Fatigue, static tensile strength and stress corrosion of aircraft materials and structures

Part II: figures

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Part II: figures

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\[ K_t \approx K_{t1} \times K_{t2} = 3 \times 3 = 9 \]

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$K_t$ 3 3.86 5.10 6.06

$1 + 2\sqrt{\frac{a}{\rho}}$ 3 3.83 5.00 5.90

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\[ C = \frac{K}{SV\pi a} \]

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Figure 4.9. Edge crack in a semi-infinite sheet.

Fig. 4.10. Small edge crack at notch root.
Fig. 4.11. Crack edge loading.

Case I: $K = \frac{P}{\sqrt{\pi a}}$

Case II: $K = \frac{P}{\sqrt{\pi a}} \left( \frac{\pi \frac{2a}{W}}{\sin \left( \frac{\pi}{2a} \right)} \right)^{1/2}$

Case III: $K = \frac{P}{\sqrt{\pi a}} \left( 1.297 - 0.297 \cos \left( \frac{\pi a}{W} \right) \right) \left( \frac{\pi \frac{2a}{W}}{\sin \left( \frac{\pi}{2a} \right)} \right)^{1/2}$

Figure 4.12. Finite width effect on crack edge loading.
Fig. 4.13. The stress intensity factor for a crack under internal pressure (case 3).

\[ K_1 = K_2 + K_3 \]

\[ K_3' = K_3 \]

\[ K = 2K_3 = K_4 + K_5 \]

\[ K_1 = K_2 + \frac{1}{2} (K_4 + K_5) \]

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Circular crack radius $a$

$$K = \frac{2}{\pi} S \sqrt{\frac{\pi a}{c}}$$

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Pure Al.
After 5000 cycles ($2\varepsilon_a = 0.5\%$) only three slip lines were visible. A static load until $\varepsilon = 7.6\%$ opened the three slip lines (see picture) which showed the slip lines to be cracks. All other slip lines were produced by the static loading.

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0.3 μm = 1000 interatomic distances

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- ridge due to overlapping cracks

Fig. 7.11d: Fracture surface with growth bands, due to alternating low and high $\sigma_a$. (low: $\sigma_a = 64$ N/mm$^2$, 60000 cycles; high $\sigma_a = 103$ N/mm$^2$, 9000 cycles).

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\[ \frac{p}{a} = 0.1 \]
\[ K_t \approx 8 \]

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<table>
<thead>
<tr>
<th>Test</th>
<th>N (cycles)</th>
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<tbody>
<tr>
<td>without blocks</td>
<td>8000000</td>
</tr>
<tr>
<td>with blocks</td>
<td>100000</td>
</tr>
<tr>
<td>blocks removed after 12000 cycles</td>
<td>1300000</td>
</tr>
<tr>
<td>removed after 30000 cycles</td>
<td>100000</td>
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<table>
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<tr>
<th>Alloy</th>
<th>σ₀.2 (N/mm²)</th>
<th>σₚ (N/mm²)</th>
<th>δ</th>
</tr>
</thead>
<tbody>
<tr>
<td>2024-T3 Alclad</td>
<td>364</td>
<td>474</td>
<td>19%</td>
</tr>
<tr>
<td>7075-T6 clad</td>
<td>464</td>
<td>520</td>
<td>9%</td>
</tr>
</tbody>
</table>

Figure 8.22: Crack growth data da/dn-ΔK, comparison between two Al-alloys and between two heat treatments of the same alloy.

<table>
<thead>
<tr>
<th>Alloy</th>
<th>σ₀.2 (N/mm²)</th>
<th>σₚ (N/mm²)</th>
<th>δ</th>
</tr>
</thead>
<tbody>
<tr>
<td>2024-T3 Alclad</td>
<td>365</td>
<td>468</td>
<td>21%</td>
</tr>
<tr>
<td>2024-T8 Alclad</td>
<td>456</td>
<td>478</td>
<td>6.5%</td>
</tr>
</tbody>
</table>
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\( D = \text{damage} = \frac{\text{consumed energy}}{n/N} \)

9.2a

Miner: same linear relation for all \( \sigma_a \)

\[ \sum n/N = 1 \]

9.2b

Shanley: same non-linear relation for all \( \sigma_a \)

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thickness of steel bush $\geq 0.1d$

steel bush with interference fit in Al-alloy lay

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undersized hole
diameter too large
tapered pin

Fig. 10.17a:
Before pulling.

plastic zone with high residual compressive stresses

Fig. 10.17b:
After pulling operation.

CYLINDRICAL PART
TAPER ANGLE

COLD EXPANSION MANDREL
PULLER
PULLER NOSE PIECE
LUBRICATED SLEEVE
FOR THE MANDREL OF THE PRESENT TEST SERIES:
CYLINDRICAL PART 4.5 mm
TAPER ANGLE 1:45

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Figure 10.21a

Figure 10.21b

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All sheets and straps of equal thickness.
Calculations for $\sigma_{\text{tension}}/E = 0.001$, bending factors do not change very much at higher $\sigma$

Figure 10.28: Results of calculations on secondary bending (formulas from Ref. 15).

$G_a$ at $N=10^6$ ($R=0.4$)

a  
\[ t=1\text{mm} \]  
15 N/mm$^2$

b  
23 N/mm$^2$

c  
27 N/mm$^2$

d  
\[ 6\text{mm} \]  
32 N/mm$^2$

e  
40 N/mm$^2$

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B-747 REMOUS

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2 3 3 4 4 3 6 4 2 8 4 4 3 3 1 2 3 2 5 3 7 4 3 6 4 4 2 4 2

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Fig. 11.9: The range-pair count method.
Figure 11.10: Power spectrum of a stochastic Gauss process. Different shapes.

Figure 11.11: A large Fourier series as an approximation of random load.

Figure 11.12: Different types of random load with corresponding power spectral density functions and irregularity factors $K = N_1/N_0$ (Ref. 3).
Figure 11.13: Flight profile of a short flight of the F-28 split up in a number of intervals (Ref. 5).

Figure 11.15: Periods and numbers of several types of aircraft fatigue loads (Ref. 6). Orders of magnitude.
Fig. 11.14a: Deterministic loads.

Fig. 11.14b: Stochastic load superimposed on deterministic load.

Fig. 11.14c: Time-compressed flight-simulation.
10 different types of flight, A to K. A = severe storm, K = very nice weather.

\[ S_{mf} = \text{mean stress in flight} \]

**Fig. 11.16a:** Sample of a history applied in flight-simulation tests according to the F-28 gust spectrums for the wing (Ref. 8).

**Fig. 11.16b:** Sample of a load history applied in flight-simulation tests according to a fighter manoeuvre spectrum for wing bending (Ref. 9).

Figure 11.16: Samples of load-time histories applied in flight-simulation tests.
Figure 11.17: Gust load spectrum applied in flight-simulation tests (Ref. 10). Gust amplitudes relative to mean stress in flight ($S_{mf}$).

<table>
<thead>
<tr>
<th>Specimen</th>
<th>$\sigma_{mf}$ [N/mm$^2$]</th>
<th>Truncation level (fig. 11.17)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Open hole specimen (diameter 20 mm, $K_t=2.56$)</td>
<td>69</td>
<td></td>
</tr>
<tr>
<td>Lug (hole diameter 25 mm, $K_t=2.8$)</td>
<td>98</td>
<td></td>
</tr>
<tr>
<td>Riveted strap joint Snap rivets</td>
<td>69</td>
<td></td>
</tr>
<tr>
<td>Riveted strap joint Counter Sunk rivets</td>
<td>69</td>
<td></td>
</tr>
</tbody>
</table>

Figure 11.18: Larger fatigue lives in flight-simulation tests at higher truncation levels (Ref. 10).
Figure 11.19: Effect of truncation level on crack growth rate under flight-simulation loading. Central crack in 160 mm wide specimen, t = 2 mm (Ref. 11).

Figure 11.20: Symmetric and asymmetric wing bending spectra (civil transport aircraft and fighter aircraft).
Fig. 12.1: Problem areas, disciplines involved and significant aspects of aircraft fatigue.
Fig. 12.2c: Example of a multiple-load path structure

Fig. 12.2b: Example of a multiple-element component

stiffeners not indicated

4 planks (skin + stiffeners)

spanwise joint connecting 2 planks

Fig. 12.2c: Spanwise joints for crack arrest.

Fig. 12.2d: Fuselage with circumferential crack stopping bands attached to the skin.

Figure 12.2: Some examples of fail-safe design.
Figure 12.3: Milestones for crack detection in operational service.

Figure 12.4: Comparison between crack growth lives in solid plate and in laminated sheet material (2024-T3 Alclad) (Ref. 2).
cross section

<table>
<thead>
<tr>
<th>Al-alloy</th>
<th>fiber/adhesive</th>
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<tr>
<td>Al-alloy</td>
<td>fiber/adhesive</td>
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<tr>
<td>Al-alloy</td>
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</tr>
</tbody>
</table>

0.3 or 0.4 mm

0.2 mm

typically

Al-alloy: 2024-T3, 7075-T6, ........
Number of thin sheets: 2, 3 or more
Fibers: Aramid (-ARALL) or R-glass (-GLARE)
Adhesive: AF163-2 (epoxy) or other types of adhesives.

Figure 12.5: Composition of ARALL.

Figure 12.6: Schematic picture of fatigue crack bridging fibers.

Figure 12.7: Fracture mechanics picture of the ARALL principle.
Figure 12.8: Fatigue crack growth in a central crack sheet specimen under TWIST flight-simulation loading. Comparison of ARALL with 2024-T3. Note the higher mean stress in flight \( S_{mf} \) for ARALL. Effect of prestraining ARALL (Ref. 5).

Figure 12.9: Unfavourable residual stress system after curing is reversed into a favourable system by plastic prestraining.
Figure 12.10 A fatigue curve (A) with two extensions (B and C) and two load spectra (H and H') used in illustrative calculations.

Figure 12.11 Three procedures for predicting fatigue lives in the design phase (Ref. 1)
Figure 12.12 Some fatigue test load sequences, main variables and testing purposes (Ref. 1).

Fig. 12.13a Diverging S-N curves.

Fig. 12.13b Intersecting S-N curves.

Figure 12.13 In conventional fatigue tests the result of comparing design A and design B depends on the choice of the fatigue load amplitude.
Figure 12.14: The effect of limit load application on crack propagation in a full-scale wing structure under random flight-simulation loading (Ref. 7)
Wing loading:
gust cycles, constant amplitude,
calculated to give some fatigue
damage as full gust spectrum

pressurization cycle
on fuselage

ground-to-air cycle

1 flight
same load history
in all flights

Figure 12.15 "Simplified" load history, applied in early flight-simulation tests on a complete aircraft structure.

load amplitude

scatter band of fatigue strength of helicopter blades

average

early failure

non failure

Figure 12.16. Safety factor on life of non-failure result is meaningless.
MIL-A-83444 initial damage

2 options: inspectable/non-inspectable
3 types of structures: SCG = Slow Crack Growth
MLP = Multiple Load Path
CAFS = Crack Arrest Fail Safe

non-inspectable → requirements on life
inspectable → requirements on crack growth period

Figure 12.17: Information of the USAF damage tolerance requirements