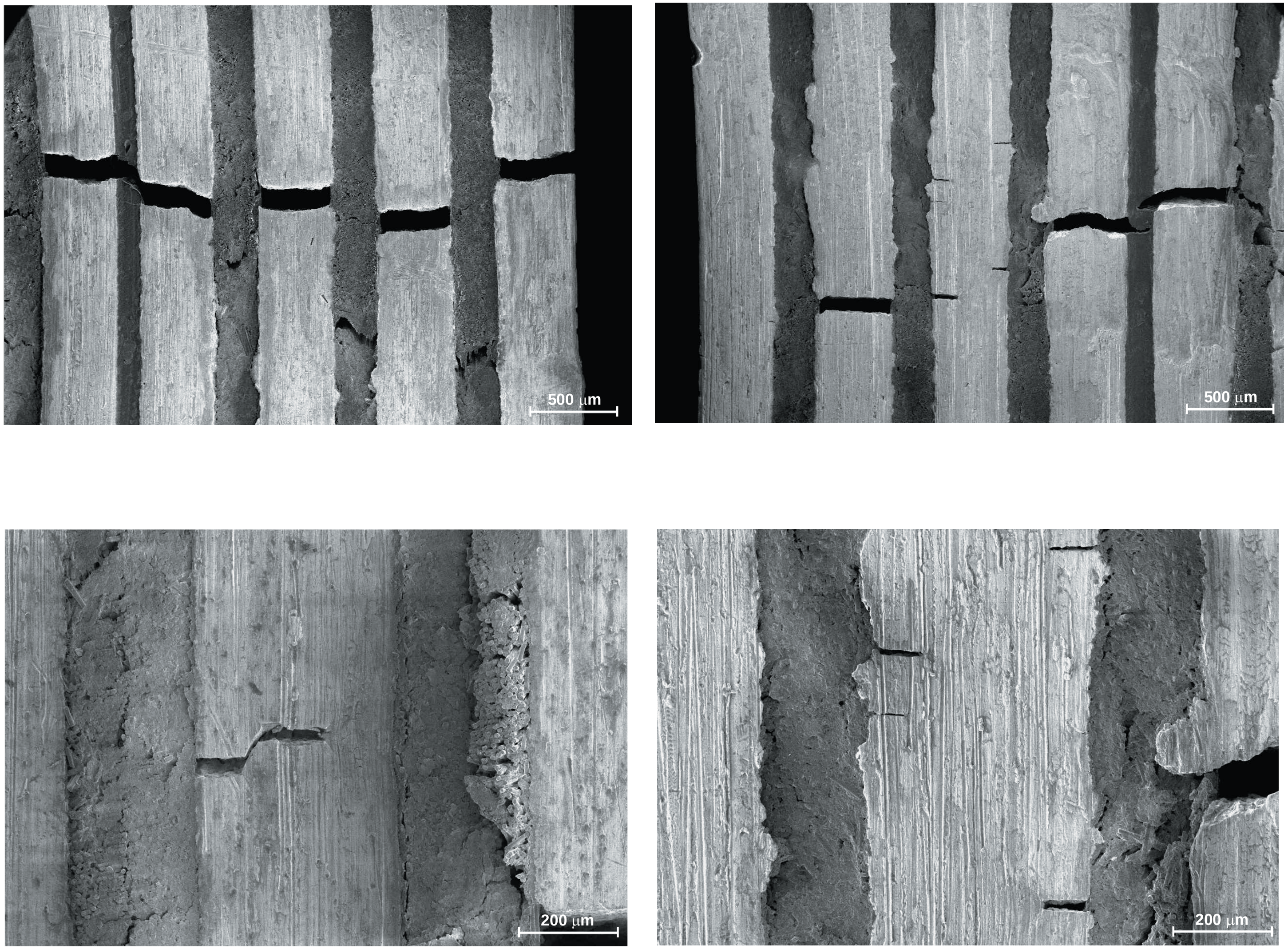
Figure 15 Comparison of crack growth rates for the longest cracks in the F7 and F4 locations

severe cracking

- glass fibre layers starting to protrude and crack

less severe cracking

- fatigue crack "initiation" often at aluminium layer corners
- occasional heavy scoring on aluminium surfaces leading to crack "initiation" away from corners



a

b

Figure 16 Example of fatigue cracking and damage in the bore of fastener hole 34

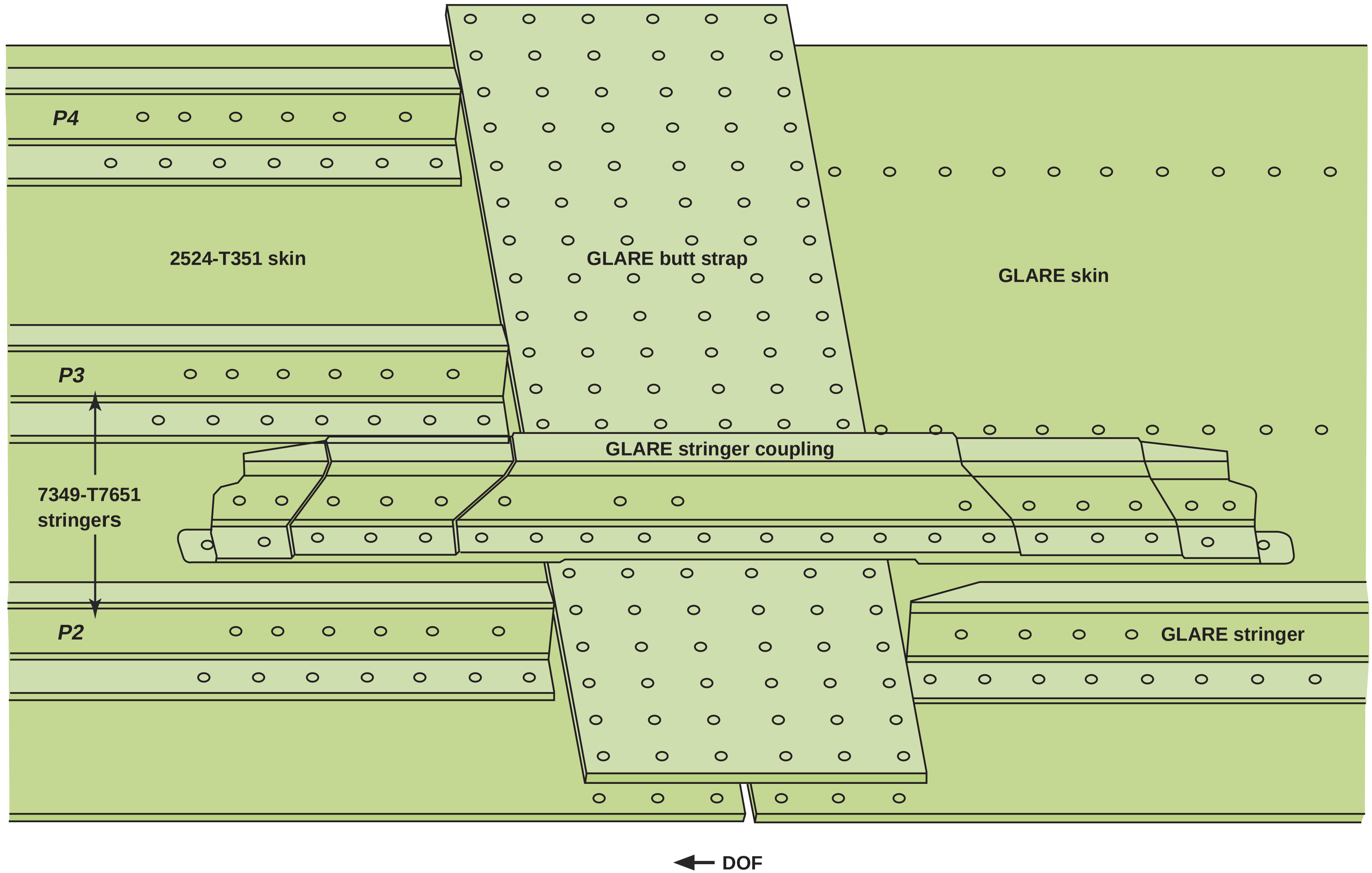


Figure 17 Schematic of partly disassembled F6 stringer coupling assembly: DOF = Direction Of (simulated) Flight

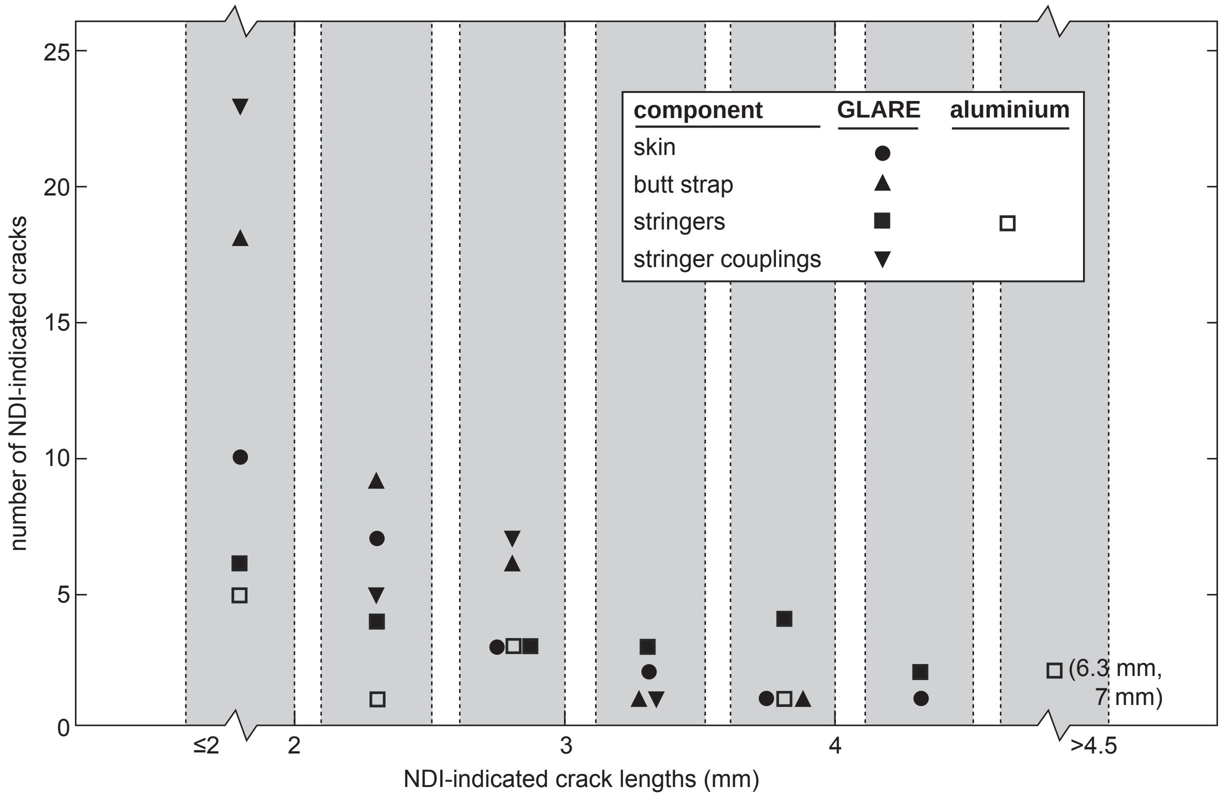


Figure 18 Classification of NDI-indicated cracks for the F6 stringer coupling assembly