Blunt and sharp notch behaviour of Glare laminates

T.J. de Vries
Blunt and sharp notch behaviour of Glare laminates

Proefschrift

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When I started my Ph.D. study in 1994, it was impossible to foresee how
the future of Glare would develop. Although Glare was already investigated
for ten years, the Glare “community” was rather small and the scepticism in
the aerospace industry remained a big hurdle. Only with the help of a
relatively small group of extremely enthusiastic and innovative people, we
kept on promoting this new material. Now, 7 years later, Glare will be
applied on the new aeroplane of the 21st century, the Airbus A380, an
enormous success for this small group!

This book describes only a part of this Glare research and would not have
been finished without the help of my family, many friends and colleagues
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Abstract

The present thesis describes the behaviour of Glare laminates in the presence of holes or blunt notches and through-the-thickness cracks or sharp notches. These laminates, which are bonded arrangements of thin aluminium layers that are connected by glass fibre reinforced epoxy (prepreg) layers, are now ready for industrial applications. It is most likely that they will be applied in the Airbus A380.

Blunt notches like fasteners holes are difficult to avoid in a structure and decrease the static and dynamic properties of the material. The presence of blunt notches in Glare material has a large impact on the remaining strength of the material. The failure mechanism of blunt notches in Glare has been investigated and described for different Glare laminates, hole configurations (including filled holes) and load combinations. For uni-axial loaded holes a prediction method is derived based on two material parameters: the Metal Volume Fraction (MVF) and the Fibre Volume Fraction (FVF) of the fibres in loading direction. For more complex loading conditions, a 3D fracture surface is proposed based on the ultimate strain of the fibres and the aluminium alloy. This calculation method yields conservative results since delamination is not taken into account.

Sharp notches or cracks are the results of damage. For Glare, a distinction must be made between damage caused by fatigue where only the metal layers are cracked, and through-cracks where both the metal and fibre layers are cracked. The residual strength of Glare laminates with fatigue cracks and fibres in loading direction is superior compared to aluminium alloys. This thesis focussed on the behaviour of through-cracks. Like for the blunt notch research, much attention is given to understand the fracture mechanism. The influences of many different parameters like the characteristics of the constituent properties, the bonding quality, the thickness of the aluminium layers, etc have been investigated. This test program illustrated the applicability of the $K_{ir}$-curve approach as a material parameter. A calculation method is derived that is based on the MVF and the FVF in loading direction of a specific laminate. Based on the amount of load carrying layers and a derived ratio between the different layers, a multiplication factor is obtained which is to be multiplied with a standard $K_{ir}$-curve for 2024-T3 to obtain the laminate curve.

Additionally, the blunt and sharp notch behaviour of spliced laminates has been investigated. For the current splice configuration based on the self forming technique and overlapping aluminium layers, the blunt notch
behaviour is not influenced by the presence of the splice. The sharp notch behaviour of a Glare laminate can even be improved based on the splice location. It is concluded that the blunt and sharp notch behaviour of spliced laminates can be regarded as Glare laminates without splices.

A study towards the occurrence of WFD from a row of holes with initial flaws turned out to be very unlikely for Glare within the DSG of an aeroplane, especially when the holes are filled with rivets. Since linkup of these fatigue cracks does not occur, the very unlikely event of a FOD in a row of rivet holes has been investigated. It turned out that it is possible to predict linkup from the FOD crack tip to the holes (and initial flaws with fatigue cracks) in front of this lead crack and final fracture can be predicted rather accurately with the $K_{IC}$-curve approach.

Finally, a study has been carried out towards the effectivity of extra fibre bands or titanium doublers in Glare as crack stopper solutions. It turned out that the use of extra glass fibres perpendicular to the crack direction gives better results. The residual strength of Glare laminates with glass fibre crack stoppers can be predicted with the derived calculation method. This is verified by large-scale fuselage tests containing two-bay cracks.
List of symbols

A  cross section
a  half crack length (total crack length in CT specimen)
    half notch width
B  matrix
b  half notch height
    half MSD crack length
a, b, c  constant factors in second order equation
d  cross sectional area
    diameter
e  distance between hole and edge
E  Young’s modulus
F  Force
f  frequency
G  shear modulus
    crack driving force or energy release rate
g, h  combination of material parameters
i k  counter
Ki  mode I stress intensity factor
Kt  stress concentration factor
L  specimen length between clamps
M  curve fit for residual strength as function of T
N  cycles
n  number
P  force
p  rivet / hole pitch
pv  pixelvalue
ps  pixel
R  crack resistance
    Stress ratio $\sigma_{\text{min}} / \sigma_{\text{max}}$
r  notch root radius
    Irwin plastic zone radius
s_d  splice distance
s_p  splice pitch
t  thickness
    ligament length
v_c  cruise speed
v_t  fibre volume fraction
u, v, w  displacements in x, y and z direction respectively
x, y, z  Cartesian co-ordinate
    centre locations
List of indices

\( Y \)  distance between COD gauges
\( W \)  width
\( \alpha \)  thermal expansion coefficient
curve fit factor
\( \beta \)  correction factor
\( \delta \)  partial derivative
\( \varepsilon \)  strain
\( \phi, \varphi, \theta \)  angle
\( \rho \)  density
\( \sigma \)  tensile or compressive stress
\( T \)  function that combines MVF and FVF\(_{LD}\)
\( \tau \)  shear stress
\( \mu \)  variable
\( \nu \)  Poisson's ratio
\( \psi \)  shear strain

List of indices

AE  acoustic emission
b  bearing
bn  blunt notch
c, cr  critical
cs  crack stopper
det  detectable
e  engineering or apparent
eff  effective
fl, fr  fatigue left / fatigue right
fin  finite width
G  crack driving force
gross  gross section
i  index
isotropic
initiation
inf  infinite width
max  maximum
net  net section
nom  nominal / net section
o  orthotropic
<table>
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<th>Abbreviation</th>
<th>Description</th>
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<tr>
<td>o</td>
<td>orthotropic</td>
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<tr>
<td>phys</td>
<td>physical</td>
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<tr>
<td>pr</td>
<td>prepreg</td>
</tr>
<tr>
<td>R</td>
<td>crack resistance</td>
</tr>
<tr>
<td>sq</td>
<td>squeeze</td>
</tr>
<tr>
<td>tot</td>
<td>total</td>
</tr>
<tr>
<td>ult</td>
<td>ultimate</td>
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<tr>
<td>y</td>
<td>yield</td>
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<tr>
<td>0</td>
<td>initial</td>
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**List of abbreviations**

<table>
<thead>
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<th>Abbreviation</th>
<th>Description</th>
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<tbody>
<tr>
<td>AB</td>
<td>Anti Buckling (guides)</td>
</tr>
<tr>
<td>AC</td>
<td>Advisory Circular</td>
</tr>
<tr>
<td>Al</td>
<td>Aluminium</td>
</tr>
<tr>
<td>APU</td>
<td>Auxiliary Power Unit</td>
</tr>
<tr>
<td>ARALL</td>
<td>ARamid Aluminium Laminate</td>
</tr>
<tr>
<td>CA</td>
<td>Constant Amplitude</td>
</tr>
<tr>
<td>CCD</td>
<td>Charged Coupled Device (camera)</td>
</tr>
<tr>
<td>CCT</td>
<td>Centre Cracked Tension (specimen)</td>
</tr>
<tr>
<td>CCU</td>
<td>Camera Control Unit</td>
</tr>
<tr>
<td>CFRP</td>
<td>Carbon Fibre Reinforced Plastics</td>
</tr>
<tr>
<td>CGL</td>
<td>Crack Growth Life</td>
</tr>
<tr>
<td>COD</td>
<td>Crack Opening Displacement</td>
</tr>
<tr>
<td>CT</td>
<td>Compact Tension (specimen)</td>
</tr>
<tr>
<td>CTOA</td>
<td>Crack Tip Opening Angle</td>
</tr>
<tr>
<td>DOF</td>
<td>Direction Of Flight</td>
</tr>
<tr>
<td>DSD</td>
<td>Discrete Source Damage</td>
</tr>
<tr>
<td>DSG</td>
<td>Design Service Goal</td>
</tr>
<tr>
<td>ESG</td>
<td>Extended Service Goal</td>
</tr>
<tr>
<td>FAR</td>
<td>Federal Aviation Regulations</td>
</tr>
<tr>
<td>FE</td>
<td>Finite Element</td>
</tr>
<tr>
<td>FML</td>
<td>Fibre Metal Laminate</td>
</tr>
<tr>
<td>FOD</td>
<td>Foreign Object Damage</td>
</tr>
<tr>
<td>FVF</td>
<td>Fibre Volume Fraction</td>
</tr>
<tr>
<td>Glare</td>
<td>Glass reinforced</td>
</tr>
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<td>Impress</td>
<td>Image processing residual strength</td>
</tr>
<tr>
<td>Abbreviation</td>
<td>Definition</td>
</tr>
<tr>
<td>--------------</td>
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<tr>
<td>IPP</td>
<td>Image Pro Plus (software package)</td>
</tr>
<tr>
<td>IPS</td>
<td>Image Pixel Size</td>
</tr>
<tr>
<td>JAR</td>
<td>Joint Airworthiness Requirements</td>
</tr>
<tr>
<td>L</td>
<td>Longitudinal</td>
</tr>
<tr>
<td>LL</td>
<td>Limit Load</td>
</tr>
<tr>
<td>LP</td>
<td>Load path</td>
</tr>
<tr>
<td>MSD</td>
<td>Multiple Site Damage</td>
</tr>
<tr>
<td>MVF</td>
<td>Metal Volume Fraction</td>
</tr>
<tr>
<td>NDI</td>
<td>Non Destructive Inspection</td>
</tr>
<tr>
<td>NSY</td>
<td>Net Section Yielding</td>
</tr>
<tr>
<td>PSE</td>
<td>Principal Structural Elements</td>
</tr>
<tr>
<td>RD</td>
<td>Rolling Direction</td>
</tr>
<tr>
<td>SEM</td>
<td>Scanning Electron Microscope</td>
</tr>
<tr>
<td>SF</td>
<td>Safety factor</td>
</tr>
<tr>
<td>SFT</td>
<td>Self Forming Technique</td>
</tr>
<tr>
<td>SIF</td>
<td>Stress Intensity factor (K)</td>
</tr>
<tr>
<td>T</td>
<td>longitudinal - Transverse</td>
</tr>
<tr>
<td>UD</td>
<td>Uni Directional</td>
</tr>
<tr>
<td>UL</td>
<td>Ultimate Load</td>
</tr>
<tr>
<td>VA</td>
<td>Variable Amplitude</td>
</tr>
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<td>WFD</td>
<td>Widespread Fatigue Damage</td>
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1 INTRODUCTION

1.1 The development of a new aeroplane material; Glare

The Second World War left the Dutch aeroplane industry Fokker with a destroyed production plant. While rebuilding the plant, Fokker could not afford the enormous investments required for large milling machines to produce integrally stiffened panels. Therefore, a different structural concept was adopted to tailor the local strength: laminated structures with bonded stiffeners and doublers. With this decision, Fokker started the manufacture of metal laminates that would lead to the development of Fibre Metal Laminates, FML's\(^1\).

At that time it was not known that the fracture toughness of a laminated sheet is superior compared to a solid sheet. Also the fatigue properties are better. Schijve et al.\(^2\)\(^3\) found a 60% increase in crack growth life for through cracks in a laminated sheet compared to a solid sheet with the same total aluminium thickness, see Table 1-1. This is the result of the plane stress condition that occurs in thin sheets and results in lower crack growth rates and an increase in fracture toughness. In case of a part through crack, the crack growth life of a laminated sheet was found to be even 10 times higher due to the stress concentration reduction of the adhesive layers in thickness direction and the restraint on crack opening caused by the intact sheets.

Table 1-1, Crack growth life comparison in centre cracked specimens for solid- and laminated 2024-T3 alloy\(^4\).

<table>
<thead>
<tr>
<th>Material</th>
<th>Crack &quot;thickness&quot;</th>
<th>Configuration</th>
<th>Fatigue result</th>
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<tbody>
<tr>
<td>5 mm solid</td>
<td>1 × 5 mm</td>
<td>Through crack</td>
<td>1.6 x longer fatigue life for the laminated material compared to the solid material</td>
</tr>
<tr>
<td>5 × 1 mm laminate</td>
<td>5 × 1 mm</td>
<td></td>
<td></td>
</tr>
<tr>
<td>5 mm solid</td>
<td>1 mm</td>
<td>Part through cracks</td>
<td>10.6 x longer fatigue life for the laminated material compared to the solid material</td>
</tr>
<tr>
<td>5 × 1 mm laminate</td>
<td>1 mm</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
In the late seventies Fokker started a study on reinforcing the bond lines with fibres. Fokker carried out some Constant Amplitude (CA) fatigue tests. These tests gave promising results, but flight simulation tests with Variable Amplitude (VA) carried out at the Delft University of Technology indicated only marginal improvements. In the following years, the Delft University developed the FML’s by optimisation of the aluminium sheet thickness, the type of fibres and the fibre volume fraction ($v_f$). In this way an optimised laminate of thin aluminium layers combined with fibre/epoxy layers was created as a semi-finished product like aluminium sheet material. The optimal metal fraction relative to the fibre/epoxy content appeared to be crucial for a satisfactory fatigue performance. Experiments on this new laminate showed that even with large fatigue cracks present in the aluminium layers, an FML still shows excellent fatigue performance. Under realistic loading conditions, FML’s exhibit crack growth rates 10 to 100 times slower than their monolithic aluminium counterparts. The key to this behaviour is the so-called “fibre crack bridging” mechanism due to the fact that the fibres remain intact under fatigue loading. The stress intensity at the crack tip in the aluminium layers is reduced by effective crack closing stresses in the fibres. The fibres do not fail during crack opening because shear stresses in the adhesive create a controlled delamination under fatigue loading.

The first generation of FML’s was based on aramid fibres and is called “ARALL”. These laminates had unidirectional fibre layers and were developed primarily for wing applications. ARALL became mature when its promising characteristics were finally demonstrated by testing a full scale Fokker 27-wing panel in the period 1984-1987, illustrated in Figure 1.1. This experiment proved that a weight saving of 20% was possible.

Unfortunately, ARALL appeared not to be a good candidate for fuselage structures. Due to the differences in thermal expansion coefficient between the aluminium- and aramid/epoxy prepreg layers and the fact that the curing process takes place at elevated temperature, residual compressive stresses remain in the aramid fibres at room temperature. Under fatigue loading of the laminate with a minimum stress close to zero, compressive stresses in the aramid fibres leads to fibre failure since aramid fibres are sensitive for buckling under compression. Therefore, the material needs to be post-stretched to obtain an initial tensile stress in the aramid/epoxy layer. This complicated and expensive process to post-stretch ARALL with cross-plied fibre/epoxy layers makes ARALL less useful for fuselage skin applications.
In 1987 a second generation of FML’s was born based on high strength glass fibres with better compression properties. The material is called Glare and is applicable in both uni-axially and bi-axially loaded structures. Today, Glare is a product of Structural Laminates Industries (SLI), which is a fully owned subsidiary of Fokker Aerostructures since 1999.

1.2 Glare® lay-up

Glare materials are built up of aluminium sheets with thicknesses between 0.2 and 0.5 mm and intermediate unidirectional glass fibre prepreg layers. This is illustrated in Figure 1.2. The material is cured in an autoclave at elevated pressure and temperature. The outside layers of Glare or FML’s in general are always of aluminium.

Table 1-2 summarises the most common commercially available Glare variants. All variants are based on the fracture tough aluminium alloy 2024-T3 and the high strength S2 glass fibres. The intermediate fibre adhesive layers always consist of two or more Uni-Directional (UD) prepreg layers. The direction of the fibres in the prepreg layers is related to the rolling direction of the aluminium sheets; 0º stands for fibres in the Longitudinal rolling direction (L) while 90º indicates fibres in the Longitudinal-Transverse direction (T).
Figure 1.2, Illustration of a FML with three aluminium layers and two intermediate prepreg layers with fibres in 0°- and 90°- direction.

The lay-up is always symmetrical except for Glare 3. For this variant, the UD prepreg layers closest to the outside aluminium layers are directed in 0° direction. In this way the laminate is symmetric except for the cases with an odd amount of intermediate prepreg layers where the middle prepreg layer causes asymmetry.

Every Glare variant has a variable amount of layers and a variable aluminium layer thickness. The kind of aluminium alloy, the prepreg orientation and the kind of fibres are represented in the variant or type identification. Therefore the following three-fold notation for a standard Glare sheet is currently applied and used in this report:

Glare 4B 5/4 0.4

indicates the thickness of the aluminium layers in mm

indicates the amount of aluminium- (5) and intermediate fibre prepreg layers (4)

identifies the type of Glare, see also Table 1-2.
Table 1-2, Commercially available Glare laminates.

<table>
<thead>
<tr>
<th>Variant</th>
<th>Prepreg lay up between two Al. layers</th>
<th>Thickness prepreg layer</th>
<th>Typical applications</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 2A</td>
<td>0° / 0°</td>
<td>0.25 mm</td>
<td>Unidirectional loaded parts with RD-Al(^1) in loading direction (stiffeners)</td>
</tr>
<tr>
<td>Glare 2B</td>
<td>90° / 90°</td>
<td>0.25 mm</td>
<td>Unidirectional loaded parts with RD-Al(^1) perpendicular to loading direction (butt straps)</td>
</tr>
<tr>
<td>Glare 3</td>
<td>0° / 90°</td>
<td>0.25 mm</td>
<td>Bi-axially loaded parts with ratio 1 :1 of principal stresses (Fuselage skins, bulkheads)</td>
</tr>
<tr>
<td>Glare 4A</td>
<td>0° / 90°/ 0°</td>
<td>0.375 mm</td>
<td>Bi-axially loaded parts with ratio 2 :1 of principal stresses with RD-Al(^1) in main loading direction (Fuselage skins)</td>
</tr>
<tr>
<td>Glare 4B</td>
<td>90° / 0° / 90°</td>
<td>0.375 mm</td>
<td>Bi-axially loaded parts with ratio 2 :1 of principal stresses with RD-Al(^1) perpendicular to main loading direction (Fuselage skins)</td>
</tr>
<tr>
<td>Glare 5</td>
<td>0° / 90°/ 90° / 0°</td>
<td>0.5 mm</td>
<td>Impact critical areas (Floors and cargo liners)</td>
</tr>
</tbody>
</table>

\(^1\) Rolling Direction aluminium

1.3 Application of FML’s in aeroplane structures

The feasibility of Glare as an aeroplane material is determined by its potential to establish significant weight savings and reduce maintenance costs for a competitive price. This section discusses the properties of Glare that are important for application in aeroplane structures.

1.3.1 Static mechanical properties

The static mechanical properties of the Glare laminates are based on the static properties of the constituents; the aluminium- and fibre layers. The
most important properties are listed in Table 1-3. In this table, the subscript 1 identifies properties in direction of the fibres or in rolling direction (L) of the aluminium while 2 represents the direction perpendicular to the fibres or the rolling direction (T) of the aluminium layers. No yield strength is given for the fibre-prepreg since the fibres remain elastic up to failure.

Table 1-3, Properties S2-glass fibre prepreg \(^{9,10,11,12}\).

<table>
<thead>
<tr>
<th>Property</th>
<th>Dim.</th>
<th>UD S2 prepreg ((\nu = 60 %))</th>
<th>2024-T3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Young's modulus, (E_1) ([\text{GPa}])</td>
<td>54.0</td>
<td>72.2</td>
<td></td>
</tr>
<tr>
<td>Young's modulus, (E_2) ([\text{GPa}])</td>
<td>9.4</td>
<td>72.2</td>
<td></td>
</tr>
<tr>
<td>Ultimate strength, (\sigma_{\text{ult}}) ([\text{MPa}])</td>
<td>2640</td>
<td>455</td>
<td></td>
</tr>
<tr>
<td>Ultimate strain, (\varepsilon_{\text{ult}}) ([%])</td>
<td>4.7</td>
<td>19</td>
<td></td>
</tr>
<tr>
<td>Poisson's ratio, (\nu_{12}) ([-])</td>
<td>0.33</td>
<td>0.33</td>
<td></td>
</tr>
<tr>
<td>Poisson's ratio, (\nu_{21}) ([-])</td>
<td>0.0575</td>
<td>0.33</td>
<td></td>
</tr>
<tr>
<td>Shear modulus, (G_{12}) ([\text{GPa}])</td>
<td>5.55</td>
<td>27.6</td>
<td></td>
</tr>
<tr>
<td>Density, (\rho) ([\text{kg/m}^3])</td>
<td>1980</td>
<td>2770</td>
<td></td>
</tr>
<tr>
<td>Thermal expansion coefficient, (\alpha_1) ([1/\degree\text{C}])</td>
<td>6.1·10(^{-6})</td>
<td>23.4·10(^{-6})</td>
<td></td>
</tr>
<tr>
<td>Thermal expansion coefficient, (\alpha_2) ([1/\degree\text{C}])</td>
<td>26.2·10(^{-6})</td>
<td>23.4·10(^{-6})</td>
<td></td>
</tr>
</tbody>
</table>

Figure 1.4 presents the main static material properties of some different Glare variants compared to the current most widely used aluminium alloy 2024-T3. The properties presented are for Glare variants with a 4/3 0.4 lay-up and represent 90% of the average values except for the gross blunt notch behaviour and the \(E_- / G\)-modulus\(^9\). The presented values are indexed, what means that they are presented as a percentage of the 2024-T3 properties (100 represents the 2024-T3 value). The absolute properties for the 2024-T3 material are also written down in this figure.

The test procedures to obtain these values are based on standard test procedures for either metals or composites and are sometimes modified for FML's\(^{13}\). The definition of the bearing strength, \(\sigma_{\text{b-ult}}\), and the specimen geometry is given in Figure 1.3 a). For the determination of the blunt notch strength, \(\sigma_{\text{bn}}\), no standard is available. The blunt notch values presented in Figure 1.4 are based on the definition and specimen geometry given in Figure 1.3 b).
Introduction

\[ A_b = d \cdot t \]

\[ \sigma_{b,\text{ult}} = \frac{P_{\text{max}}}{A_b} \]

\[ \sigma_{bn} = \frac{P_{\text{max}}}{(W \cdot t)} \]

a) bearing strength  

b) gross blunt notch strength

Figure 1.3, Definition of blunt notch- and bearing strength.

Figure 1.4, Indexed static properties of some Glare grades.

All static Glare parameters presented in Figure 1.4 are lower than 2024-T3 except the ultimate strength and the gross blunt notch strength. The yield
strength and the Young's modulus of the laminates are lower since the Young's modulus of the fibre prepreg is lower. The G modulus and the bearing strength are lower since the fibre-prepreg is not effective under these loading conditions. On the contrary, the blunt notch- and ultimate strength are higher because of the higher ultimate strength of the fibre-prepreg compared to 2024-T3. The large difference between the yield stress and the ultimate strength for the Glare laminates illustrates the extensive strain hardening that the materials exhibit.

However, the specific weights or density of the Glare laminates are also lower. Figure 1.5 presents all properties divided by the density, to allow a comparison towards the weight saving potentials. The specific yield strength and Young's moduli are comparable.

![Indexed specific static properties of some Glare grades.](image)

Figure 1.5, Indexed specific static properties of some Glare grades.

In general, the static material properties vary with the Metal Volume Fraction, MVF, defined as the sum of the aluminium layers thicknesses divided by the laminate thickness.

\[
\text{MVF} = 100\% \frac{\sum \frac{t_i}{t_{\text{tot}}}}{\text{eq. 1.1}}
\]
A larger MVF represents a larger area percentage of aluminium in the laminate. 100% reflects a solid aluminium sheet. Research has shown that all basic properties given in Figure 1.4 are linearly related with the MVF\textsuperscript{14,15}. For a larger MVF, the yield stress and the Young’s modulus of the laminate will increase while the ultimate strength decreases.

### 1.3.2 Corrosion

The main source of damage in the service life of current flying aeroplanes is corrosion, followed by fatigue- and impact damage. All aluminium sheets used in the production of FML’s are anodised and coated with a corrosion-inhibiting primer prior to the bonding process. Furthermore, outer aluminium surfaces can be supplied cladded instead of anodised/primed to improve surface corrosion resistance. Therefore, corrosion prevention of the outside aluminium layers in a FML should be comparable with solid aluminium sheet. Actually, the corrosion properties of a thin sheet are found to be better than for a thick sheet\textsuperscript{9}. Additionally, through-the-thickness corrosion in a FML is prevented due to the barrier role played by the fibre-epoxy layers. This limits the extent of corrosion damage in severe environments. Figure 1.6 compares acid bath corrosion damage in ARALL 3 and 2024-T3 after 4 weeks of exposure\textsuperscript{15}. While monolithic metal is fully penetrated, the laminate is merely pitted to the first fibre-epoxy interface.

![ARALL 3 and 2024-T3 comparison](image)

Figure 1.6, Comparison of corrosion damage in ARALL 3 and 2024-T3.

### 1.3.3 Fatigue crack initiation

The fatigue crack initiation period to obtain flaws in the aluminium layers of a Glare laminate is significantly less than for a comparable 2024-T3 sheet. This is due to:

- the lower stiffness of the fibre layers compared to the aluminium layers,
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- the residual stress system after curing that leaves tensile stresses in the aluminium layers and
- the fact that the crack bridging effect of the fibres is only functioning at a crack length ≥ 2 mm.

However, in realistic structures cracks will emanate from rivet holes or damages due to corrosion or impact. In case of a rivet row, the rivets are squeezed in the hole until the rivet head expansion is approximately 1.5 times the rivet diameter. Due to the lower stiffness of the laminate and the comparable squeeze force, the expansion of the hole in the laminate will be larger resulting in a larger area with compressive stresses around the hole. This larger area has a favourable effect on the crack initiation period and balances the worse crack initiation behaviour of the laminate itself.

1.3.4 Fatigue crack growth

In monolithic aluminium, the largest part of the fatigue life is consumed by crack initiation, while for FML’s, the largest part of the fatigue life is the of crack growth life. This is further discussed in chapter 2. Due to the crack bridging effect, FML’s exhibit a superior total fatigue life compared to aluminium. The crack growth rates remain very low and at an approximately constant level which simplifies the determination of inspection intervals.

In case of a lap or butt joint, the fatigue cracks initiate at the mating surface due to fretting and bending stresses. In contrast to aluminium, fatigue cracks in the laminate will not grow through the thickness but remain in the mating aluminium surface for a large part of the life. As a result, the residual strength of a lap joint is hardly affected during its design service life.

1.3.5 Impact

Impact damage in aeroplane structures can be caused by low and high velocity sources like runway debris, hail, maintenance damage (i.e. dropped tools), collision between service cars or cargo and the structure, bird strikes, engine debris, etc. Glare has a higher resistance to cracking due to impact compared to 2024-T3 or carbon composites. This impact performance is attributed to the behaviour of the glass fibres that exhibit an increasing strength when loaded at higher strain rates. From a maintenance point of view, it is favourable that the internal damage in Glare is always
smaller than the plastically deformed dent to be found with general visual inspection.

### 1.3.6 Residual Strength

When discussing the residual strength of Glare it is important to make a distinction between two situations:

1. In case of a fatigue crack, corrosion damage or a minor impact damage for example, one or more of the aluminium layers may be broken, but the fibres are still intact. As a consequence, the residual strength under these circumstances is much higher than for a comparable aluminium counterpart since the fibres are still able to carry load.

2. Through cracks with broken fibres are always caused by accidental damage for example due to runway debris, bird impact or engine non-containment, see Figure 1.7. This thesis will focus on the residual strength for this kind of damage.

![Figure 1.7, Uncontained engine failure MD-88 at Boston Airport, 1998](image)

### 1.3.7 Flame Resistance

The flame resistance of Glare is much better than for monolithic aluminium alloys. In case of an outside kerosene fire, the aluminium skin of current aeroplane fuselages will melt away in 20 to 30 seconds. Consequently,
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passengers will possibly be exposed to these flames within the ninety-second escape time required by the airworthiness authorities. Glare has shown to resist fire conditions for much longer time periods. The glass fibres with its high melting temperature and the insulation due to delamination of the material will protect the second aluminium layer from melting for a significant period and will therefore protect the passengers for a significantly longer time.

1.3.8 Manufacturing

Industrialisation of Glare has recently led to the splicing concept, what allows the production of larger single sheets compared to the sizes of monolithic aluminium sheets. As a result additional weight and production costs savings can be achieved due to the reduction of the amount of joints. The current manufacturing philosophy is to cure the laminate in 2D or 3D moulds simultaneously with local doublers and the stiffening elements. In this way the forming and assembly costs are integrated into the laminate production. This will make the laminate price competitive with the price of comparable aluminium structures.

1.3.9 The A380; the future application?

The combination of properties described in this paragraph makes Glare very suitable for specific parts in high capacity aeroplanes where weight savings, reduction of maintenance costs and safety are of high importance. Market forecasts for civil aeroplanes by both Boeing and Airbus predicted a huge growth of air traffic over the next twenty years: expecting a demand of more than 1400 aeroplanes with 400 or more seats. Because of the economical recession in Asia, this market demand is delayed for a few years but the forecast is still expected to be correct for the long term. As an answer to this predicted growth, Airbus Industry has been studying the economical and technical feasibility to extend their range of aeroplanes with a new family of very high capacity aeroplanes for 480 to 660 seats. In June 2000, the Airbus organisation started to offer the aeroplane to their customers. These were very interested according to the 50 firm orders and 42 additional options that were placed in less than 6 months. In December 2000, the A380 was launched in Paris and is planned to have its first flight at the end of 2004. By launching the A380, Airbus Industrie will extend its product range in the large capacity direction and will thus create an answer to Boeing’s 747 monopoly.
In the current design and production developments of wide-body aeroplanes, reduction of production and operating costs play a key role. However, to maintain the current number of accidents per year, the safety level of the aeroplanes has to increase significantly to counterbalance the growth of air transport. With the A380, Airbus Industrie aims to reduce operating costs by at least 15% below that of the B747-400. A technology break-through is necessary to reach this very ambitious target while at the same time the safety level must be increased. Therefore, Airbus is studying the application of several new materials and production techniques for the A380, like Carbon Fibre Reinforced Plastics, CFRP, in the centre wing box, the outer wings, the upper deck floor beams and the pressure bulkheads, the application of thermoplastic fixed leading edges and the application of weldable alloys for the lower part of the fuselage. The use of Glare is studied for application in the entire top half of the A380 fuselage around the passenger cabin which is critical for fatigue, blunt notch and damage tolerance criteria like the two-bay crack criterion. This is illustrated in Figure 1.8 and discussed in the next section. The use of Glare offers a reduction in operating costs through lower weight and a better maintainability. Additionally, a reduction of production costs is achievable.

Figure 1.8, Illustration of the fuselage area of the A380 for which Glare is a potential candidate.

To obtain technology readiness for the application of Glare in the A380 fuselage, a Glare Technology Development Plan has been defined between the Airbus partners and the Dutch industry. This plan is carried out at the
Delft University of Technology, the National Aerospace Laboratory and Fokker Aerostructures and focuses on items like the qualification of the material, the development of design methods, manufacturing and maintenance support. Part of the process was the introduction of the material to the airlines, which took place in autumn of 1998. The airlines expressed their confidence in the material and did not see any show stoppers.

1.4 Objectives of study

The objective of this study is to develop a better knowledge of the fracture behaviour of statically loaded Glare material with through cracks and holes and the development of design tools for residual strength and blunt notch predictions.

The aim of this thesis is to obtain a better understanding of mechanisms and to develop valid design tools for the fracture toughness and blunt notch behavior of Glare material with and without the presence of structural details like splices, doublers and crack stoppers.

The contribution of the Civil Engineering department is described by Hashagen and comprises the derivation of several finite element types and the development of a plasticity-, crack- and delamination model. It uses a large amount of the experimental results that are described in the present report for verification and optimisation.

There is also a clear link with the work of Vermeeren. However, the work of Vermeeren was a pilot study that investigated the applicability of existing fracture mechanics tools for Glare and compared the material with other aeroplane materials. The present study builds on that work but is broader and more geared to the design of real structures. A much larger variety of different Glare variants have been tested and attention is given to details like splices, and crack stoppers but also to high-speed testing and the influence of low temperatures.

1.5 Outline

This introductory chapter has given an overview of the development of Glare, the composition of the material and its properties. Special attention was given to the development to use Glare as a skin material for the A380 fuselage. For this application, it is expected that the residual strength
behaviour of the material in case of a two-bay crack and the blunt notch behaviour are critical design criteria for large parts of the fuselage. The aim of this thesis is to obtain a better understanding of the material behaviour in the presence of notches, to develop a valid design tool and to optimise the material composition or structure for the presence of notches. The report can be divided in three parts; a theoretical part, an experimental part and a development and application part.

1.5.1 Part 1: damage tolerance philosophy and literature survey

Before discussing the development of design tools and the necessary validation with experiments, it is important to discuss the damage tolerance philosophy of Glare and to summarise the available tools and experimental results. The damage tolerance requirements and their applicability to Glare is discussed in chapter 2. Chapter 3 will give a summary of a literature survey that was carried out to collect the experimental results on Glare and the available design tools for both blunt and sharp notches. The available methods will be applied on the collected results and evaluated. This chapter also focuses on the fracture mechanism that has been observed in sharp notch tests for crack initiation and crack extension respectively and pays attention to the occurrence of delamination.

1.5.2 Part 2: Experimental program

To support the development of a Glare Finite Element by Hashagen and to observe the fracture mechanisms in more detail, a digital imaging system was developed. This system consists of a CCD camera mounted on a movable platform, both computer-controlled. It is able to distinguish markers on a specimen surface and to determine the relative displacement of these markers. The development and functioning of this system is described in chapter 4. Chapter 5 and 6 discusses the experimental program that was carried out on blunt and sharp notch specimens respectively. This includes the development of both a blunt notch and residual strength design tool.

1.5.3 Part 3: Applications

Part 3 discusses the application of the design tools to structural details and the behaviour of the material in real structures. Chapter 7 discusses the influence of a splice on the blunt and sharp notch behaviour. In principle, the stress people, designers and aeroplane-manufacturing specialists should have to pay minimum attention to the presence of splices in their
work. This means for stress engineers that the mechanical behaviour of a spliced Glare sheet must be comparable to that of an undisturbed sheet, independent of the location of holes or cracks. Chapters 8 and 9 describe the application of Glare in a real structure. The following questions will be addressed:

- How does the material behave in case of a two-bay crack?
- Must Multiple Site Damage (MSD) be taken into account? and
- Is there a difference between the testing of specimens with a saw-cut in the laboratory and panel with an actual damage due to a high velocity projectile in a fuselage with internal pressure?
References


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[18] [http://www.aviationpics.de](http://www.aviationpics.de)


Part 1:
Damage tolerance philosophy and literature survey
2 DAMAGE TOLERANCE CONCEPT FOR GLARE

2.1 Damage tolerance concept

To ensure a satisfying aeroplane service performance, five failure modes have to be considered during the design process. These are given in Table 2-1 together with the corresponding design criteria or requirements set by the airworthiness authorities. The third column presents the material properties that play a role in the failure mode and determine the design allowable for a specific material.

Table 2-1, Design criteria for sizing aeroplane structures

<table>
<thead>
<tr>
<th>Mode of failure</th>
<th>Design criteria</th>
<th>Design allowable</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. Static strength of undamaged structure</td>
<td>Structure must support Ultimate Loads (UL) without failure for 3 seconds</td>
<td>Static properties</td>
</tr>
<tr>
<td>2. Deformation of undamaged structure</td>
<td>• No permanent deformation below Limit Loads (LL) and</td>
<td>Static properties and</td>
</tr>
<tr>
<td></td>
<td>• deformation of undamaged structure at limit loads may not</td>
<td>creep properties for</td>
</tr>
<tr>
<td></td>
<td>interference with safe operation</td>
<td>elevated temperature</td>
</tr>
<tr>
<td></td>
<td></td>
<td>conditions</td>
</tr>
<tr>
<td>3. Fatigue crack initiation of undamaged</td>
<td>• Damage tolerant structure must meet service life requirements</td>
<td>Fatigue properties</td>
</tr>
<tr>
<td>structure</td>
<td>• Safe life components must remain crack free in service</td>
<td></td>
</tr>
<tr>
<td>4. Crack growth life of damaged structure</td>
<td>For damage tolerant structure inspection techniques and frequency</td>
<td>Crack growth properties</td>
</tr>
<tr>
<td></td>
<td>must be specified to minimise risk of</td>
<td>and Fracture toughness</td>
</tr>
<tr>
<td></td>
<td>catastrophic failures</td>
<td>properties</td>
</tr>
<tr>
<td>5. Residual static strength of damaged</td>
<td>Damaged structure must support</td>
<td>Static properties and</td>
</tr>
<tr>
<td>structure</td>
<td>limit loads without catastrophic failure</td>
<td>Fracture toughness</td>
</tr>
<tr>
<td></td>
<td></td>
<td>properties</td>
</tr>
</tbody>
</table>

Especially designing for the last three failure modes; crack initiation, crack growth and residual strength, determines the airworthiness and the economical aspect of the aeroplane design. They deal with the fatigue and damage tolerance properties of materials and structures and will be discussed in this section with regard to Glare. The residual strength capability is defined as the amount of static strength available at any time.
during the service exposure period considering that damage is initially present and grows as a function of service exposure time. The strength of a structure can be significantly affected by the presence of a damage or crack and is usually substantially lower than the strength of the undamaged structure.

The damage tolerance design philosophy presumes that any damage initiated by material processing, manufacturing, fatigue, corrosion or accidental damage is either found before catastrophic failure occurs or never reaches a dangerous size during the design lifetime of the aeroplane. The safety of the aeroplane heavily depends upon finding cracks before they reach a critical length.

The airworthiness requirements for civil aeroplanes are specified in the European Joint Airworthiness Requirements (JAR) and the American Federal Aviation Regulations (FAR). The requirements of both organisations are very similar. The requirements with respect to fatigue and damage tolerance have been formulated in § 571 of JAR/FAR part 25.

An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, manufacturing defects, or accidental damage, will be avoided throughout the operational life of the airplane(...). This evaluation must be conducted for each part of the structure that could contribute to a catastrophic failure(...). Each evaluation must include the typical loading spectra, temperatures, and humidities expected in service and the identification of principal structural elements, the failure of which would cause catastrophic failure of the airplane.

The evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage (...). The residual strength evaluation must show that the remaining structure is able to withstand loads (considered as static ultimate loads) corresponding to the following conditions:

- Limit manoeuvring conditions at speeds up to Vc
- Limit gust conditions at speeds up to Vc
- For pressurised cabins:
  - the normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditions and
  - the maximum value of normal operating differential pressure multiplied by a factor 1.15 omitting other loads.

The airplane must be capable of successfully completing a flight during which likely structural damage occurs as a result of

- impact with a 4 pound bird
- uncontained fan blade impact;
- uncontained engine failure; or
- uncontained high energy rotating machinery failure
The damaged structure must be able to withstand the static loads (considered as ultimate loads), which are reasonably expected to occur on the flight. Dynamic effects on these static loads need not be considered.

It requires that all the Principal Structural Elements (PSE’s) in a structure have to be identified and must be designed according to the damage tolerance rules, see Figure 2-1. PSE’s are elements that contribute significantly to carrying ground-, flight- and pressurisation loads and whose failure could result in catastrophic failure of the aeroplane. The objective of the damage tolerance evaluation is to provide an inspection program for each PSE, such that cracking initiated by fatigue loading, accidental damage or corrosion will not propagate to catastrophic failure prior to detection.

Figure 2-1, Damage tolerant approach JAR/FAR 25.
Only when inspection of a PSE is complicated for example because the damage is difficult to detect, the accessibility is limited or the inspection interval would be unpractical, the PSE design should be in accordance with the safe life requirement. This means that the structure should be free from any detectable damage during its Design Service Goal (DSG).

Within the damage tolerance approach, distinction is made between single- and multiple load-path damage tolerant structures. In a single load-path structure, failure could lead to loss of the aeroplane. Therefore, damage should be easily detectable and contain demonstrated slow crack growth or crack-stopping capability. For the determination of the inspection schedule, an initial crack is assumed to be present at the introduction of the aeroplane into service. Therefore, the inspection interval and inspection threshold are equal and based on the predicted Crack Growth Life (CGL) divided by a safety factor (SF). The CGL is the amount of flights necessary to let a detectable crack length, $a_{\text{det}}$, grow to a critical crack length, $a_{\text{cr}}$. The SF is based on several different aspects like the accessibility, lighting, surface, inspection methods, etc.

A multiple load-path structure consists of different load-paths, which carry the load to some extent. Failure of one load-path results in a load redistribution to the other load-paths. For determination of the inspection schedule the fail-safe approach is used, which assumes that one of the primary components failed. The inspection threshold is now determined by the fatigue analyses of the remaining structure and the interval is based on damage tolerance analyses.

The damage tolerance analyses must be based on:

1. a deterministic approach to determine
   - $a_{\text{det}}$,
   - CGL, and
   - the residual strength

2. a probabilistic approach to determine the fatigue life of the remaining elements after load redistribution due to the total failure of the most critical element.

Both the threshold and the interval are based on analytical assessments and experimental results. The deterministic analyses of crack growth and residual strength are generally based on linear elastic fracture mechanics, see Figure 2-2. To verify the analytical results, fatigue and damage
tolerance tests are carried out. These tests also function to identify critical locations and to check the inspection methods used. These tests vary from small coupon tests to component tests and full-scale tests.

Figure 2-2, Different phases of the fatigue life\(^5\).

The units for inspection intervals are either flights for impact and fatigue damage or calendar time in case of inspections for corrosion or other environmental effects. The derived inspection schedule for every PSE should be fit into a general inspection program for a certain aeroplane, described by the aeroplane manufacturer. Table 2-2 gives an example of such an inspection program. Thus, if an accidental damage occurs with a size close to \( a_c \) that should become catastrophic during the next flight, this damage must be detected during the daily check of the flight crew before the catastrophic flight.

Table 2-2 Example of inspection program\(^5\).

<table>
<thead>
<tr>
<th>Check</th>
<th>Interval</th>
<th>Definitions</th>
</tr>
</thead>
<tbody>
<tr>
<td>daily</td>
<td>36 hours</td>
<td>Walk around to detect obvious discrepancies (systems checks to be covered by flight crew)</td>
</tr>
<tr>
<td>A</td>
<td>400 flying hours</td>
<td>General visual inspection from ground with the access to landing gear / hydraulic bays and simple operational checks from flight deck</td>
</tr>
<tr>
<td>C</td>
<td>12-15 months or 3000 flying hours</td>
<td>General visual inspection, external and internal through quick access</td>
</tr>
</tbody>
</table>

As a summary, Figure 2-3 shows a schematised relation of the residual strength capability of- and the damage size within a structural element, both as a function of the lifetime. The service life of an aeroplane can be divided in two stages:
• **The safe-life interval.** During this interval, complete failure can only occur when the applied load exceeds the design ultimate strength. The length of this interval is determined by the static- and fatigue properties of the structure.

• **Fail-safe or damage tolerance interval.** During this interval, the initiated crack or damage starts to grow. As a consequence the failure load decreases. Complete failure can occur for loads below the ultimate design load but above the limit design load.

*Figure 2-3, Illustration of damage accumulation.*
The length of the fail-safe interval defines the inspection interval and is a function of the residual strength reduction rate and the design criteria set by the certifying agencies. Whenever cracks are found in a structure, they need to be repaired immediately, even if the structure is still capable to maintain design ultimate load.

As described above, safety can be achieved by designing the aeroplane elements either based on slow crack growth or fail safe by applying multiple load paths. Another possibility to create fail safety is to add crack arresting capabilities to the structure.

This thesis focuses mainly on the remaining residual strength of a damaged Glare material or structure. These damages can be either caused by fatigue cracks, corrosion or impact damage and have a different impact on the residual strength of Glare laminates. This was explained in chapter 1. Although the damage might have several shapes, they are conservatively idealised as cracks in this thesis.

### 2.2 Ageing aeroplanes

Aeroplanes are designed and certified for a certain DSG. Most of the long-range aeroplanes like the Boeing 707, Boeing 747 and MD11 were designed for a DSG of 20,000 flight cycles while the American short range aeroplanes were designed for DSG’s up to 75,000 flights. Nowadays there is a large demand from the airliners towards the aeroplane manufacturers to keep their aeroplanes longer into service than the certified DSG’s. However, structural ageing of an aeroplane may significantly reduce its strength below an acceptable level and raises many important safety issues. This means that the airliners must certify their aeroplanes for an Extended Service Goal (ESG) and come up with necessary modifications and/or a modified inspection program. Aeroplanes that are flying beyond their DSG are called ageing aeroplanes.

One of the most notable problems in ageing aeroplanes is the occurrence of Widespread Fatigue Damage (WFD) defined as the simultaneous presence of fatigue cracks at multiple structural details that are of sufficient size and density whereby the structure will no longer meet the damage tolerance requirements.

Multiple Site Damage (MSD) is one of the sources of WFD characterised by the simultaneous presence of fatigue cracks in the same structural element. MSD can occur when the structural details are geometrically similar and
experience comparable stress levels. This results in a similar crack driving force at each of the probable crack nucleation sites, typically occurring at locations like a rivet row or lap joint. The cracks appear at about the same time, grow simultaneously and the individual cracks can be close enough to each other to influence the crack growth of adjacent cracks. They can suddenly coalesce to a much larger crack, which may lead to catastrophic failure of the structure, see Figure 2-4. The critical size of the individual cracks can be very small, even less than the length easily detected during visual in-service inspections. Due to the character of MSD, it occurs mainly in fuselage structures and asks for special, dedicated inspections.

![Figure 2-4, Residual strength capability in case of MSD](image)

MSD was generally recognised after some major accidents. The most referred accident regarding MSD is probably the Aloha Boeing 737 illustrated in Figure 2-5). In this accident, significant small individual unrepaired cracks in an aluminium fuselage lap joint linked up to form a large crack, which resulted in the loss of a portion of the fuselage.

In the past, aeroplanes have been designed not to crack within one lifetime and to tolerate certain damages, which are the results of either initial manufacturing flaws or a certain degree of discrete source damage. The inspection programs according to the damage tolerance philosophy are based on finding these cracks before they become critical. In general, deriving a realistic stress spectrum for each PSE and determining the resulting CGL starting from a detectable crack length up to the critical crack length tries to achieve this. This is illustrated in Figure 2-6. Based on the
Damage tolerance concept for Glare

type of structure, this period is divided by an SF 2 or 3 and determines the inspection interval and -threshold.

Figure 2-5, Pictures of Aloha accident 1988.

![Diagram showing crack growth and inspection intervals](image)

**Figure 2-6.** Reduced lead crack inspection frequency resulting from reduced critical crack size due to MSD$^6$.

However, since the trend is to extent the operational lifetime beyond the aeroplane DSG, the occurrence of MSD (and therefore the possibility of WFD) increases with the amount of extra operations and reduces the residual strength capability. As a consequence, the safe crack growth life reduces and the time between required inspections becomes shorter, see Figure 2-6.
According to the airworthiness authorities, managing structural safety in the presence of WFD is not reliable with current in-service inspection sensitivity. Swift stated:

"Damage tolerance was not intended as a safety management tool for structures operating beyond their initial design life goals or beyond the point where WFD is likely to occur".

Therefore, the airworthiness authorities have published several Advisory Circulars (AC’s) that consider new aspects regarding WFD. They advise sufficient full scale testing to ensure that WFD will not occur within the DSG of an aeroplane.

The preliminary DSG’s for the A380 are given in Table 2-3. Reserves have to be considered for locations potentially susceptible to WFD. If the point of WFD takes place within the DSG of the aeroplane, this type of structure has to be retired, modified or repaired before reaching the ESG. This will lead to unacceptable costs.

Table 2-3, Design Service Goals for the A380.

<table>
<thead>
<tr>
<th>DSG</th>
<th>Years</th>
<th>Flight hours</th>
<th>Flight cycles</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inspection threshold</td>
<td>10</td>
<td>50.000</td>
<td>8.000</td>
</tr>
<tr>
<td>Inspection interval</td>
<td>5</td>
<td>25.000</td>
<td>4.000</td>
</tr>
</tbody>
</table>

2.3 Two-bay crack criterion

Testing of pressurised fuselage shell structures at McDonald Douglas has indicated that fatigue cracks are most likely to occur at the following locations and directions:

1. In the case of longitudinal skin cracks:
   a. Along the line of attachments which attach the outer fingers of a longitudinal splice member to the skin and
   b. At the first attachment of a frame shear clip to a skin joint

2. In the case of circumferential skin cracks:
   a. At the attachment of the skin to frame shear clip midway between longitudinal stringers and
   b. In the longitudinal stringer flanges where they attach to the frame. After failure of the stringer, the skin stresses increases
Damage tolerance concept for Glare

locally causing fatigue cracks in the skin which propagate into the two adjacent skin bays

As a result of these tests, the damage size and configuration to ensure damage tolerance for the DC-10 shell was therefore defined as:

1. a two-bay circumferential crack with a broken centre stringer and
2. a two-bay longitudinal crack with the centre frame intact.

Both damage configurations are illustrated in Figure 2-7.

Although the airworthiness authorities do not require a two-bay crack criterion at this moment, Airbus Industrie has set this as a design criterion for the A380 fuselage\textsuperscript{5,11} to reach the standard of their competitors. The two bay crack definition that is used to design for is more severe than the
condition that followed from the fatigue analyses described above. It assumes that a longitudinal crack in the skin of the pressurised fuselage with a length of two frame bays above a broken centre frame does not lead to complete failure of the structure under the conditions mentioned in JAR/FAR 25.571.

This damage can be caused by a Discrete Source Damage (DSD) that penetrates the skin and the central frame or eventually by fatigue of a frame followed by subsequent crack growth in the skin. Fatigue crack growth in the skin and subsequent failure of the frames is not likely to occur since the frames are usually flexible connected to the skin by riveted clips. It is more likely that a fatigue crack in the skin continuously grows in the skin until the crack stopper or the rivets of the skin-clip or clip-frame connection fail.

Of the sources that create DSD, only uncontained turbine engine and Auxiliary Power Unit (APU) rotor failure are expected be able to results in a two bay longitudinal crack. AC 20.128A\textsuperscript{12} reports about the occurrence of uncontained gas turbine engine rotor failure in the period between 1962 and 1989. In these 28 years, 676 uncontained events were reported. 93 of these events were defined as significant aeroplane damage with the aeroplane capable of continuing flight and making a safe landing. 15 events created severe aeroplane damage involving a crash landing, critical injuries, fatalities or hull loss. During this period 1,089.9 million engine operating hours were made with commercial transports. In the same period, several APU failures occurred. None of these was significant and all occurred during ground operation.

The FAA recommends to design against engine failure considering the following two models where:

1. a single one-third disc fragment that leaves the turbine under a fragment spread angle of $\pm 5$ degrees, or
2. small fragments ranging in size corresponding to the tip half of the blade airfoil that leave the turbine under a fragment spread angle of $\pm 15$ degrees,

Based on these conditions, a safety analysis must be made to determine the critical areas of the aeroplane that are likely to be damaged by rotor debris, to evaluate the consequences of an uncontained failure and to minimise the hazards.
For the A380, a two-bay longitudinal crack means a crack with a length of approximately 1.3 m! It is clear that this requirement can have an impact on the allowable stresses in the upper and side shells of the pressurised fuselage. To limit the implications of this two-bay crack criterion on the weight the following design improvements can be applied:

- selection of skin material with a high residual strength
- selection of frame material with a high static strength and sufficient fracture toughness
- improvement of the skin - frame connection if necessary
- limitation of the allowable frame distance (pitch)

Compared to the currently used aluminium alloy 2024-T3, the new alloy 2524-T3 shows significantly higher fracture toughness and was therefore chosen as the fuselage material for the Boeing 777. Another solution could be the application of Glare variants with extra fibres in the critical directions. For this reason, Glare 4 may be a good alternative for areas where the damage tolerance properties are critical. This is illustrated in Table 2-4. A disadvantage of both 2524-T3 and Glare is that they are more expensive. This thesis focuses on the residual strength of Glare as a skin material.

Table 2-4, Relative allowable stresses for fuselage skin to meet residual strength and crack growth criteria.

<table>
<thead>
<tr>
<th>Skin material</th>
<th>Allowable stress in circumferential direction (residual strength)</th>
<th>Allowable stress in longitudinal direction (crack growth / residual strength)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Alclad 2024-T3</td>
<td>100%</td>
<td>100% / 100%</td>
</tr>
<tr>
<td>Alclad 2524-T3</td>
<td>120%</td>
<td>113% / 110%</td>
</tr>
<tr>
<td>Glare 4 clad</td>
<td>120%</td>
<td>&gt;120% / 100%</td>
</tr>
</tbody>
</table>

### 2.4 Damage tolerance philosophy for Glare

To discuss the damage tolerance behaviour of Glare, a distinction must be made between two different kinds of damage of the laminate:

1. **Cracks in one or more aluminium layers while the prepreg layers are still intact**
   These cracks are mainly caused by fatigue loading and originate for example at a rivet hole. In this case one or more of the aluminium layers may be broken but the fibres remain intact. They bridge the
crack and reduce the stress intensity at the crack tip. Other causes that create "part-through cracks" in Glare are scratches, corrosion damage or lightning strikes.

2. **Cracks in the prepreg- and aluminium layers**

   These cracks occur due to severe DSD when an object penetrates the structure. This damage creates cracks with broken fibre- and aluminium layers. Examples of DSD are the earlier described engine non-containment but also a bird-strike and runway debris.

   This thesis focuses mainly on the second kind of damage. Only some paragraphs discuss fatigue cracks. Both types of damage and the related damage tolerance philosophy are described in this section.

### 2.4.1 Fatigue cracks

The fatigue behaviour of the fibre prepreg and therefore of Glare is different from the fatigue behaviour of aluminium alloys like 2024-T3. The fibre layers are to a large extent insensitive for fatigue loading and fatigue crack growth in Glare is therefore, limited to the aluminium layers.

Due to the lower stiffness of the fibres compared to the aluminium layers within the laminate, the crack initiation period in Glare is smaller compared to monolithic aluminium because the aluminium layers in the area of a stress concentration attract more load. However, as soon as the fibres start to bridge the cracks in the aluminium layers, the crack growth rate is significantly reduced and the failure strength remains high for a longer service life compared to aluminium. A Glare sheet is actually a multi load path structure in itself where the fibre layers become effective when the aluminium layers partly failed. This is illustrated in Figure 2-8a.

In case of a riveted joint, the crack initiation period is significantly increased due to the applied squeeze forces. When cracks originate, they will start normally at the upper rivet row, see Figure 2-8b. Due to secondary bending, the cracks will start in the mating aluminium layer. Since cracks need to be reinitiated for every subsequent aluminium layer, also the aluminium layers are part of the multi load part structure.

Because of the slow crack growth behaviour, the allowable design stresses for Glare will be higher than for aluminium causing again shorter crack initiation periods. The occurrence of MSD will be advanced, however, the
occurrence of WFD is most likely delayed or will never occur within the DSG.

Figure 2-8, Fatigue crack growth in Glare applications.

Figure 2-9 illustrates the difference in damage tolerance behaviour between aluminium and Glare. The amount of flight cycles to initiate cracks, $N_i$, or safe life period is smaller for Glare. On the contrary, the period between initiation and the moment that the failure load drops below the design ultimate load is significantly larger. This means that a Glare structure, although it contains small cracks, is longer able to withstand the design ultimate load.

Based on the current damage tolerance requirements, cracks in a Glare structure need to be repaired when they are found. If flying with small cracks is not allowed, this would either imply that the allowable design stress must be decreased or that inspections must be started early in the design life of the aeroplane. This has large consequences for the economical feasibility of the material.
Figure 2.9. Difference in damage tolerance behaviour between aluminium and Glare.13
Therefore, discussions are taking place with the European airworthiness authorities to allow flying with Glare with small anticipated fatigue cracks as long as the structure can sustain design ultimate load. The fuselage will not need other inspections than normal routine maintenance (A-/ B-/ C-Check). This philosophy provides additional safety compared with the behaviour of the monolithic aluminium structure, which must be able to withstand only limit load for unknown damages up to the next inspection.

The fatigue – and damage tolerance specialists have to derive design allowables (maximum permitted stresses) based on validated calculation methods or experimental results below which the crack growth rate is sufficiently low. The goal for the designers is to design the fuselage in such a way, that the allowable stress level will remain above the design ultimate load within the design life. This means that after the service life the structure is still able to withstand ultimate load.

Table 2-5 gives a comparison between the application of the damage tolerance philosophy in case of aluminium and Glare.11,14.

The discussion above illustrates the changed use of the damage tolerance philosophy. Not only the material properties demand for such an adaptation. The acceptance to fly with Glare structures with small, anticipated cracks without a repair required is essential for the economic attractivity of the material. If flying with small fatigue cracks would not be allowed, either the allowable stresses would need to be decreased even below the level of aluminium allowables, or the Glare structure would require earlier inspections than an aluminium structure. In both cases Glare would be less attractive.

The application of Glare in the fuselage will change the damage scenario sketched by Swift, see § 2.3, in some aspects. In circumferential direction, the stringers will be bonded to the skin. Only at those places where fuselage sections are coupled during final assembly and at stiffener repairs, the stringers will be riveted to the skin. Therefore, a two-bay circumferential crack criterion with a broken central stiffener is only representative for these locations. The crack propagation rate will be much lower since the fibres will remain intact. If it can be proven that ultimate load can be carried during the design service life, the aeroplane needs to be inspected and maintained for circumferential fatigue cracks only at fuselage section couplings.

In longitudinal direction, the damage scenario in Glare will be comparable with a design of monolithic aluminium. The clips that connect the frames to
the skin will be connected with rivets and will cause crack initiation in the skin. The frames will remain intact and due to intact fibres and the slow crack propagation, it is expected that the aeroplane will be able to carry limit load during the DSG. The critical situation of a two bay crack configuration due to fatigue loading described by Swift is never expected to occur in a Glare skin within the DSG due to fatigue loading. However, this must be verified.

Table 2-5, Damage tolerance differences between aluminium and Glare

<table>
<thead>
<tr>
<th>Aluminium</th>
<th>Glare</th>
</tr>
</thead>
<tbody>
<tr>
<td>• MSD leading to WFD must be avoided</td>
<td>• MSD for small cracks can be allowed since the residual strength degradation is very limited. (Ultimate load requirement has to be met)</td>
</tr>
<tr>
<td></td>
<td>• Large single cracks (are not expected to occur) will be treated similar to single fatigue cracks in monolithic aluminium.</td>
</tr>
<tr>
<td></td>
<td>• Inspection intervals to be determined by means of crack propagation analysis or residual strength fatigue life.</td>
</tr>
<tr>
<td></td>
<td>• Inspection intervals for Glare exceed Design Service Goal due to slow crack propagation (exceptions may be possible).</td>
</tr>
<tr>
<td></td>
<td>• A structural component must be able to carry limit load in the cracked condition until detected.</td>
</tr>
<tr>
<td></td>
<td>• A structural component must be able to carry ultimate load in the cracked condition if no directed inspections are prescribed.</td>
</tr>
<tr>
<td></td>
<td>• A detected crack has to be repaired immediately.</td>
</tr>
<tr>
<td></td>
<td>• Flying with small fatigue cracks is allowed, since they do not lead to failure below ultimate load condition.</td>
</tr>
<tr>
<td></td>
<td>• Directed fatigue inspections are required to meet damage tolerance criteria.</td>
</tr>
<tr>
<td></td>
<td>• Directed fatigue inspections are not required to meet damage tolerance criteria (exceptions may be possible).</td>
</tr>
</tbody>
</table>

2.4.2 Accidental damage

In case of accidental damage in Glare, both the fibre- and the aluminium layers will be broken. The kind of accidental impacts that have to be considered for the Glare upper part of the A380 fuselage are hail and engine debris. Bird strikes occurrences are most likely to be limited to the cockpit area, see Figure 2-10, and will occur at a lower altitude where the
Damage tolerance concept for Glare

internal pressure in the fuselage is limited. Runway debris will take place at the lower part of the fuselage. Impact due to hail has been investigated by Vlot\textsuperscript{15} and did not cause serious damage in Glare. Therefore, only engine debris must be taken into account and the two bay-crack with broken centre frame or stringer must be verified at the most likely locations of impact. This is in areas of the fuselage that are within the ±15° fragment-spreading angle of the engines. Regarding these kind of damages, AC 25.571-1C\textsuperscript{5} advises about the load conditions that are reasonable after this impact. This AC expects the pilot to be aware of the impact to anticipate in order to avoid severe load environments for the remainder of the flight consistent with his knowledge that the aeroplane may be in a damaged state.

Figure 2-10, Damage due to a bird strike\textsuperscript{16}. 
Chapter 2

References


3 AVAILABLE BLUNT AND SHARP NOTCH GLARE DATA

3.1 Introduction

Notches in a structure are unavoidable. The disadvantage of notches is that they cause local stress redistributions in the notched material and create stress concentrations. The intensity of these concentrations is described with the stress concentration factor \( K_t \), which is defined as the ratio between the maximum stress \( \sigma_{peak} \) at the notch root and the average stress \( \sigma_{nom} \) in the net section, see Figure 3-1:

\[
K_t = \frac{\sigma_{peak}}{\sigma_{nom}} = \left[ \frac{\sigma_{peak}}{\sigma_{gross}} \right]
\quad \text{eq. 3-1}
\]

For an isotropic, elastic material with an elliptical hole in an infinite sheet that is loaded in tension, \( K_t \) is equal to:

\[
K_t = 1 + 2 \frac{a}{b} = 1 + 2 \sqrt{\frac{a}{r}}
\quad \text{eq. 3-2}
\]

Figure 3-1, Definition of blunt notch parameters.

In the limit situation that the height \( b \) of the ellipse and the notch tip radius \( r \) decrease to zero, the notch turns into an infinitely sharp notch. In this case eq. 3.2 predicts an infinite stress concentration. In reality this will not occur since the material becomes plastic at the crack tip, which limits the validity to apply the theory of elasticity.
In case of sharp cracks, the severity or intensity of the stress field around the crack tip becomes important. It is derived that the stress field in front of a crack can be described with the stress intensity factor $K_i$ as long as the amount of plasticity around the crack tip is limited. This factor is based again on the theory of elasticity:

$$K_i = \beta \cdot \sigma_{\text{gross}} \sqrt{\pi a} \quad \text{eq. 3-3}$$

where $I$ represents the opening mode, $a$ is equal to half of the crack length for a centre crack and $\beta$ is a correction factor for the finite width, $W$.

Roebroeks$^1$ and Vermeeren$^{2,3}$ investigated the static strength of specimens of different FML’s that contained central notches varying from round holes to sharp cracks. Figure 3-2 illustrates the experimental results for the materials Glare 2 3/2 0.3 and 2024-T3 as a function of the stress concentration factor $K_t$. The nominal strengths are related to the materials uni-axial ultimate tensile strength $\sigma_{\text{ult}}$. The $K_t$ factors for the finite width 2024-T3 specimens originate from Peterson$^4$.

For non-isotropic (orthotropic) infinite Glare sheets the $K_t$ values are calculated with$^{1,5}$

$$K_{t,o} = \frac{\sigma_{\text{peak}}}{\sigma_{\text{nom}}} = 1 + \frac{a}{b} \left( \frac{\mu_1 + \mu_2}{\mu_1 \mu_2} \right)$$

$$\mu_1 = \sqrt{\frac{g-h}{2}} + i \sqrt{\frac{g+h}{2}}$$

with

$$\mu_2 = \sqrt{\frac{g-h}{2}} + i \sqrt{\frac{g+h}{2}} \quad \text{eq. 3-4}$$

$$g = \frac{E_{22}}{E_{11}}$$

$$h = \frac{E_{11}}{2G_{12}} - v_{12}$$

where the subscript 22 reflects the loading direction and 11 is perpendicular to the loading direction (y and x respectively in Figure 3-1). To obtain the $K_t$ values for orthotropic, finite width specimens, these values are divided by the isotropic, infinite sheet solutions (eq. 3.2) and multiplied with the results from Peterson for isotropic materials and finite width:

$$K_{t,\text{Glare}} = \frac{K_{t,\text{inf}}}{K_{t,\text{inf},\text{Peterson}}} \cdot K_{t,\text{fin}} = \left( \frac{\text{eq. 3.4}}{\text{eq. 3.2}} \right)^{\frac{\text{handbook result of Peterson}}{}} \quad \text{eq. 3-5}$$

In this equation represents the subscripts o = orthotropic material, i = isotropic material, inf = infinite width and fin = finite width.
Available blunt and sharp notch Glare data

Figure 3-2, Relative nominal strength values for different notch geometries.

The results for 2024-T3 show a consistent decrease of the nominal strength as a function of Kt. The figure also contains a line that represents the strength reduction for a notch geometry that represents a crack (b = 0.5). The measured reduction is in the order of 30% for the given configuration.

The Glare results illustrate that the presence of a notch causes large strength reductions while the shape of the notch is of less importance. Above a Kt value of 3, the influence of Kt on the notch strength seems to be disappeared. Notches that are identified blunt for aluminium alloys (Kt<4) seem to behave like sharp notches in case of FML’s. This is another reason why crack initiation for FML’s is quicker. However, Figure 3-2 does not say anything about crack growth. In contradiction to aluminium, the blunt and sharp notch behaviour of FML’s are strongly related.

In the design of the aircraft, the blunt notch strength is an ultimate design load for static load cases while the yield strength $\sigma_y$ of the material is a limit design load. Because the ultimate load is defined to be equal to 1.5 times the limit load, this means that the blunt notch strength becomes critical when the ratio $\sigma_{bn} / \sigma_y$ decreases below 1.5. With the results presented in Figure 3-2, it is clear that this might be the case for Glare laminates.

This chapter gives an overview of the available Glare notch data in the literature. § 3.3 and 3.4 deal with blunt notches. § 3.3 discusses the blunt notch results obtained with specimens with round holes. Items like the
influence of the width, the relative notch diameter, multiple holes and biaxial loading are discussed. § 3.4 discusses the available fracture models for blunt notches.

§ 3.5 to § 3.8 deals with the strength of sharp notches or cracks. This second part was published before. § 3.5 gives an overview about the kinds of damage that can occur, which effect they will have for a Glare material and how the material behaves in case it contains a through crack. In § 3.6 experimental residual strength results are discussed to give a general overview but also to discuss the influence of, for example, temperature and material orientation. § 3.7 discusses the available residual strength prediction tools that have been investigated for the use for Glare. § 3.8 presents the Glare residual strength results in comparison with the aluminium alloys 2024-T3 and 2524-T3. Finally, § 3.9 summarises the most important results discussed in this chapter.

This chapter is not only a summary of the available literature; after collecting the available data, the data are compared and discussed.

3.2 Elastic strain distributions around blunt notches

Around a uni-axially loaded hole in an infinite sheet of isotropic linear-elastic material, a stress concentration is created where $\sigma_{\text{peak}}$ in loading is equal to three times the applied stress $\sigma$ while the maximum compressive stress occurs perpendicular to the loading direction and is equal $-\sigma$, see Figure 3-3 a). As long as the material behaves linear elastic, the stress fields of different loading conditions can be superimposed. This is illustrated in Figure 3-3 b) for a bi-axially loaded hole. When tensile loads are applied in both directions, $K_t$ decreases compared to the uni-axially loading situation.

In case of a hole that is loaded in shear, the shear loads can be replaced by tensile and compressive load of equal size that are acting on the hole under a 45º angle compared to the shear load direction. This is illustrated in Figure 3-3 c). As a result the absolute $K_t$ values increase to 4.

It is clear from these linear-elastic analysis that a bi-axial tension loading condition reduces the stress concentration while pure shear load or a bi-axial tension-compression loading leads to an increase of the stress concentration.
Available blunt and sharp notch Glare data

\[ (3\alpha - \beta) \sigma \]
\[ \alpha \sigma \]
\[ (\beta - \alpha) \sigma \]

\[ \sigma = \tau \]
\[ 4\sigma \]
\[ -4\sigma \]

Figure 3-3, Stress distribution in an infinite sheet of linear elastic isotropic material around a round hole for different loading conditions.

Figure 3-4 illustrates the calculated tensile stress distribution in x- and y direction and the shear stress distribution along the hole edge, both for a uni-axial loading and a bi-axial loading condition. The material is quasi-isotropic Glare 3 with 50% of the fibres in loading direction and 50% perpendicular to the loading direction and the results are obtained with the theory of elasticity. The stress distribution can be understood by following the stress trajectories along the hole and will be explained for the uni-axial loading condition. The hole edge is stress free. At \( \theta = 0^\circ \), the stress in x direction is perpendicular to the hole edge and must be zero. From here, the stress increases to three times the far field stress \( \sigma \) at \( \theta = 90^\circ \) where all the stress trajectories come together. The stress in y direction is equal to \(-\sigma\).
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at $\theta = 0^\circ$ where the stress trajectories must divert around the hole. However, the surrounding material confines movement of the material and causes compressive loading of the material locally. At $\theta = 90^\circ$, the stresses in $y$ direction at the hole edge must be zero. Following the stress trajectories from $\theta = 0^\circ$ to $\theta = 90^\circ$, the concentration of stress trajectories along the hole edge creates tensile stresses in $y$-direction created by the surrounding material. Finally, the shear stress must be zero at $\theta = 0^\circ$ and $\theta = 90^\circ$. When the stress trajectory is diverting sideward in $y$ direction, the shear stress becomes positive (or negative, depending on the hole side). However, when the trajectory is bend again towards the hole edge, the shear stress changes sign.

![Stress Trajectories Diagram](image)

Figure 3-4, Elastic tensile and shear stress distributions along the hole edge in an infinite Glare 3 3/2 0.3 sheet (quasi isotropic) for different loading conditions (fibres in loading direction).

The bi-axial loading consists of a tensile load in $x$-direction that is twice the tensile load in $y$-direction. It can be seen that the stresses for the uni-axial
load case are more severe than for the bi-axial load case (stress concentration of 3 versus 2.5 respectively). Additionally, the compressive stress in y direction at $\theta = 0^\circ$, has changed into a tension stress equal to $\frac{1}{2}\sigma$ ($= 3 \cdot \frac{1}{2}\sigma - \sigma$ ) and the shear stresses no longer change sign from $\theta = 0^\circ$ to $\theta = 90^\circ$.

Although there are no Glare blunt notch results available in the literature for off-axis loading, the elastic stress distribution under these circumstances will be discussed here as introduction for the test program presented in chapter 5. Figure 3-5 illustrates the calculated elastic stress distribution along the hole edge for an infinite Glare 3 sheet with fibres under an angle of 30º (and -60º) with the loading direction.

Figure 3-5, Elastic tensile and shear stress distributions along the hole edge in an infinite Glare 3 3/2 0.3 sheet (quasi isotropic) for different 30º off-axis loading conditions.
First the uni-axial loading condition will be discussed. Due to the angle between the loading direction and the main fibre direction in the material, the sheet is actually loaded by a uniform tensile load in one fibre direction and a shear load in the other fibre direction. As illustrated in Figure 3-3, both loading conditions cause a stress concentration but at different locations. As a result, the location of the stress concentration is moved along the edge but less than 30º. The stress concentrations in x-direction and in compressive direction in y-direction are reduced while the tensile stress concentration in y-direction is increased. The reduction of the stress concentration in x-direction for the bi-axial loading situation is now much smaller.

When the stress concentrations in material direction are considered (the x and y axes are rotated over 30º), the stress concentration in x direction decreases significantly while the stress concentration in y direction increases and is equal in magnitude. This is illustrated in Figure 3-6. The shear stress in the laminate is significant and is expected to play an important role in the fracture process of the laminate for these large angles.

![Bi-axial load case](image)

Figure 3-6. Elastic tensile and shear stress distributions along the hole edge in material direction in an infinite Glare 3 3/2 0.3 sheet (quasi isotropic) for a 30º off-axis loading condition.

### 3.3 Blunt notch experimental results

This paragraph presents available experimental blunt notch results for Glare laminates. The attention is confined to round, open holes.

The blunt notch strength is no generally used material parameter. Therefore, no standard definition or test practice is available. The definitions of the gross and net blunt notch strength used in this study are...
Available blunt and sharp notch Glare data

\[ \sigma_{\text{bn-gross}} = \frac{P_{\text{max}}}{Wt} \quad \text{and} \quad \sigma_{\text{bn-net}} = \frac{P_{\text{max}}}{(W - id)x} \quad \text{eq. 3-6} \]

where \( P_{\text{max}} \) = maximum load
\( W \) = specimen width
\( t \) = specimen thickness
\( d \) = diameter hole, and
\( i \) = amount of holes in rivet row

\( i \) is only used in case of a row of holes that are evenly spaced like in a rivet row. For a single hole, \( i = 1 \). The definition of an applicable test practice to describe the blunt notch strength in general is discussed in chapter 5.

### 3.3.1 Different Glare variants

Mattousch\(^8\) has carried out a large amount of tests on uni-axially loaded open hole specimens of different Glare variants. The specimens were 25 mm wide and every specimen contained a hole with a diameter of 5 mm. All Glare variants were tested both in the L- and T directions. The net blunt notch results are presented in Figure 3-7, together with the ultimate tensile strength of the different Glare variants. Unfortunately no aluminium data are available for 2024-T3 for this geometry.

![Figure 3-7](image)

Figure 3-7, Net blunt notch results of several Glare laminates.

The first observation from this figure is that the presence of a hole results in a considerable strength reduction. Disregarding the results for Glare 2 in
the T direction that contains only fibres perpendicular to the loading direction, the strength reductions are between 40 and 50%.

A larger amount of anisotropy of the material results in a larger difference between the blunt notch strength in the L- and T directions. In general, the application of thicker aluminium sheets in a certain Glare variant and lay-up (increase of the MVF) results in a decrease of the blunt notch strength in the L-direction. This means that the fibre layers are relatively more effective in carrying the applied load than the aluminium layers in the L direction. The results for the Glare 3 with 2/1 lay-ups illustrate smaller blunt notch strength values than might be expected based on their MVF. The only explanation that can be thought of is the asymmetrical lay-up in these specimens that causes an additional bending moment in the specimen. Comparable results were found by Van Rijn\(^7\), who measured lower net blunt notch strength values for Glare 3 2/1 0.3 than for Glare 3 3/2 0.2.

### 3.3.2 Geometry influence

For a round hole in a finite sheet of isotropic material, the stress concentration factor can be approximated with the Heywood formula:

\[
K_t = 2 + \left(1 - \frac{d}{W}\right)^3
\]  
\[\text{eq. 3-7}\]

According to this equation, \(K_t\) varies between 3 for an infinite sheet and 2 for a hole that approaches the edges of the specimen. Although these results must be factorised for anisotropic materials, see eq. 3-4, the trend remains the same. It also shows that the \(K_t\) remains constant for a constant \(d/W\) ratio.

The influence of the geometry (diameter hole and width of specimen) on the blunt notch behaviour of Glare laminates has been investigated and described by Roebroeks\(^1\). The results of a program on two different specimen widths (40 mm and 100 mm respectively) are plotted in Figure 3-8 as a function of \(d/W\). In the same figure but along the secondary axis at the right side, the factorised \(K_t\) values are plotted for the different \(d/W\) ratios. If the \(K_t\) value should describe the net blunt notch strength of Glare laminates, a higher \(K_t\) value would have resulted in a lower \(\sigma_{bn}\). However, this is not confirmed by these results.
Available blunt and sharp notch Glare data

Roebroeks observed three remarkable tendencies:

- the small hole effect \( (d \rightarrow 0) \), and
- the small ligament effect \( (d \rightarrow W) \), and
- the specimen width effect \((W \text{ increases})\)

The small hole effect

As illustrated in Figure 3-8, the net blunt notch strength increases for decreasing hole diameter and constant width. This occurs in spite of a larger \( K_t \). Roebroeks qualitative explanation is based on the fact that the stress gradient increases for smaller holes and that the plastic zone in the aluminium layer is smaller. This should result in significant shear stresses between the aluminium and fibre layers that are likely to create local delaminations. As will be discussed in chapter 6, delamination is able to postpone fibre failure. If the fibre layer is delaminated, the in-plane shear stress in the delaminated fibre layer may result in longitudinal splitting at the edge of the hole which results in an extra stress reduction.

Van Rijn\(^9\) carried out blunt notch experiments on Glare 3 specimens and varied \( d \) and \( W \). He observed that for a given \( d/W \) ratio, the net blunt notch strength decreased with an increase in notch diameter. This is illustrated in Figure 3-9 for Glare 3 \( 3/2 \) \( 0.2 \) specimens. This is a comparable effect as explained by Roebroeks above for the small hole effect.
The small ligament effect
For specimens with a hole diameter that approaches the specimen width, the Kt factor reaches its minimum as indicated with eq. 3-6 and in Figure 3-8. Therefore, it is expected that the blunt notch strength increases for larger d/W ratio. However, its is remarkable how fast the blunt notch strength increases for large d/W while the decrease of Kt is only minor. Roebroeks explained that the stress concentration at high loads decreases for the extremely small ligaments due to plasticity in the aluminium layers and movement of the ligaments inwards.

The specimen width effect
Van Rijn found that the net blunt notch strength for Glare 3 laminates with a constant hole diameter decreases for increasing specimen width. This is illustrated in Figure 3-9. Also when the d/W ratio is kept constant, the net blunt notch strength decreases for increasing width. This is explained in § 3.6.1.

3.3.3 Multiple holes
Vermeeren investigated the blunt notch strength of ARALL and Glare specimens with multiple holes. The results are illustrated in Figure 3-10. It is clear that the net blunt notch strength results are comparable if the hole pitch (distance between to hole centres) is kept constant. Also these results illustrate for the single notch specimens with a comparable d/W ratio that the blunt notch strength decreases with increasing width.
3.4 Failure mechanism and criteria

3.4.1 Failure mechanism

As described in the previous section, Van Rijn\textsuperscript{a}\引用{9} carried out blunt notch experiments on Glare materials. These experiments were instrumented with a probe to measure the acoustic noise within the material during static loading.

Two characteristic gross stresses were selected to characterize the measured acoustic noises:

- The gross stress at which the acoustic noise level started to increase, \( \sigma_{AE1} \), and
- The gross stress level at which the peak signal attained an offset. \( \sigma_{AE2} \). Approximately at the same stress, the Root Mean Square value of the signal started to increase.

Compared with the measured gross blunt notch strengths, \( \sigma_{AE1} \) occurred in the range between 63 and 74\% of the blunt notch strength while \( \sigma_{AE2} \) occurred between 85 and 92\%.
Van Rijn concluded that the absolute difference between the blunt notch strength and the stress determined from acoustic emission appeared to depend mainly on the $d/W$ ratio. A (preliminary) conclusion he drew is that the damage initiation (first fibre failure), indicated by the acoustic emission results, mainly depends on the notch diameter, whereas the additional load carrying capacity of the specimen is determined by the diameter-to-width ratio.

In a previous research on ARALL laminates, Van Rijn measured the strain distribution in the net section of uni-axially loaded blunt notch specimens as a function of the applied load. Simultaneously, a photo elastic layer bonded to the specimen gave a qualitative view of the strain distribution around the hole. The strain gauges illustrated that the strain in the net section initially increased linearly with the applied load. At a certain load, the material at the notch root became plastic and the strain started to gradually increase more. Increasing the load further resulted in an increase of the plastic zone size in the net section from the notch root towards the specimen boundary. The hole net section was yielding when the strain at the notch root reached the ultimate strain of the Aramid fibres. Approximately at this moment, a strong increase of the strain at the notch root was observed. This is expected to be due to local fibre breakage, although the specimen did not cause immediate failure. This strain increase was also noticed for the strain gauges in the net section further away from the notch root. Images obtained from the photo elastic layer just before and after assumed local failure at the notch root illustrated the occurrence of a strong local strain increase. The typical strain distribution fringes had moved to the edge of the local failure.

Roebroeks chemically removed (etching) the outer aluminium layers of some blunt notch specimens. He clearly observed indications of the presence of delamination between the outer aluminium layers and the adjacent fibre layers while the fibres were still intact. After the occurrence of delamination, the in-plane shear stress in the fibre layer can result in longitudinal splitting at the edge of the hole. Also this was observed in some of the etched specimens.

3.4.2 Lamina failure criteria

Based on the observed failure phenomena in the previous paragraph, it is complicated to define applicable failure criteria. It is expected that failure is in general initiated by fibre failure; however, the occurrence of delamination and the plasticity of the aluminium layers will delay direct failure of the material after local fibre failure at the hole root. When the principal loading
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directions are under large angles (± 45º) compared to the fibre directions, it is expected that failure is dominated by the aluminium properties.

In this paragraph an overview of the most relevant failure criteria that are expected to be applicable to FML’s are described and reviewed\textsuperscript{11}. Apart from local plasticity in the aluminium that could be categorized as local failure too, the following failure modes can be distinguished:

- Fibre breakage
- Transverse matrix cracking (loading perpendicular to fibres)
- Shear matrix cracking (splitting between fibres)
- Laminate delamination (between fibres and resin or between resin and aluminium layers)

Since a general delamination criterion is difficult to obtain, delamination will be disregarded which is expected to result in conservative predictions.

Lamina failure criteria for the first three failure modes can be categorized into three groups; limit criteria, interactive criteria and separate mode criteria. The first two will be described shortly.

**Limit criteria**
These criteria do not take in account any interaction between the stresses in different principle directions to predict failure. The three lamina stresses are compared with the corresponding strengths. Failure is expected if:

\[
\frac{\sigma_{11}}{X} = 1 \quad \text{fiber failure,} \\
\frac{\sigma_{22}}{Y} = 1 \quad \text{transverse matrix cracking} \\
\frac{\tau_{12}}{S} = 1 \quad \text{shear matrix cracking.} \\
\]

in which X and Y are the lamina tensile strength in L and T direction. These equations are generalized for either tensile or compressive stresses; the corresponding strength must be chosen based on the sign of the applied stress.
Interactive criteria

These criteria predict the failure load by using a polynomial equation including all stress components. Failure is assumed when the equation is satisfied. An example is the Hill criterion:

\[
\left( \frac{\sigma_{11}}{X} \right)^2 - \left( \frac{1}{X^2} + \frac{1}{Y^2} - \frac{1}{Z^2} \right) \sigma_{11} \sigma_{22} + \left( \frac{\sigma_{22}}{Y} \right)^2 + \left( \frac{\tau_{12}}{S} \right)^2 = 1 \quad \text{eq. 3-9}
\]

where Z is the lamina strength in thickness direction and S is the shear strength. For GLARE3 the strength of the prepreg lamina is the same in both material directions and therefore X = Y.
3.5 Sharp notch behaviour

3.5.1 Residual strength; a material parameter?

The residual strength of a material is defined as the remaining static strength of the material in the presence of any damage that can occur during the service life of an aircraft. This is an imprecise definition since the term damage is not defined and can vary between a smooth dent and a sharp crack. For design verification it is conservative to represent damages as sharp cracks.

The residual strength of a material depends on the size of the damage or crack present. One of the aims in the field of fracture mechanics is to find a set of parameters that describe the residual strength of a material independent of the geometry. In other words, can the residual strength be described with a material parameter?

Within the fracture mechanics field, several fracture-predicting parameters or methods have been developed, like the stress intensity factor K, the crack resistance or R-curve, the J-integral and more localised parameters like the Crack Tip Opening Angle (CTOA). Their applicability depends on the brittleness of the material and/or the ability to allow stable crack growth before fracture as in metals. The next paragraph therefore, describes the observed fracture behaviour of Glare.

3.5.2 Fracture mechanism for through-the-thickness cracks

Glare material with a through-the-thickness crack that is loaded to fracture exhibits a capability for slow stable crack growth prior to rapid failure. This rapid failure or unstable crack growth occurs when the specimen is loaded under force control. The fracture behaviour is similar to the response of monolithic aluminium sheet material. Figure 3-11 shows a typical curve of the gross stress versus the physical crack extension for a Glare laminate, obtained with a uni-axially loaded Centre Cracked Tension (CCT) specimen. The crack extension is the extension measured in the visible outer aluminium layers.

De Vries and Pacchione carried out residual strength tests on Glare 2 and Glare 3 laminates with initial saw cuts and followed the fracture process with a high-resolution Charged Coupled Device (CCD) camera. The tested specimens were CCT specimens. Figure 3-12 shows some images of the crack tip in a uni-axially loaded Glare 2 specimen. The initial saw cut in this
specimen was made with a jeweller’s saw and the specimen was first fatigue loaded to sharpen the starter crack.

![Figure 3-11, Typical crack extension versus applied load curve for Glare.](image)

**Fatigue pre-cracking**
The crack extension created during fatigue loading is visible in Figure 3-12 a). During fatigue cycling, the fibres stay intact and bridge the crack in the aluminium layers. Consequently at the start of the residual strength test, the crack length within the prepreg layers differs from the crack length in the aluminium layers. Due to the fatigue pre-loading, debonding occurs between the fibre prepreg and the aluminium layers above and below the fatigue crack. This kind of debonding is called “fatigue” delamination and occurs when the fibres are still intact.

**Static loading**
During static loading, the fatigue crack in the aluminium layers opens. This is visible in Figure 3-12 b). At a certain load, the fatigue crack in the aluminium layer starts to propagate. The fracture process generates slant crack faces, differing from the fatigue crack faces as the brighter crack surface. Also the creation of plastic zones around the crack tip is visible.

**Fracture process**
At approximately 90% of the residual strength of the specimen, the first fibre bundle in the visible prepreg layer fails at the tip of the fatigue crack. This is visible in Figure 3-12 c) where the broken bundle appears brighter compared to the surrounding prepreg layer that is still intact. During further
load increase, random failure of fibre bundles is observed between the tip of the saw cut and the tip of the fatigue crack in the aluminium layer.

It is expected that the crack extensions in the prepreg layers and the aluminium layers become approximately equal during the remaining fracture process. However, this is not visible due to the slanted crack faces of the outer aluminium layer. Because the 4.5% ultimate strain of the glass fibres is significantly smaller than the 19% for 2024-T3, the glass fibre layer is expected to control further crack extension up to fracture. After fibre failure, the adjacent aluminium layer is unable to take over the high loads and will consequently fracture. During static loading, “static” delamination is created due to high strain gradients and plasticity in the aluminium layer. This reduces the strain in the glass fibres and consequently delays fibre failure.

Figure 3-12, Crack growth sequence for a Glare 2 CCT-specimen\textsuperscript{12}.
Fracture surface
Figure 3-12 d) (notice different magnification) is taken after final failure and shows the length of the protruding broken fibres. The length of the protruding fibres is an indication of the static delamination size since the fibres tend to break at the edge of the static delaminated area and not at the same location as the fracture surface in the aluminium layers. The maximum normal stress in the fibres occurs at the edge along the boundary of the delamination area and is therefore the most likely location for the fibres to fracture. This is illustrated in Figure 3-13.

Figure 3-13, Illustration of normal stress distribution in fibre prepreg layer due to delamination.

The length of the protruding fibres is significantly longer for the first two millimetres of crack extension. This is possibly due to the bi-axial stress state in the aluminium in contrast with the uni-axial stress state in the glass prepreg that caused static delamination. This will be further discussed in chapter 6.

When a fibre bundle fractures, the elastic energy stored in this bundle must be transferred suddenly via the matrix to the neighbouring load carrying members. This process causes delamination also and is often referred to as "dynamic" delamination. To prevent confusion for the reader between fatigue and dynamic delamination, this report will use the term “fibre failure delamination”. After the residual strength tests, the outer aluminium layers of several specimens were chemically removed to determine the extent of fibre failure delamination. This is illustrated in Figure 3-14. A small semi-elliptical static delamination is visible at the crack tip of the Glare 2 specimen. This area is followed by delamination caused by fibre failure during the residual strength test, i.e. fibre failure delamination. For the tested 3/2 lay-ups, the etching method can only detect fibre-failure-
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delamination at the interface between the outer aluminium layers and the adjacent prepreg layers. The observed delamination occurred between fibre rich and resin rich interface in the prepreg layer at the side of the outer aluminium layer. The fibre-failure-delamination length does not show any correlation with the length of the protruding fibres.

Figure 3-14, Static- and fibre-failure-delamination after fracture.

The static delaminated area in the Glare 3 laminate, i.e. the length of the protruding fibres, is slightly smaller than for Glare 2 and the fibre failure delamination is significantly smaller, see Figure 3-14. This is due to higher loads at failure in Glare 2 and the corresponding larger elastic spring back after fibre fracture.
3.5.3 **Standard test procedures**

All fracture toughness tests described in this report are performed on the basis of the ASTM E561\textsuperscript{13} practice for R-curve determination, using flat CCT panels, see Figure 3-15. If not mentioned explicitly, the specimens contain through-cracks what means that both the fibre- and the aluminium layers are cut.

![Diagram of CCT specimen](image)

Clamping area

$2a_0$ = length centre crack
$W$ = gross width
$L$ = free length between clamps
$\sigma$ = applied gross stress

**Figure 3-15, Centre Cracked Tension (CCT) specimen geometry.**

The majority of the specimens discussed in this chapter are either tested in L-T or T-L direction. The first letter reflects the direction of the external load relatively to the rolling direction of the aluminium and the second letter the direction of the crack. L stands for Longitudinal or in rolling direction and T indicates Transverse or perpendicular to the rolling direction.

**L - T:**
- *direction of the crack:* $T$ = perpendicular to the rolling direction of the Al-layers
- *direction of the external load:* $L$ = in the rolling direction of the Al-layers
3.6 Residual strength results of Glare laminates

This paragraph gives an overview of research done and the available residual strength data of Glare laminates available in literature. Several aspects that influence the residual strength are discussed like:

- the specimen geometry (§ 3.6.1),
- the rolling direction of the aluminium layers (§ 3.6.2),
- post stretching and temperature effects (§ 3.6.3),
- crack direction compared to the material orientation (§ 3.6.4),
- crack edge buckling (§ 3.6.5) and
- fatigue cracks and artificial damage versus impact damage (§ 3.6.6 and § 3.6.7)

Figure 3-16, Figure 3-17 and Figure 3-18 contain residual strength results, \(\sigma_c\), found in literature\(^2\,12\,14\) for respectively Glare 2A, Glare 3 and Glare 4A. \(\sigma_c\) is defined as the maximum gross stress that the CCT specimens could carry before final fracture. The results are plotted as a function of the relative initial crack length \((2a_0/W)\) for several different specimen widths \(W\).

\[\text{Relative initial crack length } 2a_0/W \]

\[\text{Residual strength } [\text{MPa}]\]

**Figure 3-16, Residual strength as a function of relative initial crack length (saw cut) for Glare 2 2/1 0.3 and Glare 2 3/2 0.3.**
Figure 3-17, Residual strength as a function of relative initial crack length (saw cut) for Glare 3 3/2 0.3.

Figure 3-18, Residual strength as a function of relative initial crack length for glare 4A compared with Glare 2 and Glare 3 results.

The amount of available data is rather limited and mainly for Glare 2A and Glare 3 with three 0.3 mm thick aluminium layers and loaded in the rolling direction of the aluminium layers. For Glare applied in an aeroplane, this would be representative for cracks in circumferential direction.
Figure 3-18 compares the residual strength results of three different Glare variants for 400 mm wide CCT specimens. It illustrates the residual strength superiority of Glare 2 L-T followed respectively by Glare 4A L-T, Glare 3 L-T and Glare 4A T-L. This is obviously related to the amount of fibres in loading direction in the intermediate prepreg layers of respectively 100%, 67%, 50% and 33%.

3.6.1 Influence of specimen geometry

1. Shape of specimen
Zaal\textsuperscript{15} investigated the residual strength of Glare with Compact Tension (CT) specimens, see Figure 3-19, and faced several problems;

- The load introduction gave rise to bearing problems in the laminate,
- buckling, defined as the out of plane displacement of the crack edges, was difficult to prevent and
- the initially straight crack became curved during crack extension.

Figure 3-19, Geometry Compact Tension Specimen and illustration of specimen related difficulties for residual strength testing.

Since these problems do not occur or can be easily prevented by using CCT specimens (Figure 3-15) these specimens are more appropriate for residual strength investigations on Glare laminates.
2. Relative crack length and specimen width

The residual strength results illustrated in Figure 3-16 and Figure 3-17 illustrate the influence of the CCT specimen geometry:

- $2a_0/W = \text{constant}$: the residual strength increases with decreasing specimen width
- $W = \text{constant}$: the residual strength decreases for increasing relative crack length

The results for $W=\text{constant}$ is not difficult to understand since the net section that has to transfer the load along the crack decreases while the gross section $(W \cdot t)$ remains the same, see Figure 3-20a.

![Diagram](image)

- **a)** Reduction of $\sigma_c$ for constant $W$ and increasing $2a$

![Diagram](image)

- **b)** Reduction of $\sigma_c$ for constant $2a/W$ and increasing $W$

Figure 3-20, Influence specimen geometry on residual strength.
The fact that the residual strength increases for constant relative crack length and decreasing specimen width can be explained with Figure 3-20b. The residual strength is defined as the maximum load in y- or loading direction that the net-section can sustain, divided by the gross section. Based on the linear elastic theory of elasticity and the assumption that the plastic zone size is small compared to the specimen size, K describes the stress distribution around the crack tip and the size of the plastic zone. Under these circumstances, a similar stress distribution is established around the crack at fracture for laminates of the same type (Glare 2A, Glare 3, etc.). Looking to Figure 3-20b explains that the highly stressed area at the notch root results in a higher residual strength for smaller specimen width because the average stress in the net section and thus the residual strength is higher for the specimen with a smaller width. This explanation holds as long as the material behaves mainly elastic. In case of large plastic zones, the explanation must be found in the fact that specimens with a larger width absorb more energy. When this larger amount of energy becomes available for the creation of fracture surfaces, which is supposed to be independent of the size, it is obvious that the failure stress for a larger specimen is lower.

Especially for residual strength testing of aluminium alloys with elastic-plastic material behaviour, the width of the specimens and the relative crack length need to be chosen carefully. For these alloys it must be avoided that the net section starts to yield since this influences the failure mode. It is not clear what is the influence of net section yielding on the failure mode of Glare since this material behaves bi-linear and illustrates significant strain hardening after yielding of the aluminium layers in the net section. This will be discussed in § 3.7.2.

### 3.6.2 Influence of rolling direction aluminium

There is not much data available about the influence of the rolling direction of the aluminium layers on the residual strength of Glare. Research by De Vries and DERA, reported by Schwarmann illustrated that the residual strength of the aluminium alloy 2024-T3 loaded in rolling direction is in the order of 10% larger than perpendicular to the rolling direction.

The influence of the aluminium rolling direction on the residual strength of Glare can only be properly determined if the results of so called Glare A variants are compared with Glare B variants (for example Glare 4A 3/2 0.3 with Glare 4B 3/2 0.3). In this case, only the direction of the aluminium
layers changes in the laminate while the lay-up and the direction of the different fibre layers remain the same.

It is less correct to draw conclusions for the rolling direction influence from tests on the same laminate but tested in different directions. In these cases also the direction of the fibre layers changes what can have an influence on the support of the aluminium layers by the adjacent fibre layers. However, no data are available for A and B variants. Vermeeren\textsuperscript{2} found for two 800 mm wide Glare 3 3/2 0.3 specimens that the residual strength in L-T direction was 10\% higher compared to the T-L direction. Bresser\textsuperscript{19} found a smaller difference of 2.5 \% in favour of Glare 3 3/2 0.3 L-T for a single, 395 mm wide specimen. These results are illustrated in Figure 3-35 for the data points (100+0).

Based on this small amount of data, it seems that the rolling direction of the aluminium layers in the laminate has an influence on the residual strength but that this influence is less significant than for monolithic aluminium.

### 3.6.3 Influence of pre-stretching and temperature

An important item for the use of Glare as a fuselage material is its behaviour for low and high temperatures. Horst \textit{et. al.}\textsuperscript{20} have investigated the influence of pre-stretching and temperature on the sharp notch strength of Glare. Both effects have in common that they change the internal stress system within the laminate. This paragraph starts to give a qualitative explanation of both effects.

#### Influence of temperature

Glare is cured at an elevated temperature. Due to differences in the thermal expansion coefficients of the prepreg and aluminium, cooling the laminate to room temperature gives rise to:

- compressive stresses in the prepreg layers, compensated by
- tension stresses in the aluminium layers.

This is illustrated in Figure 3-21a. The differences become larger if the temperature decreases for example to the outside temperature at cruising altitude.

#### Influence of pre-stretching

Stretching a Glare laminate after its curing process can modify the established residual stresses created during curing. If the laminate is
Available blunt and sharp notch Glare data

stretched until the aluminium layers become plastic, a plastic deformation will remain in the aluminium layers after unloading. This is illustrated in Figure 3-21b. The fibres are elastic and want to retract to their original position, which will be stopped by the elongated aluminium layers. In this way, the tension stresses in the aluminium and the compressive stresses in the prepreg are reduced or even reversed. One should realize that pre-stretched Glare behaves differently only for tensile stresses between the yield stress of the unstressed material and the newly created yield stress. In this region, the aluminium in the pre-stretched Glare is still elastic with the original Young's modulus while the aluminium in the baseline material starts to yield, which reduces the Young's modulus considerably.

Test results and discussion
Horst et al. carried out blunt notch and residual strength experiments on 100-mm wide specimens with holes and notches with a diameter or crack length of 25-mm. They tested cured as well as pre-stretched Glare and found that both the blunt notch and the residual strength increased with increasing pre-stretching. Their explanation was that the plasticity of the material in the net section is retarded due to the pre-stretching. If the material at the notch becomes plastic, the elastic material in the net section attracts the load away from this area, since its stiffness is higher. As a result, the load carried by the net section will be higher, as long as the net section does not fully yield for the pre-stretched material.

Horst also found an increase of the notch strength for lower temperatures. This is more complicated to explain. When the temperature is decreased, the residual tensile stresses in the aluminium layers are increased, which is...
expected to result in a lower apparent yield stress of the laminate. According to the explanation for the pre-strain effect, this should result in lower notch strength. However, Horst found an effect of the temperature on the Glare stiffness, see Figure 3-22. Assuming that the aluminium static properties do not change significantly between –50 and 80 degrees Celsius, the prepreg stiffness increases for lower temperatures. Based on these experimental results, the increase of stiffness seems to overrule the temperature decrease with respect to the yield strength. Consequently, the yield stress increases for a decrease in temperature. This is confirmed by the experimental results presented in Figure 3-22. Thus, comparable for the results of pre-stretching, the blunt and sharp notch strengths increase for lower temperatures like at cruising altitude.

![Figure 3-22, Influence of temperature on the yield strength and the Youngs Modulus of Glare 3 3/2 0.3²°](image)

**3.6.4 Material orientation and crack orientation**

Teigen²¹ and Zaal²² investigated the influence of the material orientation and the crack orientation on residual strength. Figure 3-23 shows the difference in specimen definition; Teigen tested 100 mm wide CCT specimens with an initial saw cut of 25 mm and varied the material orientation while Zaal tested 160 mm wide specimens with 40 mm long initial cracks and varied the crack orientation. The difference in loading of the cracked area for these different specimen definitions is explained in Figure 3-23b;
• in the case of Zaal, the angled crack is only loaded by a normal stress $\sigma$ in fibre direction, while
• the crack in the specimen of Teigen is loaded by a normal stress $\sigma$ in fibre direction and an additional shear stress $\tau$.

Zaal found that the cracks in his specimens with cracks under an angle relative to the loading, propagated in the direction perpendicular to the load. His residual strength results are plotted in Figure 3-24 as a function of the projected relative crack length. This is the projection of the initial crack length in the direction perpendicular to the load. The obtained results are compared with residual strength results obtained with 200 mm wide specimens with a crack perpendicular to the loading direction. They show the same trend. It is therefore expected that not the angle of the crack but the amount of fibres cut perpendicular to the loading direction and therefore
the projected crack length is more important for the residual strength. This agrees with the results of Vermeeren\(^2\), discussed in § 3.6.

![Figure 3-24](image) Residual strength results as a function of crack orientation and projected crack length. Material was Glare 3/3/0.3.

Due to the varying material orientation in the research of Teigen, the material is actually loaded in tension and shear. The maximum shear stresses occur if the fibre orientation \(\phi\), makes an angle of 45° with the loading direction. The experimental results for Glare 3 illustrate a minimal residual strength for \(\phi = 45^\circ\), see Figure 3-25. This is expected since the fibre-prepreg can only handle tension and compression stresses and does not contribute in carrying shear stresses. These have to be carried by the available aluminium layers and to some extent by the matrix. It is expected that the relation presented in Figure 3-25 is symmetrical around 45° neglecting the rolling direction of the aluminium layers. However, no data were available for 60° and 75° to check this.

Glare 2 does not have any fibres in the 90° direction. Therefore, the residual strength in 90° direction is even lower than for the 45° direction, see figure 2.9. For these large angles between the load and the direction of the fibres, the Glare 2 material with the larger MVF has a higher residual strength while it is the opposite for small angles. This is simple to explain since the fibres contribute significantly to the residual strength for small angles but become ineffective at larger angles where the aluminium layers “have to do it on their own”.
Available blunt and sharp notch Glare data

![Graph showing residual strength as a function of material orientation]

Figure 3-25, Residual strength results as a function of the material orientation. Thus it can be concluded that

- the amount of fibres cut perpendicular to the loading direction is more important for the residual strength than the direction of the crack with respect to the loading direction, and
- that the application of shear stresses has a negative effect on the residual strength.

3.6.5 Influence of buckling

Buckling causes an out of plane displacement of the crack edges and is caused by compressive stresses along the crack edges in a sheet that is loaded in tension. This is illustrated in Figure 3-26a, which shows a statically loaded CCT specimen. The trajectories on the image are reflections that represent areas with a comparable out of plane deformation.

Due to buckling, the material at the crack tip is no longer loaded in mode I only, but is confronted with a combination of loading modes. This creates a more severe stress field at the crack tip, which results in lower residual strength values.

During testing, buckling can be prevented to a large extent by mounting Anti Buckling (AB) guides to the specimen, see Figure 3-26b. The influence of buckling for Glare is depicted in Figure 3-27 and Figure 3-28. Like for aluminium alloys, the influence of buckling is significant.
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Figure 3-26, Illustration of a) buckling in a CCT specimen and b) the application of AB guides.

Figure 3-27 Residual strength as a function of relative crack length and buckling constraint for Glare 2 2/1 0.3.
Available blunt and sharp notch Glare data

Figure 3-28  Residual strength as a function of relative crack length and buckling constraint for Glare 3 3/2 0.3.

3.6.6  Fatigue cracks

D’Allesandro Caprice and Vermeeren$^{2,23,24}$ investigated the creation of delamination during CA-fatigue loading as a function of both the notch geometry and the maximum stress level. Subsequently, they investigated the influence of delamination on the residual strength.

Figure 3-29,  Residual strength as a function of relative crack length and maximum fatigue stress level for Glare 3 3/2 0.2.
Vermeeren² created fatigue cracks starting from a 3 mm hole or a 3 mm saw cut in Glare 3 3/2 0.3 material. He observed that a more blunt starter notch results in a considerable increase of the number of necessary fatigue cycles and the size of the delamination. By increasing the maximum fatigue stress levels from $\sigma_{\text{max}} = 120 \text{ MPa}$ to 150 MPa, the amount of necessary load cycles to obtain a certain crack length decreased with more than 60% but the delamination size doubled. Subsequent residual strength testing of the Glare 3 specimens illustrated that the residual strength increased with larger statically delaminated area. This is illustrated in Figure 3-29. The experiments of Vermeeren were carried out during the “early Glare days” when the prepreg, the pre-treatment and the curing process were far from optimised. It was tried to compare the results of Vermeeren with more recent data. However, the residual strength of the recent material is improved and was therefore left out of the comparisons.

![Diagram](image)

**Figure 3-30,** Schematic overview of fracture process within Glare 2 and Glare 3, originating from a fatigue starter crack.

Compared to the extensive stable crack growth in statically loaded specimens with through cracks, the specimens with fatigue cracks did not
Available blunt and sharp notch Glare data

show any stable crack growth at all during residual strength testing. D'Allesandro Caprice et al. observed differences between the fracture mechanism of statically loaded fatigue cracks in Glare 2 and Glare 3. This is illustrated in Figure 3-30, which shows the starter crack and the development of the crack for both materials. During loading the crack opened slowly in both materials, effectively restrained by the intact fibres. At loads close to fracture, the delamination in Glare 2 grew in the fibre direction and resulted in larger crack opening of the aluminium layers. Fracture initiated as fibre failure near the fatigue crack tip while the fibres in the wake of the fatigue crack remained intact.

Figure 3-31, Relation between the residual strength and the size of the fatigue delaminated area obtained by creating the fatigue starter crack, Glare 2 2/1 0.2.

Figure 3-31 shows the residual strength and the delaminated area of Glare 2 as a function of the relative fatigue crack length for two different maximum fatigue stresses. It illustrates that a higher \( \sigma_{\text{max}} \) of 150 MPa resulted in a larger delamination area, which of course increases with the relative crack length. If we now look to the residual strength, we see that the residual strength of the fatigue crack created with \( \sigma_{\text{max}} = 150 \) MPa and thus with larger statically delaminated area results in a lower residual strength. This can be understood from the fact that a larger statically delaminated area will result in more crack opening and thus in a higher stress in the fibres at the fatigue crack tip.

This is the opposite compared to the results obtained for Glare 3. D’Allesandro Caprice found that fracture in Glare 3 is initiated in the fibres
near the hole at the centre of the fatigue crack, see Figure 3-30. This results in larger crack opening and subsequent fibre failure from the centre towards the tips of the fatigue crack. After all the bridging fibres failed, an almost sudden fracture occurred. Now, a larger statically delaminated area will delay fibre failure at the notch root and increase the residual strength. This is illustrated in Figure 3-32 where the residual strength and the delaminated area for two different maximum fatigue stresses are plotted as a function of the relative crack length for Glare 3.

![Figure 3-32](image)

Based on the opposite results of Glare 2 and Glare 3, it seems as if there is an optimum in the amount of static delamination that is favourable for the residual strength of a laminate:

- If the amount of fatigue delamination is less than the optimum, the load in the aluminium layers is not effectively transferred away from the fibres around the notch. Consequently, these fibres will fracture first after reaching their ultimate strain. This is the case in the Glare 3 material that was tested.
- If the amount of fatigue delamination is larger than the optimum, a too large amount of delamination of the fibre layers in the wake of the fatigue crack decreases the effectiveness of the crack bridging and causes fibre failure at the tip of the fatigue cracks. This was the case in the Glare 2 material that was tested.
Available blunt and sharp notch Glare data

Figure 3-33, Residual strength as a function of relative crack length and initial crack creation for Glare 2/1 0.3.

Figure 3-34, Residual strength as a function of relative crack length and initial crack creation for Glare 3/2 0.2.

The residual strength increase of a fatigue crack compared to a saw-cut of the same length is significant. This is illustrated in Figure 3-33 and Figure 3-34 for Glare 2A 2/1 0.3 and Glare 3 3/2 0.3 respectively. Comparable improvements were found for Glare 4A25,26. The residual strength of the fatigue-cracked specimens is nearly independent from the relative initial
crack length. Thus, the amount of fibres cut and the amount of delamination determine the residual strength.

Bresser and Dietrich\textsuperscript{19,27} investigated the effectiveness of the crack bridging as a function of the fatigue crack length. Several 395-mm wide Glare 3 3/2 0.3 CCT specimens with different initial saw cut lengths were subsequently fatigue loaded to obtain a combined crack of 100 mm length. The results, combined with results from Vermeeren\textsuperscript{2} for an almost full fatigue crack of 100 mm, are presented in Figure 3-35. Although test results with different maximum fatigue stress are compared, it is clear that the residual strength decreases rather rapidly if the ratio saw cut over total crack length increases. In this case the total crack is defined as the initial saw cut length plus the additional fatigue crack with only aluminium layers broken.

![Figure 3-35](image)

Figure 3-35, Residual strength results of Glare 3 specimens with initial saw-cuts that were additionally loaded by fatigue\textsuperscript{2,19,27}.

Although Bresser does not mention this in his report, it is expected that the crack initiate at the "saw cut crack tip" during the residual strength test, like at the blunt notch edge in Figure 3-30. Figure 3-36 shows the same results in L-T direction, but now compared with Glare 3 3/2 0.3 L-T residual strength results with different initial saw cuts that were not additionally fatigued. It is clear that the trend between the two starter cracks with the initial saw cut length is the same and that the saw cuts with fatigue crack extension can be seen as saw cuts only with an extra residual strength reduction because of the cracked aluminium layers in front of the saw cut.
Available blunt and sharp notch Glare data

Figure 3-36. Residual strength results of Glare 3 specimens with initial saw-cuts compared to saw cuts that were additionally fatigued.

3.6.7 Artificial saw cuts versus "realistic" foreign object damage

All research presented in this section was based on artificial created fatigue cracks or saw cuts. The question, especially for Glare, is whether these artificial damages are representative for a DSD. Bresser and Dietrich\textsuperscript{19} investigated the residual strength of 395 mm wide Glare CCT specimens that were impacted with sharp objects and compared them with saw-cuts of the same length.

Two different 100 mm wide sharp harpoon-like impactors were used; one with a thickness of 2.4 mm and one with a thickness of 4.8 mm. The impactors were either pushed slowly through the specimens by a hydraulic device or fast by dropping them from a defined height along a guidance. Unfortunately, the mass of the impactor and the speed at impact or the kinetic energy are not mentioned in this reference. This makes comparison with data from other references difficult.

The delaminated area around the crack tips was found to be small for all impact tests. Consequently, no improvements were found for the residual strength and, therefore, a saw cut seems to be representative for DSD. The results for Glare 3 3/2 0.3, both tested L-T and T-L are plotted in Figure 3-37. All impact and saw cut results show a scatter of only 3.7 % in T-L direction and of 1.7% in L-T direction.
3.7 Available fracture mechanics concepts for Glare and their applicability

3.7.1 Introduction

To successfully predict fracture behaviour and residual strength of materials, a fracture criterion independent of the width of the specimen and the length of the crack is required. This chapter discusses the applicability of the engineering stress intensity factor and the R-curve as characterising residual strength material parameters for Glare laminates with accidental damage.

Tests on Glare 2 and Glare 3 CCT specimens have shown that these materials exhibit slow stable crack growth prior to rapid fracture similar to the response of monolithic aluminium sheet. To illustrate the stable crack growth process, consider a central starter crack of length $2a_0$ situated in an infinite plate, see Figure 3-38. This plate is remotely loaded by a uniform uni-axial (mode I) tensile stress. The sheet is thin and in plane stress condition. When the specimen is loaded to a stress $\sigma_i$, the crack starts to grow slowly, as depicted in the graph of Figure 3-38. This crack growth is stable, i.e. crack propagation can be maintained only if the load is increased. When the stress is increased to $\sigma_c$ the crack has reached a...
critical length $2a_c$, where fracture instability occurs. The longer the initial crack, the lower the values of $\sigma_i$ and $\sigma_c$ and the larger the amount stable crack extension $a_c-a_i$.

Figure 3-38, Stable crack extension.

### 3.7.2 Engineering Stress Intensity

Irwin$^{28,29}$ showed that the magnitude of the stresses in the vicinity of the crack tip in an elastic material can be described with the stress intensity factor $K = \sigma \sqrt{\pi a}$. Thus, cracks in an elastic material with different sizes but with the same $K$ have similar local stress fields around the crack tip, also at fracture. The question is whether $K$ still describes the stress field when plastic deformations take place at the crack tip. Broek$^{28}$ explains that $K$ can still describe the crack behaviour in this case as long as the plastic zones are small compared to the specimen size.

The critical stress intensity factor at fracture, $K_c$, is based on the residual strength and the critical crack length. However, the critical crack length is difficult to determine in materials with stable crack extension since the crack extension is often very fast close to fracture when loaded force controlled. This can be seen in Figure 3-11. Therefore, this report uses the engineering stress intensity factor that is based on the initial crack length instead of the critical crack length:
\[ K_e = f(\sigma_c, a_o) = \beta \cdot \sigma_c \sqrt{a_o} = \frac{1}{\sqrt{\cos \left( \frac{\pi a_o}{W} \right)}} \sigma_c \sqrt{a_o} \]  

\text{eq. 3-10}

where:  
- \( a_o \) = initial crack length,  
- \( \sigma_c \) = residual strength,  
- \( W \) = specimen width and  
- \( \beta \) = the Feddersen finite width correction factor

Figure 3-39, Engineering Stress Intensity Factor \( K_e \) for different Glare laminates as a function of the specimen width.

The results for some different Glare laminates are presented in Figure 3-39. The \( K_e \) –values are largely dependent on the width. For aluminium materials these lines but tend to go to a more or less constant value for infinite width. This might be the case for Glare also. At fracture, the average stress in the net section was above the yield stress for all specimens. The onset of this event is defined as Net Section Yielding (NSY) and will influence the fracture mechanism for aluminium alloys. Some of the 400 and 800 mm wide Glare specimens were instrumented with strain gages along the edges in the net section. The strain gage readings showed that the 400 mm specimens were fully yielding in the net section at fracture while the 800 mm wide specimens showed strains clearly below the yield strain of the laminates. It is expected that the fracture mechanism for Glare will not change due to NSY due to the large strain hardening capacity of the
Available blunt and sharp notch Glare data

laminate that allows stresses in the net section that are much larger than the yield stress of the laminate.

Also the variation of $K_e$ with the relative crack length is rather large as indicated in the figure for some clouds of data points. It is clear that $K_e$ is no material constant in the range of specimen widths tested (100 to 800 mm). This was concluded in several investigations$^{2,14,27,30}$. Vermeeren$^2$ showed that $K_e$ is actually no material constant for 2024-T3 either. Based on the large amount of plasticity and the occurrence of stable crack extension that is neglected in the use of $K_e$, the application of the R-curve concept was investigated.

3.7.3 R-curve approach

The R-curve approach first introduced by Irwin$^{31,32}$, is able to cope both with stable crack extension and a limited amount of plasticity. Several researchers investigated whether the R-curve concept as applied for metallic sheets is also applicable for Glare$^{2,12,14,26,17-19,33}$. Prediction of the residual strength of fatigued specimens using the R-curve approach is not possible since there is no stable (physical) crack extension.

3.7.3.1 Method description

The R-curve concept is based on an energy balance;

\[ \text{To obtain fracture the energy available for crack growth or energy release rate } G \text{ must be larger than the crack resistance } R \text{ of the material.} \]

In general, stress intensity units are used instead of energy release rate units. The relation between them in case of plane stress is:

\[ G = \frac{K_G^2}{E} \quad \text{and} \quad R = \frac{K_R^2}{E} \quad \text{eq. 3-11} \]

where $E$ is the Young’s modulus of the material.

Where the terms crack driving force and energy release rate or crack growth resistance are used in the remainder of the text, they refer to the corresponding stress intensities $K_G$ and $K_R$. 

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Figure 3-40, Schematic $K_R$-curve and crack driving force curves for thin centre cracked plates, corrected for finite size.

Figure 3-40 shows a schematic energy-balance diagram in terms of the energy release rate $K_G$ and the crack growth resistance $K_R$. When a central crack of length $2a_o$ is present, slow crack growth commences at the initiation stress $\sigma_i$. At point A, where $K_G(\sigma_i) = K_R$. Further crack growth does not occur at this stress since $K_G(\sigma_i) < K_R$ for crack lengths larger than $2a_o$. However, further increase of the stress to $\sigma_c$ induces slow crack growth to B. If the stress is raised further to $\sigma_c$, $K_G$ remains larger than $K_R$ for whatever crack length. Therefore, the onset of instability is reached at point C where:

$$\frac{\partial K_G}{\partial a} = \frac{\partial K_R}{\partial a} \quad \text{and} \quad K_G = K_R$$

eq. 3-12

The crack growth between $\sigma_i$ and $\sigma_c$ is stable since it stops when the stress is kept constant. During this slow stable crack growth, there is a continuous balance between released and absorbed energy. If there were no balance, then either crack growth would stop ($K_R > K_G$) or become unstable ($K_G > K_R$).

$K_G$ is given by
Available blunt and sharp notch Glare data

\[ K_G = \sqrt{\frac{G}{E}} = \frac{1}{\sqrt{\cos(\pi a/W)}} \cdot \sigma \sqrt{\pi a} \]  

**eq. 3-13**

The shape of the \( K_G \) -curve and the value of \( a_0 \) will affect the tangent point on the \( K_R \) -curve so that \( K_G \) at point C in Figure 3-40 cannot be a material constant. Krafft *et al.*\(^{30}\) suggested the \( R \)-curve to be invariant for given sheet thickness i.e., the \( R \) or \( K_R \) -curve is independent of the initial crack size, see Figure 3-41. This implies that the fracture condition for a crack \( a_1 \) of larger size than \( a_0 \) follows from the tangent \( K_G(\sigma_{c1}) \) drawn to the unique \( K_R \) -curve. Similarly, the crack of length \( a_1 \) leads to failure at a stress \( \sigma_{c1} \) in point D.

![Diagram](image)

**Figure 3-41.** The \( K_R \)-curve as a material property for different initial crack length.

The shape of the \( K_R \)-curve can be determined experimentally from the \( K_G \)-values and the related crack extension during slow stable crack extension.

As said at the beginning of this section, the R-curve can deal both with stable crack extension and limited plastic deformations at the crack tip. To correct the physical crack size for the effects of crack-tip plastic deformation, the fracture toughness is described as a function of a modified crack length: the effective crack length \( a_{eff} \). The \( K_R \) - curve is plotted in terms of effective crack extension \( \Delta a_{eff} \). Two correction methods are proposed by the ASTM standard\(^{13}\): the Irwin correction and the compliance correction.
Irwin correction method

In case of the Irwin method for correction, the effective crack size, $a_e$, is derived from the measured physical crack size, $a_{phys}$, by adding a calculated value of a plastic zone adjustment, $r_y$. For plane-stress and mode I the plastic zone adjustment is given by

$$r_y = \frac{1}{2\pi} \left( \frac{K}{\sigma_y} \right)^2$$

where $K$ is a function of $a_{eff} = a_{phys} + r_y$. Because of the presence of $r_y$ in $a_{eff}$ the determination of $r_y$ and $K$ is an iterative process.

Compliance correction method

The compliance method for correction uses an analytical expression to calculate the elastic compliance of the CCT specimen:

$$\frac{E \cdot COD}{\sigma W} = \frac{2Y}{W} \left[ \frac{\pi a/W}{\sin(\pi a/W)} - \frac{2W}{\pi Y \cosh(\pi Y/W)} \frac{1 + \nu}{\sinh(\pi Y/W)} \right] + \nu$$

where:

- $E$ = Young’s modulus, N/mm$^2$
- $COD$ = crack opening displacement, mm
- $\sigma$ = remote stress, $P/Wt$, N/mm$^2$
- $P$ = load, N
- $t$ = specimen thickness, mm
- $W$ = total specimen width, mm
- $Y$ = half span of gauge to measure COD, mm
- $a$ = half crack length, mm
- $\nu$ = Poisson's ratio.

and is valid for:

$$0.2 < \frac{2a}{W} < 0.8; \quad \frac{Y}{W} \leq 0.5 \quad \text{and} \quad L \geq 1.5W$$
The crack opening displacement, further referred to as COD, is defined as the distance between the upper and lower crack edges at the centre of the crack and is measured with a clip gauge as described in the ASTM E561 standard\textsuperscript{13}. Also the determination of $a_{\text{eff}}$ with the compliance method is an iterative process.

The compliance method uses the increased flexibility of the specimen caused by the presence of a crack. The longer the crack, the more the stiffness reduces and the more the crack opens for a given level of applied load. As long as the stress field is not disturbed by non-linearity’s, the compliance expression is a unique relation between the COD and the effective crack length, $a_{\text{eff}}$. Instead of using the analytical expression, compliance calibration curves can be developed experimentally by elastically loading specimens with different crack sizes and determining the elastic slopes of the various load versus crack opening displacements records, COD/$P$.

### 3.7.3.2 $K_R$ – curve results

Vermeeren\textsuperscript{2}, Bresse\textsuperscript{27}, Sutton\textsuperscript{26} and De Vries \textit{et al.}\textsuperscript{12,14,30} investigated the applicability of the R-curve concept for Glare. They all found that the R-curve could function as a material parameter but within certain limitations of the specimen width and the initial crack length. Small values of $W$ and large values of $2a_0$ should be avoided.

Figure 3-42 shows $K_R$ -curves based on \textit{Irwin’s plasticity correction} for Glare 2 3/2 0.3 and Glare 3 3/2 0.3 determined by De Vries\textsuperscript{14,16}. Without further data investigation, this figure gives the impression that the $K_R$ -curve is indeed a material property since the curves are assembled from specimens with different widths and different initial crack lengths. However, in all these specimens the average stress in the net section at fracture was above the yield stress. As a consequence, none of the presented curves contains the onset of fracture since the iterative process of eq. 3.8 to calculate the effective crack length does no longer converge and is interrupted long before fracture. De Vries and Holleman\textsuperscript{30} showed that the major part of the effective crack length at the onset of NSY consists of the plastic zone correction while the physical crack extension is small. The arrows in Figure 3-42 indicate the points on the curve with a physical crack extension of 1 mm. As a result of the calculation interruption, the presented curves are mainly a function of the gross stress only. Because the gross stress is not in any way related to the material, the $K_R$ -curves must be comparable.
The curves presented in Figure 3-42 are actually not able to predict the residual strength of the specimen geometry with which they were determined. For example, if a designer receives the data of Figure 3-42, he is not able to predict the residual strength of a 800 mm wide Glare 2 3/2 0.3 sheet with $2a/W = 1/3$ since the point where $K_R = K_t$ takes place at larger $a_{eff}$ than represented by these curves. This is due to the fact that a large part of the data points could not be used to calculate $K_R$ because the material was already yielding during these measurements. To do predictions anyhow, the curve fit must be extrapolated beyond the valid data points.

Figure 3-42, $K_R$ curves for Glare 2 3/2 0.3 and Glare 3 3/2 0.3 for different $W$ and $a_0$, based on plastic zone correction of Irwin\textsuperscript{14,16}.

Figure 3-43 and Figure 3-44 contain $K_R$-curves obtained with a plastic zone correction based on the compliance method for Glare 2 and Glare 3 respectively. The curves were obtained in Laboratories at the Universities of Pisa (Italy) and Delft and at McDonald Douglas. As a reference, the $K_R$-curves obtained with the Irwin plastic zone correction for the same data are also included in these charts as lines. The curves obtained with the compliance method, especially for the Glare 3 material, are identical for different $W$ and $a_0$. The curves for the Glare 2 material show somewhat more scatter but tend to become identical as well. The large difference with the compliance curves compared to the Irwin curves is that the compliance curves are based on all available experimental data points while the Irwin curves are only based on the points before NSY occurred.
Available blunt and sharp notch Glare data

Figure 3-43, $K_R$ curves for Glare 2 3/2 0.3 for different $W$ and $a_0$, based on compliance plastic zone correction\textsuperscript{14,16,17}.

Figure 3-44, $K_R$ curves for Glare 3 3/2 0.3 for different $W$ and $a_0$, based on compliance plastic zone correction\textsuperscript{14,16,17}.

For both laminates, the initial part of the $K_R$ curve obtained with compliance correction lies above the curve obtained with Irwin correction. This is due to the fact that the Irwin method calculates a plastic zone and thus an effective crack extension from the moment that the applied load is larger than zero.
In contrast, effective crack extension according to the compliance method starts to increase when physical crack extension takes place. It is important to notice that only the $K_R$ - curve shapes are identical for the different methods. For a certain $a_{phys}$, $a_{eff}$ is significantly different obtained with the compliance- or Irwin plastic zone correction. Consequently, the critical $K_R$-value differs significantly, depending on the method used. This is not the case for aluminium alloys as illustrated by De Vries and Holleman.

The Glare 2 curves in Figure 3-43 need some extra discussion. The curves for the 800-mm wide specimen obtained with the Irwin correction and the Douglas curve based on the compliance correction differ from the other curves. A reason why the curve for the 800 mm wide specimen is below the other curves can be the large amount of buckling; De Vries reported that it was not possible to measure the COD for these panels due to the high out of plane displacement. For the lower Douglas curve, no explanation can be given other than that the material tested originates from the early Glare days (1991) when the material was not yet optimised and for example Glare contained a different fibre-resin combination. Vermeeren mentions these differences in material quality explicitly in his Ph.D. dissertation for Glare 3.

De Vries and Vermeeren investigated the influence of buckling and found that the curves for specimens that are free to buckle are below the curves for specimens for which buckling was restrained with AB - guides. The occurrence of buckling can be observed best from a $K_R$-curve obtained with compliance correction when the effective crack extension suddenly increases more rapidly. This is due to the sudden increase of the COD at the onset of buckling. Since the amount of stable crack extension before NSY can be rather small, a $K_R$-curve obtained with Irwin correction might not reveal buckling at all. An example is given in Figure 3-45.

Vermeeren, De Vries and Pacchione have proposed several curve fits to represent the $K_R$-curves for different laminates that are obtained with the Irwin- and the compliance method. However, most of these curves are based on specimens not wider than 400 mm. To use these curves for residual strength predictions of wider panels or larger relative crack lengths, the curves must be extrapolated and are no longer supported by experimental data. For that reason $K_R$ curves should be based on specimens that are as wide as possible to avoid extrapolation. Schwarmann and Sutton have investigated and reported $K_R$-curves for wider panels that were tested at DERA together with curve fits to present the data. Table 3-1 contains curve fits for several different laminates based on...
the compliance correction and these are plotted in Figure 3-46 as far as they are supported by experimental data.

Figure 3-45, Representation of buckling in the $K_R$ curves for Glare 2 3/2 0.3 for different plasticity corrections ($W=400$ mm, $2a/W=1/414,16$).

Figure 3-46, Curve fits for several different Glare laminates supported with experimental data, see Table 3-1.
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Table 3-1, R-curve fits based on compliance plastic zone correction.

<table>
<thead>
<tr>
<th>Material</th>
<th>Width [mm]</th>
<th>Curve fit, $K_I =$</th>
<th>Source</th>
</tr>
</thead>
<tbody>
<tr>
<td>Alclad 2024-T3 T-L</td>
<td>600</td>
<td>$33.25 - 1.02 \Delta a_e + 12.72 \Delta a_e^{0.641}$</td>
<td>DA/ALCOA/DERA$^{17}$</td>
</tr>
<tr>
<td>Alclad 2024-T3 T-L (1.6mm)</td>
<td>700-1923</td>
<td>$21.49 - 0.51 \Delta a_e + 18.08 \Delta a_e^{0.520}$</td>
<td>DERA (1989)$^{17,18}$</td>
</tr>
<tr>
<td>Alclad 2024-T3 L-T (1.6mm)</td>
<td>700-1948</td>
<td>$27.53 - 1.06 \Delta a_e + 16.12 \Delta a_e^{0.612}$</td>
<td>DERA (1989)$^{18}$</td>
</tr>
<tr>
<td>Glare 2A 3/2 0.3 L-T</td>
<td>600</td>
<td>$45.52 - 2.57 \Delta a_e + 16.14 \Delta a_e^{0.719}$</td>
<td>Douglas (1991)$^{17,18}$</td>
</tr>
<tr>
<td>Glare 2A 3/2 0.3 L-T</td>
<td>400</td>
<td>$1.74 \Delta a_e + 58 \Delta a_e^{0.2}$</td>
<td>TU-Delft$^{2,14}$</td>
</tr>
<tr>
<td>Glare 3 2/1 0.3 L-T</td>
<td>&gt;1200</td>
<td>$29.99 - 0.19 \Delta a_e + 18.37 \Delta a_e^{0.449}$</td>
<td>DERA (1995)$^{33}$</td>
</tr>
<tr>
<td>Glare 3 3/2 0.3 L-T</td>
<td>400-800</td>
<td>$34.51 - 59.29 \Delta a_e + 67.59 \Delta a_e^{0.976}$</td>
<td>DA/Douglas/SLC$^{17,18}$</td>
</tr>
<tr>
<td>Glare 3 3/2 0.3 L-T</td>
<td>&gt;1200</td>
<td>$29.94 - 0.67 \Delta a_e + 18.85 \Delta a_e^{0.546}$</td>
<td>DERA (1995)$^{33}$</td>
</tr>
<tr>
<td>Glare 4A 2/1 0.3 L-T</td>
<td>165-1219</td>
<td>$31.48 - 1.47 \Delta a_e + 15.83 \Delta a_e^{0.656}$</td>
<td>DERA (1995)$^{33}$</td>
</tr>
<tr>
<td>Glare 4A 3/2 0.3 L-T</td>
<td>600</td>
<td>$27.72 - 0.82 \Delta a_e + 16.39 \Delta a_e^{0.601}$</td>
<td>Douglas (1991)$^{15,16}$</td>
</tr>
</tbody>
</table>

It is clear from Figure 3-46 that the residual strength of Glare 2A in fibre direction is superior to the other laminates, followed by Glare 4A. The difference between Glare 4A 2/1 0.3 and Glare 4A 3/2 0.3 is small. For Glare 3, a larger MVF (2/1 lay up versus 3/2 lay-up) seems to result in a lower residual strength for that part of the curve that is covered with experimental data. At the right side of the figure, the curve fit for the 2/1 lay-up tends to go above the curve for the 3/2 lay-up.

Again, the difference between the McDonald Douglas curve fit for Glare 2A and the curve fit obtained at the Delft University show a remarkable difference. The Douglas curve seems to be too low for the laminates produced at a later stage. The Glare 2 curve of obtained at the Delft University becomes a more or less a straight line for large effective crack extensions. This is expected to be due to edge effects when the crack tip gets too close to the specimen boundary.
3.7.3.3 Residual strength predictions based on \( K_R \)-curve approach

The curve fits obtained with the compliance method presented in the previous paragraph are used in this paragraph to predict the residual strength of sheets with different widths and starter crack sizes. For prediction purposes, curve fits provide valid results as long as measured data points support them. Beyond these points, the curve fit needs to be extrapolated, which creates uncertainty. This is illustrated in Figure 3-47 for the prediction of the residual strength of an 800 mm wide specimen.

Figure 3-47, Illustration of the influence of the \( W \) on the residual strength.

Figure 3-48, Residual strength predictions compared with experimental data for 400 wide specimens of Glare 2A and Glare 3.
To do predictions, a crack driving force is calculated for various specimen widths and initial crack lengths, following the crack driving force analysis procedure that is applicable to metals. The failure load is determined graphically as the point where tangency of the crack driving force and crack resistance curve occurs. This has been explained in § 3.7.3 and is illustrated in Figure 3-47 showing the influence of the width on the residual strength predictions. A width of 200 mm results in a residual strength of 345 MPa while an increase of the width to 800 mm results in a residual strength of 320 MPa.

![Figure 3-49, Experimental residual strength data Glare 3 3/2 0.3 L-T compared with two curve fits.](image)

Based on the curve fits given in Table 3-1, residual strength predictions are made for several different initial crack lengths for 400 mm wide Glare 2A and Glare 3 specimens. Predictions for wider specimens could not be done for Glare 2A since the curve fit was not valid up to these wide specimens and no proper experimental data is available for wider Glare 3 and Glare 4A specimens. The predictions for different curve fits and the experimental results are given in Figure 3-48. This figure also contains a bandwidth around the predictions that represents an absolute difference of 5% from the predictions. Thus, if the experimental results fall within these boundaries, the predictions are accurate within 5%. It is clear that the Glare 2 curve fit of De Vries predicts the residual strength values quite accurately. The curve fit of Douglas is clearly too low. Also the curve fit of DERA for Glare 3 laminates predicts the residual strength rather well, although the results for smaller relative crack lengths are less accurate. This can be
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expected since this curve fit initially lies above the experimental data as illustrated in Figure 3-49.

3.8 Glare laminates versus current aluminium alloys

3.8.1 $K_R$-curves for aluminium alloys

The aluminium alloys currently used in fuselage structures are mainly Alclad 2024-T3 and Alclad 2524-T351 as a new alloy in the Boeing 777. Figure 3-50 shows the $K_R$-curve fits of both materials in L-T and T-L direction. All data are obtained on 1.6 mm thick sheets. The fracture toughness of the new alloy 2524 is significantly improved compared to 2024. Since all other material properties of both aluminium alloys are comparable, it is clear why 2524 was chosen as the fuselage material for the Boeing 777. Another remarkable phenomenon is the considerable difference between the residual strength in L-T direction and T-L direction; the residual strength in the rolling direction is superior.

Figure 3-50, $K_R$-curves obtained with compliance correction for different aluminium alloys (for 2024-T3 curve fits see Table 3-1)\(^{34}\).

In the following paragraph, the $K_R$-curves of Glare will be compared with these aluminium curves.
3.8.2 $K_R$-curves for Glare compared to aluminium alloys

Within Figure 3-51, the $K_R$-curve fits of Glare 2A and Glare 4A in L-T direction are compared with the curves of the aluminium alloys Alclad 2024-T3 and Alclad 2524-T351 in L-T direction. Figure 3-52 compares the $K_R$-curves of Glare 3 in the L-T direction with the same aluminium alloys.

Figure 3-51, $K_R$-curves obtained with compliance correction for different Glare laminates and aluminium alloys in L-T direction (see Table 3-1 and ref. [33]).

Figure 3-52, $K_R$-curves obtained with compliance correction for different Glare 3 laminates and aluminium alloys in L-T direction (see Table 3-1 and ref. [33]).
The residual strength of Glare 2 in L-T direction is superior. The $K_R$-curve of Glare 4A is above Alclad 2024-T3 but below Alclad 2524-T351. The $K_R$-curves for the Glare 3 laminates are close to the curve for Alclad 2024-T3. However, as discussed before, the curve fit for the Glare 3 3/2 0.3 laminate is somewhat too high to represent the experimental data and the residual strength results of the Glare 3 laminates might actually be below those of Alclad 2024-T3.

### 3.8.3 Comparison based on specific weight

An important reason to use Glare instead of aluminium is the lower specific weight of the fibre prepreg compared to monolithic aluminium. Therefore, to compare Glare with other materials, it makes sense to divide the maximum allowable stress by the specific weight. Table 3-2 gives an overview of the calculated specific weights of some Glare variants that are discussed in this report. Glare 2 and Glare 3 both have two prepreg layers between adjacent aluminium sheets. Therefore, their specific weight is equal when the lay-up and the thickness of the applied aluminium layers is equal. For the same reason, the use of a Glare A or Glare B-variant has no influence on the specific weight.

<table>
<thead>
<tr>
<th>Material lay-up</th>
<th>Thickness Al layers [mm]</th>
<th>Specific weight [kg/m³]</th>
<th>Indexed specific weight</th>
<th>MVF [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>2024-T3 or 2524-T3</td>
<td>- -</td>
<td>2770</td>
<td>100</td>
<td>100</td>
</tr>
<tr>
<td>S2 prepreg - UD</td>
<td>- -</td>
<td>2000</td>
<td>0.72</td>
<td>0%</td>
</tr>
<tr>
<td>Glare 2 or Glare 3</td>
<td>2/1 0.2</td>
<td>2474</td>
<td>0.89</td>
<td>61.5</td>
</tr>
<tr>
<td></td>
<td>2/1 0.3</td>
<td>2544</td>
<td>0.92</td>
<td>70.6</td>
</tr>
<tr>
<td></td>
<td>3/2 0.2</td>
<td>2420</td>
<td>0.87</td>
<td>54.5</td>
</tr>
<tr>
<td></td>
<td>3/2 0.3</td>
<td>2405</td>
<td>0.90</td>
<td>64.3</td>
</tr>
<tr>
<td>Glare 4</td>
<td>2/1 0.2</td>
<td>2397</td>
<td>0.87</td>
<td>51.6</td>
</tr>
<tr>
<td></td>
<td>2/1 0.3</td>
<td>2474</td>
<td>0.89</td>
<td>61.5</td>
</tr>
<tr>
<td></td>
<td>3/2 0.2</td>
<td>2342</td>
<td>0.85</td>
<td>44.4</td>
</tr>
<tr>
<td></td>
<td>3/2 0.3</td>
<td>2420</td>
<td>0.87</td>
<td>54.5</td>
</tr>
</tbody>
</table>

The specific $K_R$-curves based on these specific weights and compared with the specific weight of 2024-T3 are plotted in Figure 3-53. The superiority of the Glare 2A laminate, tested in L-T direction remains the same. However, the $K_R$-curves for the Glare 4A laminates are now comparable with the one for Alclad 2524-T351 and the Glare 3 curves are significantly above the curves for Alclad 2024-T3.
Figure 3-53, Indexed K_{IP} curves obtained with compliance correction for different Glare laminates and aluminium alloys in L-T direction.

3.9 Summary and conclusions

In contrast to aluminium alloys, the presence of notches in Glare causes significant strength reductions while the shape of the notch is of less importance. Notches that are identified as blunt for aluminium alloys (K_{t}<4) seem to behave like sharp notches in case of FML’s.

This paragraph gives a summary of the items discussed in this chapter. A distinction is made between round notches or holes and sharp cracks.

The static behaviour of FML’s with holes

§ 3.3 Blunt notch experimental results

- The presence of a hole in a Glare laminate gives a net strength reduction between 40 and 50%.
- In general, a lower MVF results in a higher blunt notch strength. The blunt notch results for Glare 3 laminates with only two aluminium layers do not confirm this trend. However, these specimens are expected to be loaded by an extra internal bending moment due to the asymmetric lay-up.
- K_{t} is not related to the blunt notch strength of Glare laminates.
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- The net blunt notch strength increases with
  - decreasing hole diameter and constant width or constant \( d/W \) ratio.
  - decreasing width and constant \( d \).
- The net blunt notch strength of multiple holes is constant if the ratio between the hole diameter and the hole pitch is kept constant.

§ 3.4 Failure mechanism and criteria
- Indications of delamination and local fibre failure at the notch root were obtained before final blunt notch failure.
- All specimens showed fully plastic net sections in the aluminium layers before failure.

The static behaviour of FML’s with cracks

§ 3.5 Sharp notch behaviour
- For Glare laminates a distinction must be made between damage caused by fatigue where only the metal layers are cracked and through-cracks where both the metal and fibres are cracked. The residual strength of Glare laminates with fatigue cracks and fibres in loading direction is superior compared to aluminium alloys.
- Glare with through-cracks exhibits a capability for slow stable crack growth prior to rapid failure.
- The amount of available residual strength data for Glare is rather limited and mainly for Glare 2A and Glare 3 with three 0.3 mm thick aluminium layers and tested in the rolling direction of the aluminium layers.

§ 3.6 Residual strength results of Glare laminates
- Residual strength investigations on thin Glare laminates with Compact Tension specimen are problematic. Therefore, the majority of the residual strength data is obtained with Centre Cracked Tension specimens.
- The influence of the aluminium rolling direction on the residual strength of a Glare laminate is not investigated thoroughly. Preliminary results give an indication that the residual strength of a Glare laminate tested in rolling direction is higher than the residual strength perpendicular to the rolling direction.
- Initial tests on 100 mm wide specimens show an increase of the residual strength as a function of the pre-stretching after curing.
Chapter 3

- The residual strength of a laminate increases for lower temperatures.
- Cross-ply CCT specimens loaded under different angles with the fibre orientation show a minimum residual strength at 45°. At this angle, the shear stresses are maximal.
- The residual strength of UD CCT specimens, loaded under different angles with the fibre orientation, reduces significantly from 0° to 30°. Between 45° and 90° the residual strength decreases only slightly.
- Buckling of the crack edges results in a significant residual strength reduction of approximately 10%.
- The residual strength – which is superior for a laminate with fatigue cracks - decreases significantly when the bridging fibres are cut and the ratio between the crack in the fibre layers and the crack in the aluminium layers decreases to a factor 0.5 or less.
- No differences in residual strength are found for a through crack made with a saw and a through crack induced by an impact of a sharp object.

§ 3.7 Available fracture mechanics concepts for Glare and their applicability

- Since the critical stress intensity factor depends on the specimen geometry, i.e., width, it is not useful as a Glare material parameter to describe the residual strength.
- The $K_{IR}$-curves obtained with an Irwin plastic zone correction that are found in the literature are independent of the specimen geometry. However, the amount of stable crack extension in these curves because of NSY is limited due to the small specimen widths applied. As a consequence, the presented curves are mainly a function of the gross stress only.
- The Glare 3 $K_{IR}$-curves obtained with a compliance plastic zone correction are a material constant. After some effective crack extension, these curves become identical with the curves obtained with the Irwin plastic zone correction. However, a comparable physical crack extension gives a different effective crack extension for both methods.
- The $K_{IR}$-curves obtained with a compliance plastic zone correction for Glare 2 laminates show more scatter than for Glare 3 but can be identified as a material constant also. Also for Glare 2, these curves are initially above the curves obtained with an Irwin plastic zone correction.
Available blunt and sharp notch Glare data

- The influence of buckling is very well reflected in the $K_R$-curves based on a compliance method. The $K_R$-curves based on the Irwin plastic zone correction show hardly any difference due to the limited amount of physical crack extension that is incorporated in these curves.
- The residual strength of Glare 2A L-T is the highest, followed by Glare 4A L-T and Glare 3 L-T respectively.
- It is possible to predict the residual strength of 400 mm wide panels with the available $K_R$-curve fits. The amount of data found in the literature is not sufficient to validate this prediction method for larger widths.

§ 3.8 Glare laminates versus current aluminium alloys
- A comparison of the $K_R$-curves of different Glare laminates and aluminium alloys gives the following ranking:
  1. Glare 2A 3/2 0.3 L-T
  2. Alclad 2524-T351 L-T
  3. Glare 4A 2/1 0.3 L-T ≈ Glare 4A 3/2 0.3 L-T
  4. Glare 3 3/2 0.3 L-T = Alclad 2024-T3 L-T
  5. Glare 3 2/1 0.3 L-T

- Based on the specific weight, the previously given ranking changes into:
  1. Glare 2A 3/2 0.3 L-T
  2. Alclad 2524-T351 L-T = Glare 4A 2/1 0.3 L-T = Glare 4A 3/2 0.3 L-T
  3. Glare 3 3/2 0.3 L-T
  4. Glare 3 2/1 0.3 L-T
  5. Alclad 2024-T3 L-T
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Part 2:
Experimental program and development design tools
4 DEVELOPMENT OF AN IMAGING SYSTEM

4.1 Introduction

To investigate the blunt and sharp notch behaviour of Glare, there was the need within this study to obtain an imaging system that is able to

- observe the fracture mechanism in the interesting areas, and
- carry out accurate measurements like
  - crack extensions,
  - crack angles,
  - local strain fields and
  - delamination extensions

This chapter describes the development of an imaging system used during this research. § 4.2 describes the hardware that was purchased and the relation between the different components. Initially, the measurements and the analysis of the data were done non-simultaneously with different software packages. Since this was rather time consuming and inefficient it was decided to develop an in-house software package called Impress (Image processing residual strength). A general overview and a description of some of its possibilities are given in § 4.3.

A large part of the development of the system focussed on the measurement of localised strain fields. This included the development of a marking technique that could be easily applied to the surface of the material and at the location of interest. § 4.4 discusses briefly the available ways to measure local strains. This paragraph also describes the development of the strain measuring technique with the developed system. § 4.5 evaluates the method while § 4.6 describes the accuracy of the system.

4.2 Description imaging system

The imaging system basically consists of a movable XYZ-table, a computer and a CCD camera. see Figure 4-1. Furthermore some in-house developed as well as purchased software packages are used to analyse and edit the acquired data.

This paragraph will discuss the main components of the system.
Figure 4-1, Imaging system set-up.
Development of an imaging system

**CCD-camera + lens**
The most important item of the imaging system is the camera. A lightweight black and white CCD camera grabs the visual image. The CCD camera contains a 0.5-inch chip with two serial pixel matrices of 780 x 576 pixels in horizontal and vertical direction respectively. This is illustrated in Figure 4-2. The pixel matrix closest to the image is light sensitive (image matrix) while the other acts as frame storage (storage matrix). The image is projected on the image matrix and all separate sensors or pixels measure the light intensity of a part of the image. After 20 ms of integration, the charge is shifted out of the image matrix into the storage matrix. During the vertical blanking from the storage matrix, the lines are shifted out one by one in the following 20ms and are transferred to the Camera Control Unit (CCU). Thus, the process to capture an image and release the data to the CCU takes 20 microseconds.

![Figure 4-2, CCD camera.](image)

The image is projected on 780 x 576 pixels. The smallest translation that the camera can detect is when the image moves one sensor or pixel on the CCD chip. This minimum translation is equal to the size of the image that is projected on one pixel or sensor. In the remainder of this chapter, this size is called Image-Pixel-Size (IPS). The IPS is not the same as the pixel size on the CCD chip. The ratio between the pixel size and the IPS is the magnification factor of the lens.
With dedicated software, based on sub-pixel algorithms, it is possible to detect smaller translations. This is possible when the moving object is represented by more than one pixel or when the grey level of each pixel is taken into account.

The system is purchased to measure strain fluctuations of 2500 micro strain based on an initial length of 1 mm and absolute IPS. As a result, the maximum IPS must be equal or less than:

\[
2500 \text{ micro strain} \times 1 \times 10^{-6} (= 1 \text{ micro strain}) \times 1 \text{ mm} = 2.5 \mu m
\]

and the lens that is purchased must be able to distinct a minimum of 400 black and white lines per mm.

In reality, the IPS was found to be 2.8\( \mu m \)\(^{3,4} \) while the frame grabber could only pass 735 x 570 pixels. Therefore, the image size that can be captured is 2.1 x 1.6 mm.

The camera is connected to a CCU that controls the video output and the black level of the captured images. It is possible to connect the CCU with a VCR and tape the images. Another output of the CCU goes to the frame grabber.

Frame grabber
The frame grabber is a selection of computer boards that allows processing of the digitised images by a computer. It consists of:

- A basic graphics board that can be extended with several optional modules. This basic board contains a programmable powerful graphical processor that takes over the full display control from the computer. This was necessary because at the time of purchasing, ISA computer boards were the standard and their bus connection was not fast enough to give the host processor on line control over the incoming images. One output of this frame grabber board is connected to a second monitor that shows the captured image real time.
- On top of the basic board, a grey digitiser module is mounted for capturing the black-and white images.
- Additionally a dedicated processing board (DSP) is added for time-consuming and on-board calculations.
**XYZ-table**

An XYZ-table supports and positions the camera with an absolute accuracy of 4 \( \mu \text{m} \). The table has a movable range of 170 mm in the two orthogonal directions parallel to the specimen (x and y directions) and 50 mm in the direction perpendicular to the specimen or z direction. Step motors that are controlled by a control unit (F9S) are responsible for the movements. This control unit is connected to the RS232 connector of a computer that gives the commands for the step motors. The F9S is also controllable by a joystick for manual displacements.

The XYZ table is mounted on a large movable device for rough positioning. This device can translate the XYZ table in three directions and rotate around the x- and y- axes.

### 4.3 Software package Impress

To carry out measurements and analysis with the system described in § 4.2, it is important to combine information of the camera position with the captured image. It is preferable to carry out these measurements on-line. This is not only illustrative and allows the user to take corrective actions, it also reduces the amount of storage and reduces the amount of operator time.

A clear example is the measurement of a crack length that covers more than one image. With the available control software that came with the system and the purchased image analysis package “Image Pro Plus (IPP)”, the procedure is the following:

1. Images are captured of the region between the initial crack tip and the extended crack tip. All these images have an overlap at one or more of their boundaries to prevent loss of data. The images with an individual size of 735 x 570 \( \approx 420 \text{ kB} \) are stored on hard disk.
2. After the test, all images are read into IPP and have to be merged semi automatically since the overlap area must be determined. This is a time consuming process and takes at least 5 minutes per merge for a skilled operator.
3. A particular length in the image is identified for the use of calibration.
4. The crack length measurement is carried out.

Figure 4-3 shows a crack that is assembled of three images. The cuts or merge lines are indicated with arrows. To measure this specific crack length, 1.2 MB of data had to be stored and it took almost half an hour for
the operator. The necessary information is no more than the co-ordinates of
the crack tip origin and the extended crack tip. A special “merge” function
within the software was tried but worked only under special circumstances
and was very slow. Besides that, the software could only merge in one
direction, thus only strings of one image high could be made.

A solution for this problem could be to use the high accuracy movements of
the XYZ table. However, the testing environment normally contains a lot of
vibrations during the measurements. Therefore, not one image should be
snapped but for example ten images. Adding these and averaging them will
reduce the vibration disturbance and allow the use of the XYZ-table for
positioning. At the time of purchasing, the high speed PCI computer bus
was not yet available for the current generation frame grabbers. Therefore,
averaging is very time consuming when this is done by the CPU of the host
computer because of the large amount of data transport from the frame
grabber to the computer and vice versa. Thus, a CPU or graphics processor
on the frame grabber should do the averaging.

This graphics processor can also be used to enhance the digital images
with functions like sharpening, etc. For the strain measurements that will be
discussed later, this processor can detect bright or dark objects within an
image, order and label them and calculate their centre points. If only
detection of the centre points is enough for measuring deformations, also
the problem of the data storage is solved. In an extreme case of a residual
strength test, a field of 100 images in front of the crack tip should be
analysed. Taken into account that this is done for a maximum of 200 load
steps and that the image size is equal to 420 kB, the storage size of one
residual strength test is close to 7 Gigabytes!

To overcome all these problems, the software package Impress was
developed. This chapter discusses briefly this software package Impress
that is able to control the XYZ table, to capture images and to perform on-line measurements.

4.3.1 General

Impress is an Object Oriented program, made with the software package Microsoft Visual C++. It contains two main sub programs:

- One part of the program controls the communication with the F9s to control the movement of the XYZ table. This communication is made multi-tasking to prevent the computer to wait with other actions until the F9s finished a movement.
- Another part controls the frame grabber to capture images and to carry out on-board calculations. This includes filter functions for images but also the control of a mouse cross on the second monitor.

Figure 4-4 shows the user interface of the program. The program contains three main boxes with functions:

- **Camera control box:** to control the frame grabber and the DSP
- **Stage control:** to control the XYZ table
- **Scan box:** to create scan areas and control scanning

![Figure 4-4, User interface Impress.](image)
Additionally, the toolbar contains several measurement functions and functions to control the second monitor.

### 4.3.2 Auto focus function

To automate the focussing of the images, Impress contains an auto-focus function. This function uses the fact that the differences in pixel values for adjacent pixels are maximal for an image that is in focus.

The auto focus routine is made such that it can be applied to the whole image but also to a confined area within the image. The focus area can be indicated with a window that is displayed over the captured images screened on the second monitor. This auto focus window can be moved and sized with a mouse pointer that is connected to the host computer.

### 4.3.3 Merge / scan function

Because the size of the single captured images is limited, Impress has the capability to create an assembled image from several single images. This assembled image consists of a selected amount of images in horizontal and vertical direction. The appropriate control box is illustrated in Figure 4-5.

![Figure 4-5, Scan function.](image-url)
This function uses the calibrated size of the captured image, together with
the positioning information of the XYZ table. The advantage of this function
is that the accuracy of operations or measurements for one image can be
extended to a larger image without losing any of this accuracy. The
scanning function makes use of averaging multiple images.

Figure 4-6 shows a schematised area build up from 9 scans or single
images; 3 in horizontal and 3 in vertical direction. The scanning area is
created at different loads. With this method it is possible to carry out
measurements that are no longer restricted to one scan. It is also possible
to set the scan origin and direction and to define measurements or
operations for every scan individually. For example before capturing but
after moving to the final coordinates for scan 01 and 03, the auto focus
function is carried out, while in scan 02 a histogram for the light intensity is
made and the video output is corrected. It is possible to compare the
measurements for load $i$ with the measurements of load $k$.

4.3.4 Measurement functions

The measurement functions incorporated in the program focussed on the
analysis of residual strength tests. That means that the system is able to
measure lengths, for example to measure crack extensions, and can
measure angles for CTOA determination. The measurements are actually
carried out with the use of the second monitor that displays the image. With
the use of a mouse cross that is projected on the second monitor, the start
and end points of a length or the three points that represent an angle can
be selected. The length to be measured is not restricted to an image size. For the measurement of CTOA’s, a circle with a radius of 1 mm is projected around the selected centre point. Additional graphic tools are available like the projection of a grid of vertical and horizontal lines over the image with an adjustable internal distance.

For the investigation of blunt and sharp notch behaviour, there was the need for a system that is able to measure strain fields in an area with high strain concentrations. It has been investigated whether the imaging system could be used for that. This is discussed in the next paragraph.

### 4.4 Strain measurements based on fine grid method

Strain fields can be presented in several qualitative and quantitative ways. This chapter does not describe or summarise them again but focuses on the implementation of the “Fine Grid method” as strain measuring technique for the digital imaging system.

<table>
<thead>
<tr>
<th>Obj#</th>
<th>Area</th>
<th>Centr-X</th>
<th>Centr-Y</th>
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</thead>
<tbody>
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<td>406</td>
<td>22.01</td>
<td>9.25</td>
</tr>
<tr>
<td>2</td>
<td>375</td>
<td>58.81</td>
<td>10.18</td>
</tr>
<tr>
<td>3</td>
<td>353</td>
<td>94.10</td>
<td>11.35</td>
</tr>
<tr>
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<td>129.03</td>
<td>11.91</td>
</tr>
<tr>
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<td>164.92</td>
<td>13.21</td>
</tr>
<tr>
<td>7</td>
<td>363</td>
<td>200.85</td>
<td>15.05</td>
</tr>
<tr>
<td>12</td>
<td>393</td>
<td>20.91</td>
<td>44.28</td>
</tr>
<tr>
<td>13</td>
<td>356</td>
<td>57.140</td>
<td>45.91</td>
</tr>
</tbody>
</table>

Figure 4-7, Illustration of marker (position) identification.
In the fine grid method, a grid is applied on the surface of the specimen at a position where strain information is needed, see Figure 4-7 a). During loading, this grid follows the deformations of the carrier material. In this way the relative translation of the markers that are surrounded by lines (grid) can represent the local deformations and strain field. A digital camera is used to capture this grid or scan the marker positions for different loading conditions, see Figure 4-7 b), c) and d). From the differences in the distances between the grid points, strains can be determined as illustrated in Figure 4-8.

**Figure 4-8, Strain determination with grid.**

An important advantage of the fine grid method over strain gauges is the ability to get hold of strains in entire areas instead of in discrete points and directions only. Furthermore, a fine grid method has the ability to detect strain gradients more accurately than strain gauges since the strain obtained with a strain gauge is an averaged value over the area covered by the gauge. Geers\(^6\) showed that the method provides good results for uniform stress fields.

This method asks for the application of a pattern of markers or a grid on the surface of the test specimen. This grid is acquired by a camera and digitised before and during loading of the specimen. From this data the grid points are determined by calculating the centre of each area between the gridlines.
This paragraph describes the development of an applicable strain measurement method based on the fine grid method. This includes the development of a satisfying grid.

### 4.4.1 Application of markers on specimen

A possible solution to have discrete points on the material surface is to equip the surface with marker points. Different processes can do this. A distinction is made between two processes.

1. *Processes that damage the material.* Examples are milling or carving and laser burning.
2. *Processes that do not damage the material surface* but add material to the surface. Examples are painting, stamping or for example a photographic process.

The requirements that the grid has to met are:

- the grid material has to behave exactly the same as the test specimen,
- the influence of the grid material on the behaviour of the specimen must be negligible, and
- it must be easy applicable around notches.

In this study, several different methods have been investigated like applying and etching photo layers on the surface, laser burning the material and cutting the material⁷. However, none of these methods resulted in markers that could be distinguished well enough for the use of the fine grid method, except for a specially, at the Structures and Materials Laboratory, designed grid that was bonded to the material surface⁸. To create this grid, a line Moiré pattern is used. The distance between two parallel lines in this grid is 100µm while the thickness of the lines is 50 µm. To build the grid, a Moiré line pattern is contact-printed twice on a photo paper, first in one direction and subsequently in the direction perpendicular to the first direction. The process is illustrated in Figure 4-9. The final step is to develop the paper.

The grid is an extremely thin photographic paper of 6 µm, on which regular straight lines were printed. The thin and sensitive photo paper is supported at one side with a thick plastic foil that protects the grid while the other side is glued to a specimen. An example is given in Figure 4-10. The protection is removed from the specimen before the actual measurements.
Based on the small thickness of the photographic layer and its low modulus of elasticity, influences on the mechanical properties of the test specimen are expected to be negligible.

Figure 4-9, Development process strippable grid.

Figure 4-10, Strippable photo grid, applied to specimen with blunt notch (left side).
With this grid distance about $15 \times 20 = 300$ grid points are captured per scan, the camera scans an area of approximately $2.1 \times 1.5$ mm.

### 4.4.2 Marker detection

Marker detection is done with the help of a histogram of the image. Because the markers show up light-grey while the surrounding lines show up mainly black, a histogram shows a clear distinction between the pixel values of points and lines. A threshold for the pixel values ($pv$) is determined that lies between the pixel values of the markers and the lines. Pixels in the original image with pixel values below the threshold (the lines) receive a new pixel value of 0 (black) while all other pixels get the value 255 (white). In this way the original image is transformed into an image with only black and white.

The white points can easily be separated from the black background. Their location is calculated using the following equations:

$$
\sum_{i=1}^{n} x_i \quad \text{and} \quad \sum_{i=1}^{n} y_i
$$

where $n$ is the amount of pixels of the marker and $x_i$ and $y_i$ are the co-ordinates of pixel $i$. With these equations the centre of a marker point is calculated. These coordinates are not determined by a specific pixel location and are, therefore, more accurate.

The centre points of the areas enclosed by the grid lines define the discrete grid points. The size of each marker area is approximately $50 \, \mu m \times 50 \, \mu m$. With an approximate pixel size of $2 \, \mu m \times 2 \, \mu m$, this results in a field of approximately 25 lines and 25 columns of pixels. To represent the image in black and white values for the grid points and the grid lines respectively, a threshold pixel value of 127 is used to mark the boundary. To determine the mean y-coordinate of the grid-points, the intermediate y-values of all columns are added and divided by the number of columns. The same procedure is carried out on the lines to determine the mean x co-ordinate of each grid point.

At the stops, the camera scans the grid and the co-ordinates are recorded and saved to a computer file by Impress. The scanning of the points is illustrated in Figure 4-11.
The output from the digital image processor is a text file with grid point centre co-ordinate (indicated as grid points from here on). A typical plot of the grid points before and during loading is given in Figure 4-11.

**Figure 4-11, Representation of measured centre points in Impress.**

### 4.4.3 Marker identification for different loads

The raw data file exported by Impress is a ‘*.txt’ file with a list of numbers in columns. The first column contains an intensity value, representing the average brightness of the grid point. A pixel value of 225=white, 0 is black. The other columns contain the x, y, z, coordinates and the grid point size.

With the co-ordinates of the grid points for several stress levels, strain fields can be visualized, for example with the software program of Geers\(^6\). However, for processing the grid data into strains, all samples have to have an equal number of grid points and the points must be arranged in the same order.

To simplify the deformation calculation, the load specific markers are stored in matrices where the horizontal lines define the rows and the vertical lines the columns. To obtain a large scanning area together with a high accuracy, the area and matrix for one load step can be assembled from several partial
scans. Finally, all the load specific marker patterns are assembled in a matrix, see Figure 4-12.

Figure 4-12, Sorting of markers in columns and rows.

Impress contains sorting routines but these still have to be optimised. Figure 4-13 illustrates the sorting of markers in lines and columns for a very regular scan. Some of the problems that arise during this process are:

1. Moving of markers in and out of the scanning area during loading
2. Non orthogonality of the grid,
3. Disappearing or multiplication of points due to the fracture process,

Figure 4-13, Sorting routine of Impress.
4.4.4 *Strain determination*

A straightforward way to derive the three strain components from a deformed grid is to divide the grid in arbitrary triangle elements. For a single element, with global node-numbers p, q and r, the displacements of the nodes are depicted in Figure 4-14.

![Figure 4-14, Grid point displacements.](image)

If a linear displacement between the nodes is assumed, then the displacement components of an arbitrary point i is described by:

\[
\begin{bmatrix}
u_i \\
v' \\
\end{bmatrix} = \begin{bmatrix}
1 & x & y & 0 & 0 & 0 \\
0 & 0 & 0 & 1 & x & y
\end{bmatrix} \begin{bmatrix}
\alpha_i \\
\beta_i \\
\end{bmatrix}
\]

**eq. 4-2**

The six coefficients \(\alpha_i\) and \(\beta_i\) (\(i = 1,2,3\)) are derived from the six displacement components of the nodes. This relationship is:
Chapter 4

\[
\begin{bmatrix}
\alpha_1 \\
\alpha_2 \\
\alpha_3 \\
\beta_1 \\
\beta_2 \\
\beta_3
\end{bmatrix} = \begin{bmatrix}
B & 0 \\
\vdots & \vdots \\
0 & B
\end{bmatrix}
\begin{bmatrix}
u^p \\
u^q \\
v^p \\
v^q
\end{bmatrix}
\]

with

\[
B = \frac{1}{2A} \begin{bmatrix}
x^2y^3 - x'y^2y' & x'y^3 - x'y^3 & x'y^3 - x'y^3 \\
y^3 - y' & y' - y' & y' - y' \\
x' - x' & x' - x' & x' - x'
\end{bmatrix}
\]

and A is the area of the triangular element:

\[
A = \frac{1}{2}(x^2y^3 + x'y^3 + x'y^3 - x'y^3 - x'y^3 - x'y^3)
\]

With the assumption that only small deformations occur, the strains can be calculated from the deformations with the following formula:

\[
\begin{bmatrix}
\varepsilon_x \\
\varepsilon_y \\
\psi_{xy}
\end{bmatrix} = \begin{bmatrix}
\delta \\
\frac{\delta}{\delta x} & 0 \\
0 & \frac{\delta}{\delta y}
\end{bmatrix}
\begin{bmatrix}
u \\
v
\end{bmatrix}
\]

Adding equation 4.2, performing the matrix-multiplication and differentiation and adding equation 4.3 results in:

\[
\begin{bmatrix}
\varepsilon_x \\
\varepsilon_y \\
\psi_{xy}
\end{bmatrix} = \begin{bmatrix}
0 & 1 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 1 \\
0 & 0 & 1 & 0 & 0
\end{bmatrix}
\begin{bmatrix}
\alpha_1 \\
\alpha_2 \\
\alpha_3 \\
\beta_1 \\
\beta_2 \\
\beta_3
\end{bmatrix}
\]

Insertion of eq. 4.3 in eq. 4.5 results in the desired relation between the deformation and the strains:
Development of an imaging system

\[
\begin{bmatrix}
\varepsilon_x \\
\varepsilon_y \\
\psi_{xy}
\end{bmatrix} = \frac{1}{2A} \begin{bmatrix}
b_1 & b_2 & b_3 & 0 & 0 & 0 \\
0 & 0 & 0 & a_1 & a_2 & a_3 \\
a_1 & a_2 & a_3 & b_1 & b_2 & b_3
\end{bmatrix} \begin{bmatrix}
u^b_p \\
u^b_q \\
u^b_r \\
v^b_p \\
v^b_q \\
v^b_r
\end{bmatrix}
\]
eq 4-6

where \( a_i \) and \( b_i \) (\( i = 1,2,3 \)) are the component separations of the node coordinates, defined as:

\[
\begin{align*}
    a_1 &= x^r - x^q \\
    a_2 &= x^p - x^r \\
    a_3 &= x^q - x^p \\
    b_1 &= y^r - y^q \\
    b_2 &= y^p - y^r \\
    b_3 &= y^q - y^p
\end{align*}
\]

4.5 Evaluation of strippable grid

A test program has been performed with the purpose to check the applicability of the grid based on photographic paper\(^1,9\). This means basically that it must be checked whether the grid deforms in the same way as the specimen surface. Therefore a grid was glued on a 1.6 mm thick dog-bone specimen together with two strain gages, see Figure 4-15. All devices were bonded to the specimen using the same adhesive. The specimen was made of bare Aluminium 2024-T3. The deformation of the grid was analysed with the image processing system.

Because the strain gages are bonded in two perpendicular directions in the directions of the grid lines, the grid deformations can be easily compared with the strain gauges output. Based on the experience that the strain gages normally give an accurate strain output, it can be checked whether the grid really deforms in the same way as the specimen surface.

The recognition of the markers, the labelling and the determination of the marker centres in this test was done with the software package Image Pro Plus (IPP)\(^10\) since the Impress software was not yet ready at this time. This software package cannot be used in-situ. Therefore, the load was applied in a stepwise manner and at every load step images of the grid are taken and saved to disk. The analysis is done after the test. IPP is able to determine the marker centres based on sub-pixel level which means that the pixel-value is taken into account.
\[
\sum_{i=1}^{n} \sum_{j=1}^{n} = n \cdot \sum_{i=1}^{n} p_{v_i}
\]

\[x_{c_{\text{marker}}} = \frac{\sum_{i=1}^{n} x_{c_i} \cdot p_{v_i}}{n \cdot \sum_{i=1}^{n} p_{v_i}} \tag{eq. 4-7}
\]

where
- \(x_{c_{\text{marker}}} = x\)-value of marker centre
- \(x_{c_i} = x\)-value of centre of pixel \(i\)
- \(p_{v_i} = \text{pixel value of pixel } i\)
- \(n = \text{total amount of pixels that represent marker}\)

Figure 4-15, Specimen geometry and instrumentation.

For the calculation of strains in x- and y-direction, \(\varepsilon_x\) and \(\varepsilon_y\), an average of several measurements is taken. The strain in y- or loading direction was measured between two marker points with an internal distance in y-direction of either 700 or 1100 \(\mu m\). The presented values are an average of the strains for 19 marker combinations.

In x-direction, the two markers were positioned 1800 \(\mu m\) apart and the strain was an average of 9 marker combinations. The results are plotted in Figure 4-16. The figure illustrates that the strain of points either 700 or 1100 \(\mu m\) apart is very well comparable. After a few loads step in the plastic region, grid detachment from the specimen occurred and the results were no longer reliable. The contraction curve shows unexpected good grid behaviour under compression. It was expected that the grid would come loose from the aluminium surface due to contraction of the aluminium.
The expected absolute accuracy of 2 µm on a length of 1 mm, comparable with a minimum strain detection of 0.2 %, seems to be far too conservative. This is the result of taking the centre of the marker points with a size of at least 15 × 15 pixels, giving the centre co-ordinates an unexpected high accuracy.

4.6 Accuracy strain measurements with grid

The markers on the photographic paper have a size of approximately 50 x 50 µm. With a pixel size of 2.8 µm, approximately 18 rows and 18 columns represent a marker. This is illustrated in Figure 4-17a. Based on this size, an upper and lower boundary can be derived for the accuracy of the determination of a marker centre. This paragraph discusses the accuracy of the centre determination.

When a marker is represented by pixels that all have the same pixel value, the centre point of the marker is the average of the centre points of all pixels. The position for example of a marker centre in row direction is the average of the centre points of 25 columns. As illustrated in Figure 4-17b, the minimum movement of one row centre is equal to half of one pixel size and thus to 1.4 µm.

Figure 4-16, Tensile test results obtained with strain gauges and grid.
Unloaded situation

Centre point definition of one single row

± 18 pixels

(n + 0.5) x ps

Column

n x ps

c

c

a) original marker size

b) definition row centre

Figure 4-17, Accuracy marker movements.

The minimum movement of a marker centre in row direction is equal to 1 µm divided by the amount of rows. For a marker size of 25 rows, this minimum movement is equal to 1.4 µm / 18 = 0.08µm. This is the maximum accuracy that can be obtained with this method of centre determination.

In the worst case, all rows move one pixel in the same direction. In that case the marker centre shifts 1.4 µm, which represents the lowest accuracy of this method and is independent of the amount of rows.

The determination of different marker centres is independent from one another. Hence the maximum inaccuracy in the length measurement between two markers is 2 x 1.4 µm = 2.8 µm. For grid points 1000 µm apart this results in a deviation of ±0.3% from the average length, this increases to ±3% for grid points 100 µm apart. Bosker\textsuperscript{11} measured the deviations from the average distance between the same two grid points, respectively 1000, 500, 300 and 100 µm apart, for 25 arbitrary measurements. All measurements fall within the boundaries so the above assumptions seem to be reasonable.
References


5 BLUNT NOTCH DESIGN TOOL FOR GLARE

5.1 Introduction

Aeroplane structures are full of blunt notches, which are hard to avoid but decrease the static and dynamic properties of the material. Most of these notches are rivet holes. However, windows, doors and inspection holes can be regarded as blunt notches as well.

This chapter describes an extended study that was carried out to find applicable fracture criteria to support the development of a general blunt notch prediction tool. This tool should not only be able to predict the blunt notch strength for an arbitrary Glare variant, amount of layers and the thickness of the aluminium layers, but also for an arbitrary bi-axial loading condition.

The first phase in the development of a blunt notch tool was to understand the fracture mechanism and to find an appropriate failure criterion. Therefore, § 5.2 discusses research that was carried out to investigate the fracture mechanisms in more detail.

§ 5.3 presents and discusses the results of a large test program and compares these results with the Finite Element (FE) predictions. Based on these results a blunt notch calculation method is proposed in § 5.4 for uni-axially loaded Glare specimens in one of the principle material directions. In § 5.5, the FE model is introduced that is used to carry out predictions for other load cases. To check the validity of the model, the results are compared with basic tensile tests. § 5.6 presents a prediction tool based on these analyses. Finally, § 5.7 contains a summary and conclusions.

5.2 Failure mechanisms

To develop a blunt notch design tool for Glare, it is important to understand the blunt notch fracture mechanism for this material first. Based on previous research on the blunt notch behaviour of Glare material, discussed in chapter 3, it is expected that some fracture within the material will occur before final failure. Therefore, research has been carried out to investigate the fracture process. Eight comparable specimens are loaded to a different level of their blunt notch strength\(^1\). Subsequently, they are unloaded and the area around the notch is inspected with different techniques. This unloading had no influence on the observed fracture damage as will be discussed.
The tested specimens all originate from a single Glare 3 3/2 0.4 sheet with a thickness of 1.7 mm. Figure 5-1 illustrates the specimen geometry. The specimens were 50 mm wide, contained a centre hole with a diameter of 5 mm and were loaded in the L-direction.

Figure 5-1, Test specimen used for fracture research.

The test matrix is given in Table 5-1. Specimen nr 1 is loaded until fracture. The measured gross blunt notch strength is considered to be representative for all specimens. The remaining specimens are loaded between 58 % and 98 % of the determined gross blunt notch strength for specimen 1.

The “comments” column in Table 5-1 indicates whether the material in the net section was fully yielding at test interruption. This could be derived from the load - displacement curves that are initially linear. Between 40 and 45% of the failure load, small deviations from the linear relation are observed, indicating small-scale plasticity. These deviations remain small up to ± 70 % of the failure load when the net section starts to yield rapidly.

Table 5-1, Test matrix fracture research.

<table>
<thead>
<tr>
<th>Specimen</th>
<th>Gross stress [MPa]</th>
<th>Net stress [MPa]</th>
<th>Interruption load (% P_max)</th>
<th>Comment</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>414</td>
<td>459</td>
<td>100</td>
<td>fracture</td>
</tr>
<tr>
<td>2</td>
<td>394</td>
<td>437</td>
<td>98</td>
<td>elastic-plastic</td>
</tr>
<tr>
<td>3</td>
<td>354</td>
<td>392</td>
<td>87</td>
<td>elastic-plastic</td>
</tr>
<tr>
<td>4</td>
<td>333</td>
<td>370</td>
<td>83</td>
<td>elastic-plastic</td>
</tr>
<tr>
<td>5</td>
<td>313</td>
<td>347</td>
<td>78</td>
<td>elastic-plastic</td>
</tr>
<tr>
<td>6</td>
<td>294</td>
<td>326</td>
<td>73</td>
<td>elastic-plastic</td>
</tr>
<tr>
<td>7</td>
<td>263</td>
<td>292</td>
<td>65</td>
<td>elastic</td>
</tr>
<tr>
<td>8</td>
<td>234</td>
<td>260</td>
<td>58</td>
<td>elastic</td>
</tr>
</tbody>
</table>

After the tests, the specimens were first checked with an ultrasonic scanner (C-scan) for delaminations. The scan results for specimen 1 to 3 are illustrated in Figure 5-2. Specimen 1 was fractured and shows a large
amount of fibre failure delamination. Specimen 2 (98%) shows local debonding at the sides of the hole (black areas). These scans cannot reveal whether the fibres in these areas are broken or still intact. Around the hole in specimen 3 (87%), no clear debonds can be detected. However, the resolution of the C-scan images is only 1 mm and is therefore not accurate enough. The C-scan is able though, to discriminate the fibres directions as illustrated in Figure 5-2.

Subsequently, the specimens have been cut along the centre line in the loading direction. One half of each specimen has been chemically etched to remove the outer aluminium layers. The results for specimen 1, 2 and 3 are illustrated in Figure 5-3. This examination is based on the experience that defects in the bond layer between the outer aluminium layers and the prepreg appears as discoloured zones in the glass epoxy underneath the outer aluminium layers\(^2\).

The fibre failure delamination in specimen 1 with the broken fibres and the light coloured resin rich layer is clearly visible. It took place between the resin- and fibre rich layers of the prepreg. The length of this delamination is more pronounced at the free edge of the specimen than in the vicinity of the hole. Specimen 2 and 3 illustrate the presence of local defects in the fibre
layers. The debonding in specimen 2, previously observed in the C-scan results, is in general static delamination as most of the fibres are still intact. Some fibres at the edge of the hole were cracked probably caused during drilling of the hole. Delamination occurred between the resin and fibre rich interface within the prepreg. The height of the discoloured zone is comparable with the size in the fractured specimen 1. The debonding area was measured to be 3.5 mm high and 2 mm wide. The other specimens except for specimen 3 showed no clear signs of debonding.

Figure 5-4 shows the etched surfaces captured with a microscope (2.5 - 8x).

![Static delamination, Fibre splitting, Drilling effect](image)

a) specimen 2, 98% failure load  

![Fibre splitting](image)

b) specimen 3, 87% failure load

![Fibre splitting, 1 mm](image)

a) specimen 4, 83% failure load  

![Fibre splitting](image)

b) specimen 5, 78% failure load

![Fibre splitting](image)

a) specimen 6, 73% failure load  

![Fibre splitting](image)

b) specimen 7, 65% failure load

Figure 5-4, Magnified images after etching of specimen 1 to 7.
These larger magnifications show that there are lighter areas at the notch root also for specimen 3 to 6. These areas either indicate, debonding or matrix cracking. The picture for specimen 3 shows an abrupt difference between the decolourised area and the neighbouring area. This indicated the occurrence of fibre splitting. These areas (but smaller) are also observed for specimen 4, 5 and 6.

The other half of the etched specimens was used for investigation with a Scanning Electron Microscope (SEM). The specimens were cut in again in two parts along the net section and the newly created surface was polished. In this way, material was removed and, therefore, the microscopic pictures presented in this section are a cross section somewhat out of the net section. This is illustrated in Figure 5-5.

Figure 5-6 shows an image of specimen nr. 2 obtained with a SEM. The first fibres at the crack tip are not damaged as explained before and illustrated in Figure 5-5. Some fibres at the edge are pulled out of the prepreg caused by the drilling process. Fibre splitting is observed between the loaded fibres along the hole edge and the cut- and therefore unloaded fibres above the hole. Many matrix cracks are observed internally within the prepreg in loading direction and between this prepreg and the aluminium layers. A larger magnification is illustrated in Figure 5-7. It is clear that the fibres in loading direction are still intact and that the defects occur within the matrix. The delamination between the aluminium layer and the prepreg layer is also larger than could be observed in Figure 5-6. No damage between the differently orientated fibre layers was observed.
Figure 5-6, SEM image of specimen 2, 98% failure load.

Figure 5-7, Detailed view of prepreg defects within specimen 2.
The SEM pictures of specimen 3 illustrated the existence of some very small delamination areas between the aluminium and the prepreg layer in the loading direction. The inspection of specimens 4 to 8 did not result in observed damage.

Figure 5-8 shows a SEM picture of specimen 2, taken from the hole in the direction perpendicular to the loading direction. This picture clearly illustrates the large delamination between the fibre layer in loading direction and the aluminium layer.

Summarizing, no indication from fibre breakage prior to failure was obtained. The broken fibres that were observed were caused prior to testing by the drilling process. During loading, fibre splitting occurs within the matrix at the hole edge at approximately 75% of the failure load. Further load increase causes an increase of matrix cracking. Close to failure, static delaminations are created between the resin and fibre rich layer of the prepreg.
5.3 Experimental program

To develop a Glare blunt notch prediction tool, several experimental programs have been performed. A large test program investigated the uni-axial blunt notch strength of Glare 3, Glare 4A and Glare 4B laminates, tested under different angles. The initial test program to determine appropriate specimen geometry is described in § 5.3.1 while the extensive research is discussed in § 5.3.2 to § 5.3.4. Additionally, the blunt notch strength of Glare specimens loaded in pure shear and bi-axially-loaded Glare specimens are discussed in § 5.3.5 and § 5.3.6 respectively.

5.3.1 Determination specimen geometry

Before setting up a large uni-axial test blunt notch program, it is important to understand the context in which the results have to be used. The blunt notch strength is especially important for the design of an aeroplane in the case of rows of holes used for the attachment of stiffening elements, windowpanes or joints like lap- and butt joints. All these holes should be filled with rivets. The quantity of single, open holes like drain holes or holes for the attachment of antennas is much lower and these are normally reinforced. Therefore, the blunt notch strength that is investigated must be representative for a row of holes, filled with rivets.

Another problem is the width of the specimens. A too small width will give rise to boundary effects or result in unloaded fibres in the case of off-axis loading, see Figure 5-9, while a too large width is a waste of material.

![Figure 5-9](image)

Figure 5-9, Possible influence specimen width on realistic loading.

To answer these questions, an initial experimental program has been carried out to determine an appropriate specimen dimension. The tests were carried out on specimens of Glare 4A 3/2 0.3 material that contained one or more notches with a diameter of 4.8 mm. The specimen geometries are illustrated in Figure 5-10. Some single hole specimens contained 4.8
mm NAS 1097 AD-6 rivets that were installed in a 4.9 mm hole, in the same way as illustrated in Figure 5-12. The applied squeeze force was 24 kN. The test matrix is given in Table 5-2.

The average results for the gross- and net blunt notch strength are plotted in Figure 5-11, while the values are given in Table 5-2 as well.

The results in Figure 5-11 illustrate a reduction of the gross blunt notch strength for a specimen with a width of 50 mm compared to specimens with a width of 100 and 200 mm. Especially in the 45º-loading direction the reduction is significant and amounts up to 7.6%. The difference in gross blunt notch strength between the 100 and 200 mm width specimens, however, is negligible and never exceeds 1%. Looking at the net blunt notch strengths, however, reveals that the major part of this difference is due to the proportional increase of the net section. For the specimens tested in L-direction, the difference in net blunt notch strength amounts no more than 1% between 50 mm and 200 mm width. From this it is concluded that the optimum specimen width is 50 mm. The use of a 100 or 200 mm specimen would add only a little accuracy and would increase material costs considerably.
Table 5-2, Test matrix and average results initial blunt notch program on Glare 4A 3/2 0.3.

<table>
<thead>
<tr>
<th>Width</th>
<th>Test</th>
<th>Test direction</th>
<th>Nr</th>
<th>$\sigma_{\text{gross}}$ [MPa]</th>
<th>$\sigma_{\text{net}}$ [MPa]</th>
</tr>
</thead>
<tbody>
<tr>
<td>50 d/W = 0.096</td>
<td>Single hole</td>
<td>L</td>
<td>2</td>
<td>531</td>
<td>587</td>
</tr>
<tr>
<td></td>
<td></td>
<td>45$^\circ$</td>
<td>2</td>
<td>277</td>
<td>306</td>
</tr>
<tr>
<td></td>
<td></td>
<td>T</td>
<td>4</td>
<td>377</td>
<td>418</td>
</tr>
<tr>
<td>100 d/W = 0.048</td>
<td>Single hole</td>
<td>L</td>
<td>2</td>
<td>563</td>
<td>591</td>
</tr>
<tr>
<td></td>
<td></td>
<td>45$^\circ$</td>
<td>1</td>
<td>299</td>
<td>314</td>
</tr>
<tr>
<td></td>
<td></td>
<td>T</td>
<td>2</td>
<td>387</td>
<td>407</td>
</tr>
<tr>
<td></td>
<td>Single riveted hole</td>
<td>L</td>
<td>1</td>
<td>565</td>
<td>593</td>
</tr>
<tr>
<td>200 d/W = 0.024</td>
<td>Single hole</td>
<td>L</td>
<td>1</td>
<td>567</td>
<td>581</td>
</tr>
<tr>
<td></td>
<td></td>
<td>45$^\circ$</td>
<td>1</td>
<td>300</td>
<td>307</td>
</tr>
<tr>
<td></td>
<td></td>
<td>T</td>
<td>1</td>
<td>385</td>
<td>394</td>
</tr>
<tr>
<td></td>
<td>Single riveted hole</td>
<td>L</td>
<td>1</td>
<td>552</td>
<td>565</td>
</tr>
<tr>
<td></td>
<td>Three open holes</td>
<td>L</td>
<td>1</td>
<td>540</td>
<td>581</td>
</tr>
<tr>
<td></td>
<td></td>
<td>45$^\circ$</td>
<td>2</td>
<td>299</td>
<td>322</td>
</tr>
<tr>
<td></td>
<td></td>
<td>T</td>
<td>2</td>
<td>373</td>
<td>402</td>
</tr>
</tbody>
</table>

Figure 5-11 illustrates also that the difference in net blunt notch strength between single hole and multiple, three hole specimens is negligible. For specimens tested in L- or T- direction, the net strength difference did not exceed 2%. Apparently, the single hole net blunt notch strength is a good indication of the multiple hole net blunt notch strength. For this reason all specimens in this study were equipped with a single hole.

Figure 5-11, Test results initial blunt notch program.
Finally, no influence of the presence of an unloaded rivet on the net blunt notch strength was observed. Müller⁴, reported a slight increase of the blunt notch strength up to 3.7% for increasing squeeze force up to 15kN. He observed this in tests on Glare 3 3/2 0.3 material. For higher squeeze forces the blunt notch strength remained constant.

The 50 mm wide single open hole specimens are expected to be representative or slightly conservative to determine the net blunt notch strength of a rivet row. However, within the uni-axial test program described in the next paragraph, the specimens are tested both with open- and filled holes to verify this conclusion with more test results. The blunt notch strength values that are presented are all based on the net section.

5.3.2 Uni-axial blunt notch tests program

To investigate the general blunt notch behaviour of Glare, a large uni-axial test program has been carried out on 50 mm wide specimens that contained either one single open hole or a hole that was filled with a rivet⁵. The specimen geometry is illustrated in Figure 5-12.

In case of an open hole, the diameter of the hole was 4.8 mm while the filled holes had a diameter of 4.9 mm to fit a rivet with a 4.8 mm diameter. The specimens were both tested at the Delft University and at the Dutch National Aerospace Laboratory. The test matrix is given in Table 5-3.
Table 5-3. Test matrix of Glare uni-axial blunt notch specimens, for open and filled holes (3/1 means 3 specimens with an open hole and 1 filled hole specimen).

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Test direction</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>0°/L</td>
</tr>
<tr>
<td>Glare 3 3/2-0.3</td>
<td>3/1</td>
</tr>
<tr>
<td>Glare 3 4/3-0.4</td>
<td>3/1</td>
</tr>
<tr>
<td>Glare 3 4/3-0.5</td>
<td>3/1</td>
</tr>
<tr>
<td>Glare 3 6/5-0.4</td>
<td>3/1</td>
</tr>
<tr>
<td>Glare 3 8/7-0.4</td>
<td>2/1</td>
</tr>
<tr>
<td>Glare 3 8/7-0.5</td>
<td>2/1</td>
</tr>
<tr>
<td>Glare 4A 3/2-0.3</td>
<td>3/1</td>
</tr>
<tr>
<td>Glare 4A 4/3-0.4</td>
<td>3/1</td>
</tr>
<tr>
<td>Glare 4A 8/7-0.4</td>
<td>2/1</td>
</tr>
<tr>
<td>Glare 4B 3/2-0.3</td>
<td>3/1</td>
</tr>
<tr>
<td>Glare 4B 4/3-0.4</td>
<td>3/1</td>
</tr>
<tr>
<td>Glare 4B 4/3-0.5</td>
<td>2/2</td>
</tr>
<tr>
<td>Glare 4B 6/5-0.4</td>
<td>2/1</td>
</tr>
<tr>
<td>Glare 4B 8/7-0.4</td>
<td>2/1</td>
</tr>
<tr>
<td>Glare 4B 8/7-0.5</td>
<td>2/1</td>
</tr>
<tr>
<td>Total</td>
<td>38/16</td>
</tr>
</tbody>
</table>

The program consisted of:

- 71 Glare 3 specimens i.e., 52 open hole specimens tested at the NLR and 19 filled hole specimens tested at DUT
- 56 Glare 4A specimens i.e., 41 open hole specimens and 15 filled hole specimens
- 81 Glare 4B specimens i.e., 59 open hole specimens and 22 filled hole specimens,
- 12 Glare 3 4/3 0.4 specimens with rivets that were applied with different squeeze forces i.e., \( F_{sq} = 10 \text{kN}, 20 \text{ and } 26 \text{ kN}. \) 4 specimens contained no rivets and were used as reference.
- 72 bare 2024-T3 specimens i.e., 54 open hole specimens and 18 filled hole specimens varying in thickness between 0.6 and 6.4 mm \( (F_{sq} = 26 \text{kN}). \) The matrix is given in Table 5-4.
- 15 specimens made of Glass prepreg i.e., 5 x Glare 3 lay-up \((0°/90°/90°/0°/0°/90°/90°/0°/0°)\) and 10 x Glare 4 lay-up \((90°/0°/90°/90°/0°/90°/90°/0°/90°).\)
The rivets in the filled hole specimens carried no load. To prevent the presence of a countersunk hole in the Glare material, the rivets were installed with unloaded aluminium fillers.

Table 5-4, Test matrix of 2024-T3 uni-axial test blunt notch specimens, for open holes and filled holes respectively (3/1 means 3 open hole and 1 filled hole specimen). Rivets: NAS1097 AD6, $F_{sq} = 26$ kN.

<table>
<thead>
<tr>
<th>Dir.</th>
<th>Thickness 2024-T3 specimens [mm]</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>0.6</td>
</tr>
<tr>
<td>L</td>
<td>3</td>
</tr>
<tr>
<td>T</td>
<td>3</td>
</tr>
</tbody>
</table>

Grey cells represent specimens tested with both open and filled holes

5.3.3 Uni-axial blunt notch tests results

Aluminium

Figure 5-13 shows the average net blunt notch strength results for 2024-T3.

Figure 5-13, Average blunt notch results 2024-T3.

The graph indicates that the blunt notch strength of aluminium 2024-T3 increases with the thickness. It also illustrates the difference between the net blunt notch strength of an open hole and a filled hole. In general it can be concluded that holes that are filled with a rivet result in slightly higher blunt notch strengths compared to open holes. The average net blunt notch strength increase for the 18 tested specimens was 2.6%. The average
results for the filled hole specimens tested in T direction and with a thickness of 2.5 mm are clearly not following the trend represented by the other results. No explanation was found. Since this is the average result of three specimens, it is expected that this is caused by aluminium batch differences.

Based on these results, the net blunt notch strength for 2024-T3 in L - and T-direction is predicted for sheet thicknesses between 0.3 and 0.5 mm. These results are given in Figure 5-13 as well.

**Prepreg**

The prepreg specimens with the prepreg lay-up of Glare 3 and Glare 4 but omitting the aluminium layers have been tested in one fibre direction only. The Glare 4 prepreg specimens were loaded in the direction with the least amount of fibres (33% of the fibres).

For the Glare 3 prepreg specimens, first fibre failure was observed at ca. 400 MPa net stress while the average net blunt notch strength was 525 MPa. The average net blunt notch strength for the Glare 4 prepreg with 33% of the fibres in loading direction was 337 MPa.

Considering the prepreg characteristics given in table 1-3, The Kt value for the Glare 3 prepreg specimens is 3.7. Assuming elastic behaviour of the laminate to failure and a failure strain of 4.2%, the predicted strength is 385 MPa what estimates with the experimental results.

**Glare open hole test results**

Figure 5-14 illustrates a typical load versus elongation curve for a uni-axial Glare blunt notch specimen. As discussed before in § 5.2, this curve illustrates an initial linear relation what means that there is no plastic zone in the net section. After some time, the curve starts to deflect from the linear relation but only slowly. This represents the occurrence of local plasticity. Suddenly, the curve turns when the net section is fully yielding. After some more elongation, the curve becomes linear again, representing yielding of the whole laminate.

Figure 5-15 shows the measured far field stresses as a function of the applied gross stress for a Glare 4B laminate that is uni-axially loaded under different angles. All specimens are fully yielding at the onset of fracture since the far field stresses are clearly above the yield strain of approximately 0.5% the location of the far field strain gage is illustrated in Figure 5-21.
Blunt notch design tool for Glare

Figure 5-14, Typical load-elongation curve for a uni axially loaded Glare blunt notch specimen (Glare 3 4/3 0.5).

Figure 5-15, Far field strain gauge results for Glare 4B laminates tested in different directions.

The specimen loaded in the L direction and 33% of the fibres (0 degr.) shows a comparable curve as the load – displacement curve presented in Figure 5-14 with a significant amount of strain hardening. The other curves for specimens tested under large angles with the main fibre directions show
hardly any strain hardening but an extensive far field strain at failure. Evidently, the shear properties of the fibre layers are poor.

The average uniaxial Glare test results for the open hole specimens are listed in Table 5-5 and presented in Figure 5-16. There are blunt notch differences for a certain Glare variant between the different lay-up and aluminium layer thicknesses. However, these are rather small and will be discussed later. The lines illustrate the trend for a certain Glare variant of the blunt notch strength as a function of the off-axis angle. The uniaxial blunt notch strength decreases towards an off axis angle of 45º for all Glare variants. The expectation that the curve for Glare 3 should be more or less symmetric compared to the 45º off axis angle is confirmed by the test results. The only differences should be due to the rolling direction of the aluminium layers and the fibre direction of the outer prepreg layer connected to the outer aluminium layers.

Table 5-5, Net blunt notch test results of Glare uni-axial blunt notch specimens with open holes.

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Test direction</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>0º/L</td>
</tr>
<tr>
<td>Glare 3 3/2-0.3</td>
<td>446</td>
</tr>
<tr>
<td>Glare 3 4/3-0.4</td>
<td>436</td>
</tr>
<tr>
<td>Glare 3 4/3-0.5</td>
<td>445</td>
</tr>
<tr>
<td>Glare 3 6/5-0.4</td>
<td>486</td>
</tr>
<tr>
<td>Glare 3 8/7-0.4</td>
<td>461</td>
</tr>
<tr>
<td>Glare 3 8/7-0.5</td>
<td>495</td>
</tr>
<tr>
<td>Glare 4A 3/2-0.3</td>
<td>599</td>
</tr>
<tr>
<td>Glare 4A 4/3-0.4</td>
<td>545</td>
</tr>
<tr>
<td>Glare 4A 8/7-0.4</td>
<td>594</td>
</tr>
<tr>
<td>Glare 4B 3/2-0.3</td>
<td>396</td>
</tr>
<tr>
<td>Glare 4B 4/3-0.4</td>
<td>382</td>
</tr>
<tr>
<td>Glare 4B 6/5-0.4</td>
<td>416</td>
</tr>
<tr>
<td>Glare 4B 8/7-0.4</td>
<td>409</td>
</tr>
<tr>
<td>Glare 4B 8/7-0.5</td>
<td>441</td>
</tr>
</tbody>
</table>

The results for Glare 4A and Glare 4B laminates are comparable if they are mirrored along the off axis angle of 45º. The largest blunt notch strength is of course obtained when the laminate is loaded in the direction with the maximum amount of fibres.
An important aspect of the blunt notch results is the amount of scatter in the results. Figure 5-17 shows the extrapolated scatter distributions for the Glare 3, Glare 4A and Glare 4B uni axial net blunt notch results. All Glare variants individually seem to have a normal scatter distribution where the distribution for Glare 4A and Glare 4B is comparable and the distribution for Glare 3 is slightly less wide.

The distributions are added to obtain a larger collection of results, which is allowed, assumed that the scatter distribution is no function of the Glare variant. This extrapolation shows clearly the symmetrical normal distribution of the scatter in the test results. More than 97% of the results show an absolute difference with the average results that is less than 3.5%.

The difference between different lay-ups and aluminium thicknesses within one Glare variant is best illustrated by plotting the net blunt notch results as a function of the MVF. This is illustrated in Error! Reference source not found. for Glare 3 laminates tested in L direction. Additionally, test results of Matoush⁹ are added for W = 25mm and d = 5 m. The results show that the blunt notch strength for this variant decreases with increasing MVF. A line is drawn in this figure that connects the obtained Glare 3 prepreg result (MVF = 0, σ_{bn} = 525 MPa) and the average aluminium blunt notch strength in L direction obtained from Figure 5-13 (MVF = 1, σ_{bn} = 410 MPa). All test results lie within a bandwidth between - 5% and 12% relatively to this line. Comparable results are found for Glare 4A and Glare 4B laminates.
Whether the blunt notch strength decreases or increases with the MVF depends on the amount of fibres in loading direction.

Figure 5-17, Illustration of deviation of results from the average values.

Figure 5-18, Uni-axial blunt notch results for Glare 3 variants tested in L-direction as a function of the MVF.
Open versus filled holes
The influence of non-load carrying rivets on the blunt notch strength is negligible for both Glare 3 and Glare 4. The test results are given in Table 5-6 and plotted in Figure 5-19 for different Glare variants that were tested in L-direction.

Table 5-6, Net blunt notch test results of Glare uni-axial blunt notch specimens with filled holes.

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Test direction</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>0°/L</td>
</tr>
<tr>
<td>Glare 3/2-0.3</td>
<td>468</td>
</tr>
<tr>
<td>Glare 4/3-0.4</td>
<td>454</td>
</tr>
<tr>
<td>Glare 3/3-0.5</td>
<td>458</td>
</tr>
<tr>
<td>Glare 6/5-0.4</td>
<td>478</td>
</tr>
<tr>
<td>Glare 3/7-0.4</td>
<td>447</td>
</tr>
<tr>
<td>Glare 3/7-0.5</td>
<td>500</td>
</tr>
<tr>
<td>Glare 4A/2-0.3</td>
<td>573</td>
</tr>
<tr>
<td>Glare 4A/3-0.4</td>
<td>546</td>
</tr>
<tr>
<td>Glare 4A/7-0.4</td>
<td>537</td>
</tr>
<tr>
<td>Glare 4B/2-0.3</td>
<td>408</td>
</tr>
<tr>
<td>Glare 4B/3-0.4</td>
<td>394</td>
</tr>
<tr>
<td>Glare 4B/6-0.4</td>
<td>404</td>
</tr>
<tr>
<td>Glare 4B/7-0.4</td>
<td>405</td>
</tr>
<tr>
<td>Glare 4B/7-0.5</td>
<td>418</td>
</tr>
</tbody>
</table>

Figure 5-19, Blunt notch results of open holes versus filled holes.
The influence of the rivet squeeze force on the blunt notch strength of Glare3-4/3-0.4 is depicted in Figure 5-20. The blunt notch strength seems to be independent of the rivet squeeze force; the increase of net blunt notch strength between an open hole specimen and a 26 kN squeeze forced rivet specimen is negligible. Müller found the same result\(^4\).

![Figure 5-20, Influence of rivet squeeze force for Glare 3 4/3 0.4 specimens.](image)

5.3.4 **Uni-axial grid deformation results**

17 uni-axial blunt notch specimens were provided with a grid to perform grid deformation measurements. The instrumentation is illustrated in Figure 5-21.

![Figure 5-21, Grid and strain gauge instrumentation of uni axial specimens.](image)
The specimens contained two strain gauges to measure the strain in loading direction; one in the far field to measure the undisturbed strain (1) and one at the notch root at the backside of the grid to verify the grid measurements (2).

Figure 5-22 shows the measured strains in loading direction in the net section for different far field strains. The material is Glare 4B 4/3 0.5 that was tested in T-direction. The first strains were measured at a distance of 0.2 mm in front of the crack tip. The figure also shows the strains at the notch root measured with a strain gauge. Since these strain results are average strains of the area covered by the gauge, they represent the strain at a certain distance from the notch root. It is clear that the grid and strain gauge results estimate very well.

Figure 5-22, Strain measurements with grid for different far field strains.

Figure 5-23 shows the measured strain development in the far field and the notch root for the same specimen, as a function of the applied load. The results for the grid illustrate that the strains at the notch root at the onset of failure are much larger than the ultimate strain of the fibres. This indicates that the fibres at the notch are either broken or delamination has occurred between the prepreg and aluminium layers. Based on the results presented in § 5.2, this must be delamination. The maximum strain measured at the notch root is approximately the ultimate strain of the aluminium alloy.
Three (linear) parts can be distinguished in the stress-strain curves presented in Figure 5-23. These are subsequently local plasticity in the net section, fully yielding of the specimen and a strong strain increase at the notch root due to delamination.

Figure 5-23, Strain in far field and at notch root as a function of the applied load.

Figure 5-24, Grid strains at notch root for Glare 4B 4/3 0.5 specimens tested under different angles with the principle fibre directions.
Blunt notch design tool for Glare

When there is a non-zero angle between the loading direction and the principal fibre directions, the prepreg layers must be partially loaded in shear. Figure 5-24 illustrates that this results in a clear decrease of the blunt notch strength. Hardly any strain hardening is observed at the notch root, which illustrates the fact that the aluminium layers have to carry the load. The next paragraph discusses a test program that is carried out to investigate the blunt notch strength of specimens that are loaded in pure shear.

5.3.5 Blunt notch under shear loading

A total of 5 Glare blunt notch specimens have been tested in shear only in a so called 3-rail shear test. This test is derived from the ASTM standard D-5379 M-93 to test materials in shear. The set up and the definition of the specimen orientation with the external load is given in Figure 5-25. The specimens have two measurement sections that should be loaded in pure shear. Load is introduced at the edges and in the centre via bolts. To protect the specimen, these locations are reinforced with bonded aluminium doublers.

Figure 5-25, Shear test set-up.
Before drilling holes in the measurement sections, it has been investigated whether the measurement sections are really loaded in shear only. Therefore, a single test was carried out on a specimen without holes and instrumented with strain gauges. The results of the strain gauge readings that are illustrated in Figure 5-26 show that a situation of pure shear exist in the specimens up to a shear stress of 160 MPa.

Figure 5-26, Investigation of strain field in 3 rail shear test.

Figure 5-27, Strain gauge results blunt notch 3 rail shear test.
In every measurement section of the 3 rail shear specimens, 4 mm holes were drilled. Around the holes and in the far field, strain gauges were bonded on the specimens. Figure 5-27 shows the strain readings as a function of the applied shear load. The holes did not fail but deformed extensively. This is illustrated by the large strains that were measured under 45 degrees. In a real situation, the holes are normally filled with rivets that will prevent these deformations. Unfortunately, this has not been tested. The experiment was stopped when the large deformation of the specimens caused problems for the loading mechanism. The far field strain distribution was comparable with the results obtained in the reference specimen without holes.

The theoretical difference between the results obtained with the 3 rail shear test and the uni-axial test results of specimen loaded under under 45º is the presence of an additional bi-axial loading component in the direction of the fibres, see Figure 5-28. As described in chapter 3, this has a favourable effect on the stress concentration around the notch.

Figure 5-28, Stress situation in a uni-axial loaded specimen with fibres under an angle of 45º with the external load.

A qualitative comparison of the 3 rail shear stress-strain results around the crack tip of Figure 5-27 with the uni-axial grid results of Figure 5-24 for Glare 4B tested under 45º illustrates that yielding of the material occurs earlier in the specimen loaded in pure shear. The strain results of the 3 rail shear tests are used as a verification of the FE modelling, described in § 5.5.
5.3.6  *Bi-axial blunt notch tests*

Besides the uniaxial experiments discussed in § 5.3.1 and § 5.3.2, 4 Glare blunt notch specimens have been loaded bi-axially at the NLR laboratories in the Netherlands. The specimen geometry is given in Figure 5-29 and the test matrix with the tested materials is given in Table 5-7.

![Figure 5-29, Bi-axial test specimen with fixtures](image)

By applying different angles between the loading direction and the fibre orientation within the laminates, a combination of bi-axially loading and shear loads around the notches is created. The off axis angle $\phi$ is defined as the angle between the horizontal axis and the rolling direction $L$ of the aluminium layers. The load ratio was kept the same for all specimens; $F_y / F_x = 0.5$. The load introduction into the specimen was done via Aramid fibre prepregs that were bonded to the specimen and steel clamping plates. These are connected with pin-loaded holes to long tension struts. This set up allows the specimen to rotate in its plane and results in a nearly 90° angle between the loading direction and the specimen edges. The tests were performed at room temperature. A picture of the test set-up is given in Figure 5-30.
Table 5-7, Test matrix bi-axially loaded specimens

<table>
<thead>
<tr>
<th>Material</th>
<th>Specimen number</th>
<th>$\phi$ [deg.]</th>
<th>Thickness [mm]</th>
<th>$\sigma_x$ [$\text{MPa}$]</th>
<th>$\sigma_y$ [$\text{MPa}$]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3/2 0.3</td>
<td>1</td>
<td>15°</td>
<td>1.46</td>
<td>423</td>
<td>214</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>30°</td>
<td>1.56</td>
<td>390</td>
<td>196</td>
</tr>
<tr>
<td>Glare 4B 3/2 0.3</td>
<td>3</td>
<td>30°</td>
<td>1.71</td>
<td>377</td>
<td>185</td>
</tr>
<tr>
<td></td>
<td>4</td>
<td>60°</td>
<td>1.71</td>
<td>360</td>
<td>177</td>
</tr>
</tbody>
</table>

1) Gross stresses at onset of clamping failure

Figure 5-30, Overview of test set-up and load introduction.

The specimens were instrumented with several strain gages in either $x$ or $y$-direction. One side of the specimen contained a grid around the hole edge while a strain gage was bonded at the other side to check the grid deformation results. The instrumentation is illustrated in Figure 5-31.
Test results
In all four tests, the aramid fibre connection between the specimen and the steel clamping plates failed before blunt notch failure occurred. This failure occurred at the location where the aramid was bonded to the specimen as illustrated in Figure 5-32. Table 5-7 gives the applied gross stresses to the specimens at the time of clamping failure. These stresses at failure of the clamps were higher than the uni-axial blunt notch values for comparable specimens and angles what illustrates that a bi-axial loading condition is indeed favourable for the blunt notch strength.

Figure 5-32, Shear failure in the bond line between the aramid prepreg and the Glare specimen.

After clamping failure, the Glare 4B-3/2-0.3-30° specimen (nr. 3) showed some small cracks (ca. 5 mm) originating from the notch and perpendicular to the direction of highest load. This is illustrated in Figure 5-33.

No decisive answer can be given whether these cracks occurred before or just after failure of the clamping. A possible explanation could be that the vertical aramid fibre connection of the specimen failed first – due to the large displacement in the horizontal direction, transverse to the fibres in the aramid prepreg – and as a result the stresses in vertical direction suddenly dropped. The remaining stress in horizontal direction causes a higher stress concentration at the top and bottom of the notch, which caused the cracks.
Blunt notch design tool for Glare

Figure 5-33, Cracks perpendicular to the highest loading direction after failure of the clamping

Figure 5-34 shows an example of the strain measurements with grid and strain gauges for a bi-axially loaded Glare 4B specimen that was loaded under an angle of 30º with the main loading direction. The applied stress along the horizontal axis is the applied stress in the highest loading direction.

Figure 5-34, Strain measurements for the 30º off-axis Glare 4B 3/2 0.3 specimen in highest loading direction (+FE predictions)

The strain in the grid was measured at the notch and is therefore not equal to the presented strain of strain gauge 5 below the grid. This strain gauge failed rather early (indicated by the zero strain). The grid strains that are developed are rather large and much higher than the ultimate strain in the fibres. The FE results that are presented in this chart are discussed later. Figure 5-35 illustrates the strain measurements with the grid in the highest
loading direction along the net section for different applied stresses. The grid and strain gauge measurements are used to verify and support the FE calculations.

Figure 5-35, Grid strain measurements for different far field stresses.

5.4 MVF approach

An easy method to predict the net blunt notch strength for a Glare laminate in L or T direction has been proposed by Roebroeks. It is based on the Metal Volume Fraction approach and assumes a linear relation between the MVF and the net blunt notch strength. This approach gives satisfying results for properties like the (shear) yield strength, Young's modulus, ultimate strength and the specific weight.

Based on the MVF approach, the net blunt net strength can be expressed as:

\[
\sigma_{bn(Glare)} = \sigma_{bn(prepgl)} + \text{MVF} \cdot \left( \sigma_{bn(Al)} - \sigma_{bn(prepgl)} \right)
\]

eq. 5-1

The average blunt notch strength of the aluminium layers with 0.3, 0.4 and 0.5 mm thickness follows from Figure 5-13 and is approximately 415 MPa in L direction and 386 MPa in T direction. The net blunt notch strength of different prepregs was discussed in § 5.3.2 as well. A Glare 3 like prepreg (0/90) showed net blunt notch strength of 525 MPa in either fibre direction. Assuming that the fibres perpendicular to the loading direction add no direct strength to the blunt notch, the net blunt notch strength of a single UD prepreg (0) should be 1050 MPa. In this way a 0/90/0 prepreg should result
in a net blunt notch strength of 1050 MPa * 2 /3 = 700 MPa and a 90/0/90 prepreg should give a net blunt notch strength of 1050 MPa * 1 /3 = 350 MPa. This estimates with the result presented before of 337 MPa (difference = 3.9%). In this derivation, it is assumed that every single UD prepreg layer has a constant thickness of 0.125 mm, also when the prepreg is assembled from different single UD prepregs.

Expression 5-1 will result in a linear relation that is specific for every Glare variant. To allow comparison between different Glare variants, this equation can be changed into:

\[
\sigma_{bn(Glare)} = FVF_{LD} \cdot \sigma_{bn(UD)} + MVF \cdot \sigma_{bn(Al)} = \sigma_{bn(UD)} \frac{FVF_{LD} + MVF}{\sigma_{bn(Al)}} \cdot \sigma_{bn(Al)} \text{ eq. 5-2}
\]

with:

\[FVF_{LD} = \text{Fibre Volume Fraction of prepreg layers in loading direction, defined in the same way as the MVF in eq. 1.1}\]

\[\sigma_{bn(UD)} = \text{Net blunt notch strength of UD layer in fibre direction (} \approx 1050\text{MPa)}\]

\[\sigma_{bn(Al)} = \text{Net blunt notch strength of Al layer, either in rolling direction or perpendicular to the rolling direction (respectively 415 or 386 MPa)}\]

Figure 5-36, Glare blunt notch results and predictions in L- and T direction.
Figure 5-36 contains the blunt notch test results in L and T direction as a function of the MFV and FVF. Additionally to the current test results, results of Mattousch\(^8\) are added to this figure which were obtained with specimens with a different \(d\), \(W\) and \(d/W\) ratio, see also § 3.3.1. The lines in this figure represent the predictions based on eq. 5-2.

The predictions are in general somewhat conservative. The deviation between the predictions and the test results is plotted in Figure 5-37.

![Figure 5-37](image)

Figure 5-37, Accuracy of predictions based on MVF and FVF\(_{LD}\).

### 5.5 FE calculations

To develop a general calculation method for the blunt notch strength predictions of an arbitrary Glare laminate that is loaded under arbitrary load combinations, an extensive amount of FE calculations have been carried out. The results of these calculations supported by the experimental results can help to find appropriate failure criteria because:

- the FE calculations provide a detailed overview of the stresses and strains within the material layers at failure, and
- are cheap and fast to generate results for different loading conditions, with the aim to create a 3D failure surface

The Glare materials are modelled in ANSYS with quadratic Shell 91 8-node layered shell elements\(^9\). The layered option of the element is used to
Blunt notch design tool for Glare

separately model the aluminium layers and the prepreg layer. Since the structure is in plane stress it was decided not to model each separate material layer. To preserve symmetry, it is decided to model two plastic aluminium layers with one elastic prepreg layer in between. The thickness of the aluminium and prepreg layers can be adjusted to the MVF, applicable for the Glare lay-up in question. The first step of the FE approach is to validate the material model. This is described in 5.5.1

The prepreg layer is modelled with homogenized orthotropic linear material properties. The aluminium layers are modelled with a multi-linear isotropic hardening material behaviour using the Von Mises yield criterion, assuming an ideal plastic behaviour. The material properties used to model the different layers are given in Table 5-8.

Table 5-8, Material properties used in FEM calculations9.

<table>
<thead>
<tr>
<th></th>
<th>2024-T3</th>
<th>Prepreg</th>
</tr>
</thead>
<tbody>
<tr>
<td>L (MPa)</td>
<td>400</td>
<td>-</td>
</tr>
<tr>
<td>T (MPa)</td>
<td>330</td>
<td>-</td>
</tr>
<tr>
<td>UD</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>cross-ply 0º/90º</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>cross-ply 0º/90º/0º</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>σ_y (MPa)</td>
<td>400</td>
<td>330</td>
</tr>
<tr>
<td>E1 (GPa)</td>
<td>71.4</td>
<td>70</td>
</tr>
<tr>
<td>E2 (GPa)</td>
<td>-</td>
<td>.54</td>
</tr>
<tr>
<td>G12 (MPa)</td>
<td>0</td>
<td>27</td>
</tr>
<tr>
<td>ψ12</td>
<td>0.33</td>
<td>0.33</td>
</tr>
<tr>
<td>ψ21</td>
<td>0.33</td>
<td>0.0575</td>
</tr>
<tr>
<td>ε_ult(%)</td>
<td>17</td>
<td>4.5</td>
</tr>
</tbody>
</table>

5.5.1 Uni-axial dog bone specimens

To verify the material model one quarter of the dog-bone specimen geometry illustrated in Figure 5-38 is modelled. The calculations have been carried out for several Glare variants and are compared with experimental results. Figure 5-39 illustrates the experimental results for a Glare 4b 4/3 0.5 material compared with FE calculations. It is clear that the model is able to describe the basic material behaviour.
5.5.2 Uni-axial blunt notch specimens

First failure is defined by the limit criterion of maximum strain. This criterion is rather simple and (therefore) widely used in the industry for composites. The maximum strain criterion predicts the failure load by comparing the locally occurring laminate strains with the corresponding ultimate strain.
There is no interaction assumed among these strains. Failure is expected to occur if:

\[
\left( \frac{\varepsilon}{\varepsilon_{\text{ult}}} \right)_{i} = 1 \quad \text{eq. 5-3}
\]

where \( i \) distinguishes whether the criterion is applied for the aluminium or the prepreg layers. In case of aluminium, the Von Mises strains are compared with the ultimate stress of the aluminium alloy. The failure criterion of the prepreg layers is the ultimate strain of the fibres. Fracture is predicted when the calculated strain in the fibres at the notch root reaches this value.

The FE model has the possibility to change \( d \) and \( W \). Since the specimen is symmetric with respect to two axes, only one quarter of the specimen is modelled. The loading is applied as a uniformly distributed running load at the location of the clamping. The nodes at this edge are fixed in y-direction to simulate the clamping. The other edge is free.

An additional model with roughly four times the amount of elements was analysed to check the convergence of the calculations.

![Graph showing strain distribution](image)

Figure 5-40, FE calculations compared with grid measurements (Glare 4B).

From Figure 5-40, which depicts the strains along the net section for Glare 4B-4/3-0.5 in T-direction for various far field strains, can be seen that the
Chapter 5

FEM results closely correspond with the grid deformation method at the net section.

Due to the occurrence of delaminations, the 4.5 % ultimate strain failure criteria used for the fibres yield very conservative results. The experimental measurements illustrated that larger strains occurred at the notch edge before fracture as illustrated for example in Figure 5-23. The failure predictions are too low; 28%, 42%, and 55% for Glare 4 in 33% fibre direction, Glare 3, and Glare 4 in 66% fibre direction respectively. Figure 5-41 shows the FE predictions for a Glare 3 variant tested in L direction, together with experimental data. These deviations are comparable over the total range of MVF’s that exist for a certain Glare variant.

![Figure 5-41](image)

Figure 5-41, Uni-axial blunt notch strength predictions based on FE calculations, not taking into account delamination.

5.5.3 **Shear blunt notch specimens**

Since the shear specimen is symmetric with respect to the y-axis, only one half of the specimen is modelled. The width and length of the model is the same as one side of the shear specimen (respectively 20 and 152 mm). The model is meshed with 150 quadrangular SHELL 91 elements. The boundary conditions were set to maintain a constant distance between the left and the right side like in a 3-rail shear test. The force was introduced in the upper right corner of the model.
Figure 5-42 illustrates the results of one 3-rail shear test together with the FE results. During the FE analyses, it turned out that the initially applied prepreg shear stiffness of 5548 MPa in the FE calculations was far too high. In reality, the shear stiffness of a 0/90 prepreg layer is dominated by the stiffness of the epoxy. When the aluminium becomes plastics the fibres will shear. Therefore, much lower shear prepreg shear stiffness appeared to be more realistic. In the FE calculations, a shear stiffness of just 400 MPa has been applied to obtain comparable results.

The new prepreg value has also been applied to analyse the grid deformation results of uni-axial blunt notch specimens with fibres under an angle with the loading direction. It was concluded that a prepreg shear stiffness of 400 MPa complies far better with the grid deformations. It was decided to use this shear stiffness for the FEM calculations with arbitrary load combinations.

### 5.5.4 Bi-axial blunt notch specimens

For the bi-axial specimens a model is created with a width and length of 50 mm and a hole diameter of 4.8 mm. These dimensions were found to be sufficient to keep any stress field disturbances from the hole away from the load introduction.

In Figure 5-34, the strains near the notch root measured with strain gages and the Grid Deformation Method are depicted and compared with the FE calculations. The figure illustrates that strains near the notch root can be
very well predicted with non-linear FE calculations. The measurement of
strains with a grid appeared to be a very reliable method for determining
strains on a very local scale, especially at higher applied loads.

5.6 Blunt notch strength prediction tool

The purpose of the blunt notch strength prediction tool is to predict the blunt
notch strength of an arbitrary Glare laminate under an arbitrary load in-
plane combination of tension, compression and shear. Based on the results
discussed in § 5.5 it is expected that the influence of the Glare variant can
be solved by the characteristic MVF, eventually extended with an effective
FVF. To obtain a tool for different loading conditions, the idea is to create
“3D fracture surfaces” depending on the MVF, which predict failure for an
arbitrary loading combination.

Since the FE calculations illustrated to be very well capable of predicting
the strain distribution around the notches under arbitrary loading conditions,
they will be used to create this 3D fracture surface.

![Finite Element model 'Bi-axial load model'.](image)

The major problem in this approach is to find one or more valid fracture
criteria. An extensive FE analysis as been carried out\(^1\) to determine the 3D
fracture surface based on the ultimate strain criterion. Therefore, two
general FE models with the same geometry and different loading and
boundary conditions were developed. The width and length of the models
are set at 50 mm. The models contain a hole with a diameter of 4.8 mm.

\(^1\)
Blunt notch design tool for Glare

The geometry and the boundary conditions for the bi-axial load case (no applied shear load) are depicted in Figure 5-43.

The geometry and the boundary conditions for the case with a uniform applied load in x-direction and an additional applied shear load (no applied load in y-direction) are depicted in Figure 5-44.

\[ F_{yI} = - F_{yII} \]

Figure 5-44, Finite Element model ‘Uni-axial + shear load’.

The shape of the deformed left side (C) in Figure 5-44 is the same as the deformed right side (A) in order to incorporate (future) possibilities to simulate rivet rows. This means that symmetry conditions on one side are not sufficient. To fulfil this requirement, the displacements of the nodes in x and y-direction on the right side are the same as the displacements of the left side nodes added with the corresponding strain. Also, the in-plane rotation of each right side node and the corresponding left side node are coupled.

With these models, several curves have been created that should represent a critical loading combination. Since these curves represent first fibre failure, a calibration is necessary to predict final failure.

The FE calculations carried out before for the uni-axial blunt notch experiments showed that the final blunt notch strength can be calculated from the FEM predicted first fibre failure (4.5% strain in fibre direction) by multiplying the first fibre failure in both directions by a factor \( C \)

- Glare3: \( C_{\text{Glare3}} = 1.42 \)
- Glare4 in 66% fibre direction: \( C_{\text{Glare4, 66\%}} = 1.55 \)
Glare4 in 33% fibre direction: \[ C_{\text{Glare4, 33\%}} = 1.28 \]

These factors are the corrections for not taking into account the occurrence of delamination and seem to be constant as will be illustrated in Figure 5-45 and Figure 5-46. In these figures, the translation from predicted fibre failure and final failure is depicted for Glare 3-3/2-0.2 and Glare 4A 3/2 0.3 respectively. In Figure 5-45, the calculated blunt notch strength for a stress combination of \( \sigma_x : \sigma_y = 2:1 \) is compared with bi-axial stress performed by Van Rijn. These tests were performed on specimens with a hole diameter of 8 mm. The test results and the predicted final failure agree very well for these bi-axial tests.

![Figure 5-45](image.png)

Figure 5-45, Predicted failure compared to final failure according to FE calculations for Glare 3-3/2-0.2 under bi-axial loading.

Bi-axial tensile loading has a favourable effect on the blunt notch strength as it reduces the stress concentration around the hole. However, when a shear stress is applied as well, the blunt notch strength decreases. This is illustrated in Figure 5-47. This means that the cross sections of the failure surfaces presented in Figure 5-45 and Figure 5-46 shrink when a shear load is applied. This is illustrated in Figure 5-48.

The four bi-axial blunt notch experiments under off-axis angles that were discussed before in this chapter were performed on Glare 3-3/2-0.3 and Glare 4B-3/2-0.3 materials. During all four tests the clamping failed before blunt notch failure occurred. The test matrix and the stresses at which clamping failure occurred translated in fibre direction are given in Table 5-9.
Blunt notch design tool for Glare

Figure 5-46, Predicted failure compared to final failure according to FE calculations for Glare 4A-3/2-0.2 under bi-axial loading.

Figure 5-47, Relative blunt notch strength (compared to the uni-axial situation) versus the percentage of shear load imposed on the uni-axial load for a Glare 4A laminate.
Figure 5-48, Predicted failure compared to final failure according to factorised FE calculations for Glare 4B-3/2-0.2 under bi-axial loading including shear.

Table 5-9, Stresses in fibre direction at clamping failure

<table>
<thead>
<tr>
<th>Material</th>
<th>Off-axis angle</th>
<th>Stresses in fibre direction</th>
<th>( \sigma_x ) [MPa]</th>
<th>( \sigma_y ) [MPa]</th>
<th>( \tau_{xy} ) [MPa]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3-3/2-0.3</td>
<td>15</td>
<td>409.0</td>
<td>227.5</td>
<td>52.4</td>
<td></td>
</tr>
<tr>
<td>Glare 3-3/2-0.3</td>
<td>30</td>
<td>341.4</td>
<td>244.2</td>
<td>84.2</td>
<td></td>
</tr>
<tr>
<td>Glare 4B-3/2-0.3</td>
<td>30</td>
<td>329.3</td>
<td>233.4</td>
<td>83.1</td>
<td></td>
</tr>
<tr>
<td>Glare 4B-3/2-0.3</td>
<td>60</td>
<td>222.8</td>
<td>314.3</td>
<td>79.2</td>
<td></td>
</tr>
</tbody>
</table>

The results for the Glare 4B specimens are depicted in Figure 5-48. It can be observed that the experimental results for Glare 4B tested under an off axis angle of 60 degrees is well below the anticipated result based on the FEM calculations. This, however, must be attributed to the fact that the specimens did fail on clamping failure prior to blunt notch failure. However, the results for Glare 4B tested under an off axis angle of 30 degrees is expected to be close to blunt notch failure.

Unfortunately, only a few points of the blunt notch strength curves of Glare 3, Glare 4A and Glare 4b could be validated with the current available test data.
5.7 Summary and conclusions

The fracture mechanism of uni axially loaded blunt notch specimens has been investigated with C-scan, with a SEM and by chemically removing the outer aluminium layers. In contrast to the expectations of other authors, the results gave no indication that fibre breakage occurred at the notch root prior to failure. During loading, fibre splitting occurs within the matrix at the hole edge at approximately 75% of the failure load. Further load increase causes an increase of matrix cracking. Close to failure, static delaminations are created between the resin and fibre rich layer of the prepreg.

An initial research on blunt notch specimens with a constant hole diameter of 4.8 mm but different widths and tested in L, T and 45° directions revealed that:

- the net blunt notch strength is independent of the width
- the net blunt notch of a single hole is representative for the net blunt notch strength of a row of holes
- filling the hole with a rivet (NAS 1097 AD 6 and $F_{sq} = 24\, \text{kN}$) had no influence on the blunt notch strength

Based on this initial research, an appropriate specimen geometry was defined for an extensive uni-axial blunt notch test program. Many different Glare variants but also aluminium and prepreg specimens have been tested. Some of the specimens were instrumented with strain gauges and a grid in the net section next to the notch. The following results were obtained:

- The aluminium specimens yielded higher net blunt notch results for thicker materials and holes filled with rivets.
- The blunt notch strength of Glare laminates is a function of the MVF. The relation between the blunt notch strength and the MVF depends on the amount of fibres in loading direction
- Filling of the Glare open holes specimens with non-load carrying rivets has no influence on the blunt notch strength. Also the applied squeeze force had no influence. 
- The blunt notch strength decreases significantly if the laminates are loaded under an angle with the fibre directions.
- The scatter in the results is small; more than 97% of the results have an absolute difference with the average results that is less than 3.5%. The scatter shows a normal distribution.
• The strain at failure directly next to the notch root is significantly larger than the ultimate strain of the fibres. This supports the observations within the fracture mechanism research that delamination occurs at the notch root before failure.

The specimens that were bi-axially loaded or in pure shear could not be tested until failure due to limitations in the test set-up. However, the strain and grid measurement have been used to verify FE analysis. It turned out that the grid measurements were very well able to represent the strain at the notch root.

For the prediction of the uni-axial net blunt notch of a specific Glare laminate in one of the principle material directions (L or T direction), a calculation method is derived based on the MVF and the FVF of the fibres in loading direction. These parameters describe accurately the blunt notch strength of a certain Glare lay-up.

For more complicated situations like loading under an angle with the fibre direction or multiple loading directions, the creation of a 3D fracture surface is proposed. An extensive FE analysis has been carried out to determine this surface based on the ultimate strain criterion. However, with the amount of current available test data, only a few points of the blunt notch strength curves of Glare 3, Glare 4A and Glare 4B could be validated. The results are conservative since delamination is not taken into account.
References


[9] Bosker, O.J., Finite Element calculations to predict the blunt notch strength of Glare and verification through test results, B2V-99-34, Delft University of Technology, 1999


6 SHARP NOTCH DESIGN TOOL FOR GLARE

6.1 Introduction

To design a damage tolerant Glare structure, a sharp notch calculation method is necessary to predict the residual strength of a material with through the thickness cracks. This method should provide the user with a tool to predict the residual strength of arbitrary combinations of materials and cracks for design studies but must also be able to create material data that function as input for residual strength calculations of structures. In general, this calculation method should be able to predict the residual strength for an arbitrary

- Glare variant (Glare 2, Glare 3, etc.),
- Lay-up (3/2, 8/7, etc.),
- Aluminium layer thickness (0.2 to 0.5 mm) and
- Loading direction (L, T, 45°, etc.)

Prediction of the residual strength for an arbitrary geometry is only possible if the residual strength is a material property. Therefore, the approach must be to determine a property that is independent of the geometry but that can be modified for realistic variations of the laminate. For this development, knowledge is necessary about the material parameters that play a role in the fracture process. Therefore, § 6.2 describes a fracture mechanism model that is developed within this thesis and that is based upon the available and obtained experimental results. This paragraph also describes a study carried out to investigate the delamination behaviour of standard Glare. § 6.3 presents several investigations that were carried out to determine the influence of the metal layer -, fibre layer – and interface properties on the residual strength. The research in this paragraph is based on the presence of mechanically made cracks only. Therefore, § 6.4 describes a study that investigated the influence of notches that were created by impact, which is more realistic. § 6.5 describes the results of a large residual strength test program that has been carried out on different Glare variants, with variable aluminium layer thicknesses a variable amount of layers.

During this study, a residual strength prediction method for uni-axially loaded flat panels in L and T direction has been developed for Glare laminates that is based on the relative amount of aluminium and fibres in
the laminate. This is described in § 6.6. A summary with conclusions is given in § 6.7.

In a real fuselage, several important aspects have an influence on the fracture behaviour of a crack:

1. Residual stresses introduced in the skin panels when bend to the radius of the structure,
2. Bi-axial loading,
3. The presence of load carrying members like stiffeners, frames and rivets and
4. Bulge out effects (only in case of longitudinal cracks)\(^1\).

All these effects should be included in the crack loading mechanism while the purpose of this work is to provide a parameter to describe the residual strength characteristics of the material.

6.2 Fracture Mechanism

The fracture mechanism of Glare material with a through the thickness crack that is statically loaded to failure is based on the constituent properties and the interfaces between the constituents\(^2\). The aluminium layers within the laminate are responsible for

- yielding of the material at high loads and
- stable crack extension before fracture.

while the elastic S2-glass-fibre-prepreg in the laminate results in

- extensive strain hardening after yielding,
- a high ultimate strength and
- a limited ultimate strain.

At the aluminium-prepreg interface a so-called "resin rich layer" will be established. This is illustrated in Figure 6-1, which shows a SEM picture of a Glare 4 cross section. The thickness of this resin rich layer, the shear strength of the adhesive and the interfaces with the aluminium and "fibre-rich" part of the prepreg play an important role in the possible occurrence of delamination during loading.
This paragraph discusses two fracture processes separately; crack initiation and stable crack extension. These processes are discussed based on a sheet with a through the thickness crack that is uni-axially loaded in the direction perpendicular to the crack.

6.2.1 Crack initiation

The presence of a crack in a material that is loaded perpendicular to the crack edges causes an increased stress intensity in front of the crack tips. Increasing of the applied load results in the creation of local plastic zones in the aluminium layers around the crack tips above a certain load. Because the fibre layers in the laminate remain elastic and are not isotropic like the aluminium layers, shear stresses are introduced between the aluminium and the fibre layers in the resin rich layer. This can be understood by looking at the deformation fields of the fibre layers and aluminium layers separately, see Figure 6-2. Assume that both materials contain a crack of the same size and are loaded to a comparable elongation far away from the crack. Under these conditions, the far field deformations of both materials will be comparable while the local deformation fields in the cracked region are different due to the occurrence of plasticity in the isotropic aluminium
layer while the orthotropic fibre layers remain elastic. Because the deformation field in both materials must be identical within the laminate, this causes shear stresses $\tau_{xz}$ and $\tau_{yz}$ along the interfaces.

Figure 6-2  Illustration of the occurrence of shear stresses within the laminate due to plasticity in the aluminium layers.

In reality, the starter crack is never infinitely sharp. Therefore, the crack needs a location to initiate. This causes crack blunting during loading, which reduces the stress intensity and results in a delayed crack initiation. Since this blunting takes only place in the aluminium layers, extra shear stresses are created on the resin rich layers at the crack tip.

Superposition of these effects (shear stresses due to plasticity and crack blunting) can result in local shear stresses that exceed the critical shear stress of the resin. In that case, delamination occurs between the fibre layer and aluminium layer. This is illustrated qualitatively in Figure 6-3 where the shear stress distribution on the resin rich interface is plotted in loading direction, in front of the crack.
In the event of local delamination, fibres close to and in front of the crack tip do no longer have to follow the large local strains of the aluminium material at the crack tip. Thus, the large local deformation can be spread over a much larger area and decreases the strain in the fibres. Figure 6-4 depicts the strain distribution in loading (y) direction in the aluminium and fibre layers, along a line in y direction directly in front of the crack tip. Without delamination the fibre layers follows the strain distribution of the aluminium layer and fails as soon as their ultimate strain is reached. However, with a sufficient amount of delamination, fibre failure can be postponed and the crack initiation process is determined by failure of the aluminium. It is expected that the strain in the fibres no longer increases in the delaminated area as illustrated in Figure 6-4. Further increase of the strain in this zone can only originate from the neighbouring cracked fibres. The load from these cracked fibres must be transferred over the resin by a shear stress \( \tau_{xy} \). However, the amount of load transfer in this direction is very limited again due to the limited shear strength of the adhesive but now between the fibres. Larger load transfers will result in fibre splitting in the fibre layers.

The two dashed areas in Figure 6-4 are equal in size since the far field deformation of the aluminium and fibre layers is the same.

Based on the previous discussion, it is expected that the occurrence of delamination is delayed and thus the crack initiation stress is lower when:

- an adhesive with a higher shear strength is applied,
• the aluminium material has a higher yield stress and
• the stiffness of the prepreg layers is increased, since these will attract more load and thus increase the yield stress of the laminate

This has been investigated and the results will be discussed in § 6.3.

Figure 6-4, Qualitative illustration of strain distribution in loading direction along in different layers along a line in front of the crack tip and in loading direction.

Additionally, a sharper starter crack decreases the amount of blunting and, therefore, delamination, which also results in a lower crack initiation stress. Besides these mechanisms that deal with delamination, aluminium with a higher fracture toughness is expected to delay crack initiation as well.

6.2.2 Stable crack extension and subsequent fracture

When the crack has extended for some millimetres, the fracture mechanism changes. This can be illustrated with experimental data for the CTOA. Within this study, the generally used CTOA definition is applied where the CTOA is the angle between the physical crack tip and the location where the two crack edges cross a virtual circle with a 1 mm radius and its centre.
at this crack tip, see Figure 6-5. The CTOA increases during loading. The angle at the onset of crack extension is defined as the critical CTOA.

A collection of critical CTOA’s for Glare 3 material, measured in different residual strength tests and for different crack lengths, are plotted in Figure 6-6. It is clear that the critical CTOA decreases after some crack initiation to a more or less constant value. Comparable behaviour is reported for aluminium³.

Because the critical CTOA represents the amount of blunting, these experimental results illustrate that blunting of the crack tip decreases for a larger crack extension. As a consequence, the shear stresses that are the result of blunting reduce. This is illustrated in Figure 6-7.
The plastic zone size grows as a function of the crack extension. Therefore, the boundary with maximum shear stresses due to plasticity in the aluminium layers moves away from the moving crack tip during the stable crack extension process. It is expected that after some crack extension, the critical shear strength of the resin rich interface is no longer reached or occurs in a significantly smaller amount of material at the crack tip, see Figure 6-7. In that case, the ultimate strain of the fibres that is much lower than for the aluminium layers, controls again the fracture process. All research carried out in this study has illustrated that this fracturing of the fibres occurs at loads close to the residual strength of the panel and that they fail in bands, random in front of and close to the crack tip. The fact that stable crack extension occurs while the elastic fibres control the fracture process and carry a much higher load than the aluminium layers is due to the occurrence of plasticity in the aluminium, which redistributes the load away from the crack tip.

Based on these mechanisms, it is expected that the residual strength is a function of the interface strength but depends more on the ultimate strain of the fibres and the yield stress of the aluminium layers. An increase of the ultimate fibre strain will increase the residual strength. The height of the aluminium yield stress has two influences. A lower yield stress advances the creation of static delamination and advances redistribution of the load away from the crack tip. Both effects increase the residual strength. In
terms of the laminate properties, the residual strength is expected to be a function of the strain hardening of the laminate.

The expected existence of static delaminations in standard Glare in front of a crack tip that is statically loaded needs experimental evidence. Unfortunately, it is not possible to see the prepreg during loading since they are covered by the outer aluminium layers. Unloading the specimen and removing the outer aluminium layers for investigation has the disadvantage that unloading of the local plastic zones can result in delamination due to buckling. Fracturing the specimen before etching will not be successful either, since the fibre failure delamination area exceeds the size of any static delamination. The next paragraph discusses a research that was carried out to answer this question.

6.2.3 Delamination research during testing

To investigate the presence of static delamination before fracture, two different test methods have been applied. These are:

- a semi-destructive test method based on the technique of chemical etching and
- a non-destructive test method based on shearography

This paragraph describes both methods and results.

Chemical etching

The idea behind the experiments with chemical etching is to remove small strips of aluminium in front of the crack tip while the specimen is still loaded. In this way, the prepreg layer can be examined locally for static and/or fibre failure delaminations since these delaminations show up lighter compared to surrounding, non-delaminated prepreg. To obtain reliable results, it is important that the etching process does not modify the strain field in front of the crack tip significantly and that the applied load is kept constant during the etching process.

For this purpose, special A.B.-guides have been designed that contain Perspex chambers with an in- and outlet and a pump system through which the etching fluid NaOH can flow. These chambers are designed such that they can be mounted on the specimen in front of the crack tips without leakage of NaOH from the chambers or into the crack, see Figure 6-8. Standard Glare material is covered with a primer that can resist the NaOH solution. To define the area to be etched, the primer is removed and a mask
coating is applied to the surrounding surface. When the specimen is loaded up to the desired crack length or extension, the chambers are filled with the NaOH solution. After the etching process but during loading, the solution is removed and the chambers are opened to allow better examination of the prepreg. Manfredi describes the optimisation of this method in more detail.

A test program has been carried out on several 400 mm wide Glare CCT specimens with $2a_0/W = 1/3$. Two different aluminium alloys were applied in the laminates; either standard 2024-T3 or high strength 7475-T761. The test matrix is given in Table 6-1 while the instrumentation plan is illustrated in Figure 6-9.

**Figure 6-8.** Specially designed Perspex parts for AB guides, containing etching chambers with in- and outlet drains.

**Figure 6-9.** Instrumentation of etching specimens.
The etching was only performed at one side of the crack and at both sides of the specimen. The other side was equipped with strain gages and a grid to detect any influences of the etching process. During the etching process, strain gage measurements were taken to notice any strain field changes in the aluminium around the etched strip. The direction of the strip was always in loading direction.

Table 6-1, Test matrix etching specimens.

<table>
<thead>
<tr>
<th>Material</th>
<th>Description</th>
<th>Destructive etching tests</th>
<th>Shearography</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3 3/2 0.3</td>
<td>2024-T3</td>
<td>2</td>
<td>1</td>
</tr>
<tr>
<td>7475-T761</td>
<td>2</td>
<td>1</td>
<td>( \delta w/\delta y )</td>
</tr>
<tr>
<td>Glare 3 2/1 0.3</td>
<td>2024-T3</td>
<td>1</td>
<td>-</td>
</tr>
<tr>
<td>7475-T761</td>
<td>1</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Glare 2 3/2 0.3</td>
<td>2024-T3</td>
<td>2</td>
<td>-</td>
</tr>
<tr>
<td>7475-T761</td>
<td>2</td>
<td>1</td>
<td>( \delta v/\delta y, \delta w/\delta y )</td>
</tr>
</tbody>
</table>

**Shearography**

Besides these semi-destructive etching tests, it was tried to find a Non Destructive Inspection (NDI) method that was able to confirm the etching results. Three methods were thought to be applicable: the use of an ultrasonic C-scan, Acoustic Emission (AE) analysis or shearography.

Regarding the small delamination sizes in the order of mm’s, the available C-scan at the Delft University with an index step of 1 mm was of no use. Besides the accuracy, the use of this C-scan would ask for a tensile machine within the C-scan immersion tank or complicated solutions to keep the specimen loaded while it is taken out of the tension machine, inspected and returned to the tension machine.

The use of AE analysis has been investigated within this study since it was noticed that static crack growth in Glare produces a typical sound. However, although it was possible to identify a frequency range for fracture in aluminium 2024-T3, the frequency band for the fibres was found to be too wide and it overlapped the aluminium range.

Shearography is a method based on laser interferometry that is able to measure the displacement derivatives of a material surface. Within this
technique, the tested object is illuminated with a coherent monochromatic light (laser), necessary to obtain interference. If the object surface has an average roughness larger than the wavelength of the illuminating laser, the reflected image will have the typical granular appearance that is called speckle, which is illustrated on the computer screen in Figure 6-10. The Speckle effect is a consequence of the way light is scattered from each point of the surface. As each reflected ray will have a random phase change, depending on the height of the surface, there will be interference between the rays scattered by adjacent points. This will result in an interference pattern that describes the position of each different point. When the illuminated surface deforms, also the scattered image changes. By comparing the deformed and undeformed speckles, it is possible to obtain an image with fringes that represent comparable deformation. Such an image based on Moiré interferometry was illustrated in Figure 3-26.

Figure 6-10, Schematic set-up shearography method.
Within shearography, not one but two sheared Speckles are recorded that will interfere and create another interference pattern which intensity distribution describes the relative distance between couples of points of the tested surface. A schematic test set-up is given in Figure 6-10. The use of two sheared images implies that each point of the tested surface is projected on two different points of the image either that two different points will overlap in the image plane. When the object is deformed the relative displacement of these two points induces a relative phase variation of each image point. As a consequence, confronting two images taken in two different states of deformation results in a series of fringes in which every fringe represents a locus of points with the same relative displacement. If the shearing of the images is not excessive and the overlapping points are not too distant, each fringe created by two sheared images represents approximately the same displacement derivative. This is illustrated in Figure 6-11 for a CCT specimen where the fringes represent comparable relative out of plane displacements ($\delta w$) in loading direction ($y$) The whole series of fringes is called shearogram. A more detailed explanation of the method is given by Steinchen$^6$, Schuth$^7$ and Yang$^8$.

![Crack area](image)

Figure 6-11, 2D (left) and 3D (right) illustration of $\delta w/\delta y$ shearogram.

At the Photoelasticity and Holography laboratory of the Universität Gesamthochschule Kassel, three additional residual strength tests were carried out, evaluated with the shearography technique. Special AB guides were used that prevented buckling but allowed illumination and capturing of the area around the crack tip. After every 5kN of load increase, a phase distribution measurement was taken. The system measured the relative out-of-plane deformations, $\delta w/\delta x$ and $\delta w/\delta y$. In one test, also the relative in-
plane deformation $\delta v/\delta y$ was measured. The test matrix is given in Table 6-1 while Figure 6-12 shows a picture of the test setup.

*Figure 6-12, Shearography test set-up*

*Etching tests results*

The first test was a Glare 3 3/2 0.3 CCT specimen with 2024-T3 aluminium layers. A strip at 1 mm in front of the initial crack tip was etched when the gross stress was approximately 200 MPa. At that time, the stable crack extension in the outer aluminium layer was 2.1 mm. Figure 6-13a shows the image after etching. The initial saw cut is located in the upper right corner. The lower-right corner shows some un-etched aluminium with a thick black border. All other locations in the image show the prepreg. The crack in the outer aluminium layer before the etching process is indicated.

In the wake of the crack extension some fibres are visible since a part of the resin rich layer is removed with the etching process. Some fibre bundles at the initial crack tip are clearly broken but the majority of the visible fibres are still intact as will be illustrated later. Delamination often takes place at the interface between the resin rich- and the fibre rich layer within the prepreg (§ 6.3.3). The removed resin rich layer in Figure 6-13 a) represents therefore an area of static delamination. The total region of static delamination is clearly visible by the brighter colour compared to the non-
delaminated areas and is advanced in front of the crack tip in the fibre layer and the removed outer aluminium layer.

Figure 6-13, Images after etching under constant load, Glare 3 3/2 0.3.

Figure 6-13 b) shows the etched part after an increase of the gross stress to 240 MPa. During this load increase, some fibre bundles broke and created fibre failure delaminations. It is clear that the size of these delaminations exceeds the earlier created static delaminations. This photo illustrates that some of the fibre bundles that are visible in Figure 6-13 a) were still intact.

Figure 6-14, Images of etched Glare 2 3/2 0.3.
Figure 6-14 shows images of the same location but now of a Glare 2 3/2 0.3 specimen with 2024-T3 aluminium layers. The saw cuts were located at the right side of both images in the centre. Figure 6-14 a), taken at a gross stress of 229 MPa and a crack extension in the outer aluminium layer of 1.25 mm, shows again the existence of some static delamination. This static delamination runs ahead of the crack tip. Some fibres bundles at the right side seem to be broken. During further load increase to 243 MPa, fibre failure occurred. Figure 6-14 b) clearly shows the size difference between static- and fibre failure delamination. Increasing the load further illustrated pulling of the broken fibres out of the lower specimen half. The static delamination length in the loading direction was measured to be 4 mm while the average fibre failure delamination in this direction was 11.4 mm.

During the tests, it was noticed that some fibre bundles failed further in front of the crack tip while bundles closer to the crack tip remained intact. This suggests that fibre breakage is determined by the presence of imperfections and that the ultimate fibre strain of 4.5% is only an average failure strain.

The other two specimens of Glare 2 3/2 0.3 and Glare 3 3/2 0.3 based on 2024-T3 were etched at higher loads and more crack extension. These tests were not illustrative since the fibres at the etching location were already broken and, therefore, only fibre failure delamination was observed.
observed more often that two cracks extend from the corners of the blunted saw cut tip in the outer aluminium layers. After some extension, one crack becomes dominant and the cracks stops extending. This seems not to be the case for the prepreg layer perpendicular to the loading direction. As a result this fibre layer shears off at several locations, see Figure 6-15. As expected, no signs of delaminations were observed at this side of the prepreg.

During the tests, the crack length in the outer aluminium layers and in the neighbouring prepreg layers have been measured for comparable load situations. It was found that the crack lengths in the aluminium layers are initially longer. However, after some crack extension, the crack in the prepreg layers became longer. This supports the theoretical fracture model discussed in § 6.2.2.

Figure 6-16 and Figure 6-17 show the strain gauge readings for a Glare 3 3/2 0.3 specimen with 2024-T3 aluminium layers before and during the etching process. The strain gage results should answer the question whether the etching process creates major local and/or global strain field disturbances. Etching occurred at a relatively low gross stress of 130 MPa.

Figure 6-16, Strain gauge readings in x-direction before and during etching.

Strain in x-direction, Figure 6-16
At location 1 and 3, compressive strains were measured in x-direction while small tensile strains were measured at locations 2 and 4, further in front of the crack tip. During the etching process, the strain in x direction at location 1 decreased slightly during etching. The comparable strain at location 3
remained constant and was clearly higher (-0.2% at location 1 versus -0.35% at location 3). This difference was built up from the very beginning of loading the specimen, before the etching process, and is therefore expected to be due to local buckling. The development of the strains in x direction at location 2 and 4 is almost identical. During etching, the strain at location 2 returned to zero and remained constant at location 4. This reduction can be explained since the etched strip blocks the stress trajectories in the outer aluminium layer in x-direction, in front of the crack tip. Because of the very small strains at this location, this is not expected to influence the fracture process.

Figure 6-17, Strain gauge readings in y-direction before and during etching.

Strain in y-direction, Figure 6-17
The strains in y-direction are of course larger at location 1 and 3 close to the crack tips than at location 2 and 4 for the same gross stress. Before etching, the strain development at both sides of the crack is comparable. During etching, the strain at location 1 decreased slightly. It seems that the etching created some load redistribution away from the crack since the strain at location 2 increased during etching. This seems to be a very local process since the strain gauge readings at location 3 and 4, at the other side of the crack remained constant during etching.

Shearography test results
The evaluation of the obtained shearograms turned out to be complicated since the results depend largely on the settings of the system. Unfortunately, the results based on Glare 3 with 2024-T3 had to be
discarded. The shearograms present derivatives of the displacement field as fringes. For example in the case of an out of plane displacement \( w \) in along a line in \( x \) direction, like in Figure 6-18a, the 3D shearography representation looks like Figure 6-18b. When the out of plane displacement would be in the other direction, the shearography representation in Figure 6-18b should first show a valley followed by a mountain. This is important information for the evaluation, since delaminations are expected to create out of plane displacements to the outside direction while plasticity causes contraction of the aluminium layers and is expected to create out of plane displacements of the outside layer towards the centre line of the laminate.

Before shearography was applied to analyse the strains in loaded CCT specimens, some specimens that were previously fatigue loaded and contained static delamination have been analysed. The shearograms of this area showed some out of plane deformation in the delaminated area but it was concluded from this evaluation that the current shearography system could only give qualitative information. Many of the observed changes in the shearogram could be explained with the knowledge obtained with the etching experiments.

Figure 6-19 shows \( \frac{\delta v}{\delta y} \) plots of the Glare 2 3/2 0.3 specimen with al 7475-T61 layers. Only the round part in the centre is of interest, the A.B. guides covered the other parts, see Figure 6-12. It shows the stress concentration around the crack tip, including an indication of the expected butterfly shape.
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In the out of plane plots of $\delta w/\delta y$ for the same test illustrated in Figure 6-20 a) and b), no clear static delamination could be observed. Up to 275 MPa, the relative derivative of the out of plane deformation was negative at the crack tip as indicated by the 3D plot in Figure 6-20b). This represents plastic deformation as explained before. Ahead of the crack, the derivative becomes positive, which indicates the presence of static delamination or first fibre delamination due to the first fibre bundle that has been broken. At 279 MPa, the out of plane deformation of the hole area became suddenly positive and much larger, illustrating the occurrence of fibre failure delamination, see Figure 6-20 c) and d). These stresses at which fibre failure delamination was recognized with shearography estimate with the stress values indicated for first fibre failure during the etching tests on comparable material. In front of the fibre failure delamination of Figure 6-20
d) an area with negative $\delta w/\delta y$ derivatives can be recognized. This illustrates the presence of plasticity in front of the delaminated area.

![Plasticity](image1.png)

**a)** 275 MPa  

![3D Plot](image2.png)

**b)** 3D plot of $\delta w/\delta y$ at 275 MPa

![Plasticity and Fibre Failure Delamination](image3.png)

**c)** 279 MPa  

![3D Plot](image4.png)

**d)** 3D plot of $\delta w/\delta y$ at 279 MPa

Figure 6-20, $\delta w/\delta y$ shearograms for Glare 2 3/2 0.3 (7475-T761).

The difference between plasticity and fibre failure delamination is even better illustrated in Figure 6-21 for the glare 3 3/2 0.3 specimen with 7475-T671 aluminium. In the Glare 3 3/2 0.3 with 2024-T3 aluminium, some delamination could be observed clearly below any fibre failure. Thus this had to be static delamination.

**Summary**
Both inspection methods confirmed the creation of static delamination before dynamic delamination although the shearography method gave poor results. The etching method showed that static delamination advanced before the crack tip. Fracture of the fibre layers occurred in bundles, randomly in the close front of the crack tip. It was also observed that the crack extensions in the aluminium layers are initially larger than in the
prepreg layers. However, after some crack extension and increased load, the cracks in the prepreg layers become larger.

![Image of shearograms](image)

a) plasticity (180 MPa)

b) fibre failure delamination (194 MPa)

Figure 6-21, $\delta w/\delta y$ shearograms for Glare 3 3/2 0.3 (7475-T761).

### 6.3 Influence of constituent properties on residual strength

This paragraph discusses an extensive research carried out to investigate the influence of the different constituents within Glare on its residual strength. § 6.3.1 discusses the influence of several aluminium properties on the residual strength. The influence of a different type of fibre is discussed in § 6.3.2., while the influence of the pre-treatment of the aluminium layers on the bond strength and consequently on the residual strength is discussed in § 6.3.3.

Every laminate tested contained only one kind of aluminium alloy, one kind of fibre or one kind of pre-treatment. All laminates were made at the Delft University of Technology, under the same circumstances and by using the same equipment. To obtain a correct relation between the laminate properties and the residual strength behaviour, both the tensile test specimens and residual strength specimens were cut from the same sheet.
In the investigation of the aluminium (pre-treatment) properties, the prepreg in all specimens originated from the same production batch. The same concept was followed for investigating a different fibre; in this case the aluminium alloy used in the sheets originated from the same batch. Consequently, the experimental results of the tested laminates can be directly related to the influence of the investigated constituent. Every laminate produced was checked for debonds and inserts with a C-scan. No defects were determined different from those that were made on purpose. The thickness of the laminates was measured with a micrometer. Several samples were taken to investigate the cross sections with an electronic microscope, allowing thickness measurements of separate layers.

All uni-axial tensile tests that have been carried out (§ 6.3.1 and § 6.3.2) are in accordance with ASTM standard D3033-76. The specimen geometry has been slightly modified as described by Mattoush and is illustrated in Figure 6-22. The results are engineering stress-strain curves up to failure of the specimens.

![Uniaxial tensile specimen](image)

Figure 6-22, Uniaxial tensile specimen. All dimensions are in mm

The residual strength tests were carried out on 400mm wide, unstiffened flat CCT specimens according to ASTM E561. In the research described in § 6.3.1 and § 6.3.2, no fatigue pre-cracking was applied to avoid the creation of delamination. During the test, the crack length was measured either with a Potential Drop (PD) method or with a CCD camera connected to a microscope.

### 6.3.1 Aluminium characteristics

**Materials**

Several Glare 3 3/2 0.3 laminates were tested with the following aluminium alloys:
the high fracture toughness alloys 2024-T3 that is used in standard Glare 3 and 2024-T81, which is the same alloy but with a different heat treatment (naturally aged for T3 vs. artificially aged for T81). As a consequence, the yield stress and ultimate strength of 2024-T81 is higher,
the high strength alloys 7075-T6 and 7475-T761 (7475-T761 has a higher fracture toughness than 7075-T6)
the weldable alloy 6013-T6.

For every aluminium type, two sheets were laminated of $530 \times 660$ mm. These dimensions were determined by the maximum available sizes of the pre-treatment baths. The test matrix is given in Table 6-2.

Table 6-2, Test matrix aluminium research

<table>
<thead>
<tr>
<th>Material</th>
<th>Tensile tests</th>
<th>Residual strength tests</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3 3/2 0.3 (2024-T3)</td>
<td>6 spec.</td>
<td>2 spec.</td>
</tr>
<tr>
<td>Glare 3 3/2 0.3 (2024-T81)</td>
<td>6 spec.</td>
<td>2 spec.</td>
</tr>
<tr>
<td>Glare 3 3/2 0.3 (7075-T6)</td>
<td>6 spec.</td>
<td>2 spec.</td>
</tr>
<tr>
<td>Glare 3 3/2 0.3 (7475-T761)</td>
<td>6 spec.</td>
<td>2 spec.</td>
</tr>
<tr>
<td>Glare 3 3/2 0.3 (6013-T6)</td>
<td>6 spec.</td>
<td>2 spec.</td>
</tr>
</tbody>
</table>

**Tensile test results**

Figure 6-23 illustrates the experimental stress-strain curves for the different Glare 3 materials in L-direction. The bilinear behaviour due to the elastic fibre and elastic-plastic aluminium behaviour is clearly visible. Fracture is determined by fibre failure at a strain close to 4.2%.

Table 6-3 contains the characterising tensile test results from Figure 6-23. The laminate strengths at a strain of 4.2% are listed here instead of the maximum strengths. This value is reached by all specimens and allows a better comparison than the ultimate strength for which the values are based on local fibre failure.
Sharp notch design tool for Glare

![Stress strain curves for different aluminium alloys](image)

Figure 6-23, Uni-axial tensile results of Glare specimens with different Al-alloys.

Table 6-3, Test results aluminium research.

<table>
<thead>
<tr>
<th>Material</th>
<th>Residual strength tests (L-T)</th>
<th>Tensile tests (L)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3 3/2 0.3 L</td>
<td></td>
<td></td>
</tr>
<tr>
<td>2024-T3, sheet 1</td>
<td>123.1</td>
<td>239.5</td>
</tr>
<tr>
<td>sheet 2</td>
<td>118.9</td>
<td>240.2</td>
</tr>
<tr>
<td>2024-T81, sheet 1</td>
<td>100.3</td>
<td>226.4</td>
</tr>
<tr>
<td>sheet 2</td>
<td>95.4</td>
<td>229.1</td>
</tr>
<tr>
<td>7075-T6, sheet 1</td>
<td>97.8</td>
<td>222.4</td>
</tr>
<tr>
<td>sheet 2</td>
<td>90.4</td>
<td>218.9</td>
</tr>
<tr>
<td>7475-T761, sheet 1</td>
<td>148.8</td>
<td>230</td>
</tr>
<tr>
<td>sheet 2</td>
<td>142.3</td>
<td>229.1</td>
</tr>
<tr>
<td>6013-T6, sheet 1</td>
<td>141.0</td>
<td>230.4</td>
</tr>
<tr>
<td>sheet 2</td>
<td>137.6</td>
<td>227.5</td>
</tr>
</tbody>
</table>

Residual strength results

The residual strength test were carried out on 400 mm wide CCT specimens with 2a₀ = 100 mm. All specimens contained a crack in T direction and were tested in L direction. The results for all tested laminates vary 10% between 219 and 240 MPa, see Table 6-3. Although only two
tests are carried out for every Al alloy, the scatter in the residual strength results is small and varies between 0.3 and 1.6%.

The starter cracks were sharpened with a special saw and not additionally fatigue pre-cracked. As a consequence, large crack blunting occurred during loading which reduced the stress intensity at the crack tip and delayed crack initiation. After initiation, the blunting is reduced significantly. Figure 6-24 illustrates the measured crack extension \( \Delta a \), as a function of the gross stress. \( \Delta a \) is an average of the measured crack extensions at both sides of the crack and sheet. If we assume that the crack initiation process takes place during the first 2 mm, the stable crack extension process takes place at loads that are close to fracture.

In Figure 6-25 to Figure 6-27, the residual strength and the crack initiation stresses are plotted as a function of the yield stress, the laminate stress at 4.2% strain and the strain hardening respectively. Strain hardening is defined here as the difference between the 4.2% laminate stress and the yield stress. It was also possible to plot the \( K_{1c} \) values instead of the residual strength values. However, as illustrated in Figure 6-24, it is rather difficult to determine \( a_c \) properly, which would result in large scatter for \( K_{1c} \).

Looking at the stress results for crack initiation, the expectation discussed in § 6.2.1 that the application of an aluminium alloy in Glare with a lower yield stress delays crack initiation due to the earlier creation of delamination.
areas, agrees well with the results presented in Figure 6-25. It is found remarkable that this trend could be observed even though crack initiation is also largely depending on the initial crack tip shape and the existence of small defects along the tip due to the tooling and craftsmanship.

Figure 6-25, Residual strength results (L-T) and crack initiation stress results versus laminate yield stress in L direction.

Figure 6-26, Residual strength results (L-T) and crack initiation stress results versus laminate stress in L direction at 4.2% strain.
No clear relation was found between the use of higher strength aluminium and the crack initiation stresses, see Figure 6-26, nor between the strain hardening of the laminate and the crack initiation stresses, see Figure 6-27. Such relations were also not foreseen.

![Figure 6-27](image_url)

Figure 6-27, Residual strength results (L-T) and crack initiation stress results versus strain hardening in L-direction.

The expectations discussed in § 6.2.2 that the residual strength should increase for a lower yield stress and a larger amount of strain hardening estimates very well with the experimental results.

In this first test program, the fibre properties were kept constant. From a static point of view it could be expected that stronger aluminium should result in a higher residual strength. However, the opposite trend is shown in Figure 6-26, also for the crack initiation stresses, which supports the opinion that the fracture toughness of the aluminium plays a role in both processes and needs to be investigated.

One additional comment must be made about the relation between the residual strength and the strain hardening. All results presented here are for panels loaded in rolling (L) direction. In general, the strain hardening of an aluminium alloy in L direction is smaller than in T direction because the rolling process pre-strains the material in that direction, which results in a significant increase of the yield stress. Because the ultimate strength is only slightly higher in L direction, the strain hardening in T direction is larger than in L direction. Based on the found relation between strain hardening and the
residual strength, the residual strength in T-L direction should be larger than in L-T material for the same material. However, this is in contradiction with the available experimental results\textsuperscript{13}. This again supports the opinion that the fracture toughness plays a role since this one is higher in L-T than in T-L direction for 2024-T3.

Unfortunately, the fracture toughness of the aluminium alloys was not tested within this program. Because $K_c$ is very much dependent on the thickness of the aluminium sheet, specific data are necessary for 0.3 mm thick sheets. These data were not found in the literature. It could have been possible to bond several 0.3 mm aluminium sheets together and determine these values, or even better to obtain a representative $K_{fr}$-curve, with residual strength tests on CCT specimens. However, as explained before, the size of the pre-treatment baths was limited and too small to obtain representative values without the occurrence of NSY.

The generated $K_{fr}$-curves for all the different laminates investigated will be discussed later in § 6.6.

### 6.3.2 Fibre-prepreg characteristics

The test program to investigate the influence of the fibre properties on the residual strength of Glare is much more limited; only one new experimental fibre is investigated\textsuperscript{14,15}. This fibre, called "Fx", has good compression properties, an extremely high modulus of elasticity, excellent durability properties but a rather low failure strain. To extend the analysis some literature data are added for the fibre metal laminate ARALL 2, a laminate comparable to Glare 2 but based on Aramid fibres\textsuperscript{16}. Table 6-4 lists the different fibre properties.

<table>
<thead>
<tr>
<th>Material</th>
<th>$E$ [GPa]</th>
<th>$\sigma_u$ [MPa]</th>
<th>$\epsilon$ [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Al 2024-T3</td>
<td>72.5</td>
<td>455</td>
<td>19</td>
</tr>
<tr>
<td>S2 fibre</td>
<td>86</td>
<td>4600</td>
<td>5.3</td>
</tr>
<tr>
<td>Fibre x</td>
<td>290</td>
<td>3400</td>
<td>1.5</td>
</tr>
<tr>
<td>Aramid fibre</td>
<td>124</td>
<td>2800</td>
<td>2.5</td>
</tr>
</tbody>
</table>

Unfortunately, the amount of Fx fibres was very limited and only two prepreg layers were available per specimen size. To keep the laminate symmetric, Glare 2 2/1 0.3 sheets were produced. The adhesive used for the Fx fibre was FM 94-U from Cytec, the same as used in standard Glare. The fibre fraction of the Fx fibres and the S2 glass fibres in the prepreg was
comparable and equal to 60%. The aramid prepreg has a lower fibre fraction of 49%.

**Tensile results**

Figure 6-28 shows the experimental tensile results for Glare based both on the Fx fibre and the S2-glass fibre and the calculated results for Arall 2, based on the data of the Young’s modulus, the yield stress and the ultimate strain of the constituents. This method uses the uni-axial laminate theory. The results are also noted in this figure.

![](Figure_6-28_Ui.png)

Figure 6-28, Uni-axial tensile results of Glare with different fibres.

It is clear that the stiffer Fx fibre increases the stiffness of the laminate and also the yield stress. The increased stiffness of Aramid fibre compared to the S2 fibre is counteracted by the lower fibre volume fraction in the prepreg. Therefore, the prepreg stiffnesses are comparable. The ultimate strength is clearly dominated by the failure strains of the different fibres.

**Residual strength results**

Due to the limited amount of Fx fibres, the CCT specimen sizes were very limited. For that reason it is not directly possible to generate $K_I$-curves or fracture toughness values with these results. However, the results allow a comparison between specimens with the same dimensions and tested under the same circumstances. Figure 6-29 shows the gross stress as a function of the physical crack extension. It is clear that Glare based on the stiffer Fx fibre shows much earlier crack initiation. The low failure strain of
the Fx fibre results in the much lower residual strength compared to standard Glare. Both trends were expected and discussed in § 6.2.2.

Schwarmann\textsuperscript{18} has reported the residual strength results of 400 mm wide Arall 2 3/2 0.3 CCT panels with a crack length of 100 mm. The residual strength of this panel was 217 MPa while a comparable Glare 2 3/2 0.3 specimen gives a residual strength of 350 MPa. The only differences between these panels are the different failure strain and strength of the fibres. These results confirm the expectations that a lower fibre ultimate strain and strength reduces the residual strength. Differences in delamination behaviour between these laminates have not been investigated.

6.3.3 \textit{Influence of interface aluminium - prepreg}

Delamination causes a loss of integrity in the structure, especially when loaded in compression. Glare makes use of an up to date epoxy adhesive as the bonding constituent in the material. Therefore, significant delamination will occur only as the result of accidental damage that is always visible and appear equal or larger than the delaminated area\textsuperscript{19}. Furthermore, the effect of moisture absorption is limited to the edges of the material and does not result in delamination. In case of fatigue loading, limited delamination occurs at the crack edges.
Delamination in a FML occurs within the intermediate prepreg layer that has to transfer load between the two neighbouring aluminium layers and the fibres within the prepreg. Various fracture or delamination paths are possible, see Figure 6-30:

1. Failure between the resin rich layer and the adherent,
2. Cohesive failure in the resin rich layer,
3. Failure between the resin rich layer and the fibre rich layer, and
4. Failure within the fibre rich layer, either within one prepreg layer or at the boundary between two prepreg layers with different fibre orientation (4')

Figure 6-30, Possible delamination paths.

The fracture path depends very much on the loading situation. Mode I loading creates tension loads in the adhesive while mode II loading creates shear stresses.

To investigate the effect of delamination on the residual strength, the bond strength between the prepreg and aluminium layer has been influenced in this study by using different pre-treatments. The normal pre-treatment of the aluminium layers before laminating them into Glare laminates is:

1. *Hand solvent cleaning*: the aluminium is wiped with paper soaked in an organic solvent.
2. **Degreasing**: the aluminium is immersed in an alkaline solution between 70 and 80°C for 30 minutes and additionally rinsed for 5 minutes in cold running tap water.

3. **Sulphuric acid dichromate etching**: The aluminium is immersed in sulphuric acid dichromate solution for 20 minutes. The temperature is between 60 and 65°C.

4. **Acid anodising**: the aluminium is immersed in the anodising solution for 40 minutes at a temperature of 40° ± 2°C while the aluminium is the anode.

5. **Priming**: the aluminium is primed within 3 hours after the anodising process with BR 127, an epoxy based corrosion inhibiting primer.

A total of 10 Glare 2 3/2 0.3 CCT specimens were tested\(^{20}\), one standard specimen and three series of specimens with a different pre-treatment of the aluminium layers. The test matrix is given in Table 6-5. One specimen type contained aluminium layers that were degreased by hand only (step 1). The second type was decreased and etched (step 1 to 3). The third series received the standard pre-treatment (step 1 to 6) but spraying mono-coat on the primed aluminium surfaces before laminating created an elliptical delamination. The process is illustrated in Figure 6-31.

![Figure 6-31, Creation of an elliptical delamination in a CCT specimen.](image)

<table>
<thead>
<tr>
<th>Material: Glare 2 3/2 0.3 L-T CCT-specimens</th>
</tr>
</thead>
</table>

<table>
<thead>
<tr>
<th>Pre-treatment</th>
<th>Lap-shear strength [MPa]</th>
<th>Peel strength [MPa]</th>
<th>nr</th>
<th>Fatigue testing da/dn [mm/kcycle]</th>
<th>Residual strength [MPa]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Step 1</td>
<td>28.9</td>
<td>1.4</td>
<td>1</td>
<td>0.18</td>
<td>372</td>
</tr>
<tr>
<td>Step 1 to 3</td>
<td>31.5</td>
<td>8.6</td>
<td>3</td>
<td>0.04</td>
<td>349</td>
</tr>
<tr>
<td>Standard</td>
<td>37(^{11})</td>
<td>11.5</td>
<td>3</td>
<td>0.03</td>
<td>330</td>
</tr>
<tr>
<td>Standard + artificial delamination</td>
<td>-</td>
<td>-</td>
<td>3</td>
<td>17</td>
<td>421</td>
</tr>
</tbody>
</table>
Before carrying out the residual strength experiments, the CCT specimens were first fatigue loaded to investigate the influence of the pre-treatment on the fatigue crack growth. The specimens contained an initial crack length \((2a_0)\) of 20 mm and were fatigue tested to a total crack length of 100 mm. The fatigue loading consisted of a CA spectrum with \(\sigma_{\text{max}} = 120\) MPa, \(R=0.05\) and \(f = 4-6\) Hz. After fatigue loading, the delaminated area was inspected with an ultrasonic device (C-scan) and the intact fibres were cut to create a through crack of 100 mm.

To quantify the bond strength of the different pre-treatments, additional lap-shear and peel tests have been carried out illustrated in Figure 6-32. The 2024-T3 sheets with different pre-treatments were bonded together with the adhesive AF 163 2K. The lap shear test was performed according to the ASTM standard D-1002 and determines the shear strength of the adhesive joint. However, the single lap-joint with the short overlap length \(L\) gives rise to peel stresses as well due to secondary bending effects and causes plastic deformation of the adherent at relatively low loads. The floating roller peel test induces more severe loads than occurring in the laminate situation. Nevertheless, it is a good indicator of the overall quality of the bonding process. An acceptable result is cohesive failure within the adhesive, while adhesive failure at the adhesive/thick adherent interface usually indicates poor surface pre-treatment.

**Lap shear and peel strength results**
Both the lap shear tests and the roller peel tests indicated an increase of the bond strength for a standard pre-treatment compared to only degreasing and etching and to degreasing only. The peel tests on the aluminium degreased by hand and the etched aluminium illustrated adhesive failure while the standard pre-treated aluminium showed cohesive failure within the adhesive. This explains the peel strength results in Table 6-5.

**Fatigue tests results**
Figure 6-33 illustrates the measured crack growth rate for the different specimens. The specimens with degreased and etched aluminium layers show the same crack growth rate as the standard specimen. Because of the slow crack growth, the standard panel was only tested up to a total crack length of 50 mm. After a crack extension of about 7 mm and approximately 60 kcycles, the crack extension per cycle \((da/dn)\) in this specimen became nearly constant.
The fatigue life of the Glare variant with aluminium sheets degreased by hand is clearly shorter than the life of the standard material. Because of the lower bond strength, the adhesive in the prepreg is less capable to transfer high shear forces from the aluminium layers into the fibres and consequently delamination occurs. Therefore, the crack bridging effectiveness is decreased and the crack growth increases.
A very high crack growth rate is visible in the Glare with the artificial delamination created by an ellipse of monocoat. All the fibres in the material remained intact, including those that bridge the crack. Because of the long delaminated length, they could extend freely, causing a dramatic loss of crack bridging effectiveness. It can be concluded that there is a clear relation between the bond strength and the crack growth rate under fatigue loading.

Figure 6-34 to Figure 6-36 illustrate the C-scan images before and after fatigue loading for the different specimens. The darker areas represent the fatigue delaminated regions were the fibres are still intact. The smallest amount of delamination occurs in the material with the degreased and etched aluminium layers, followed by the specimens with degreased aluminium layers (Figure 6-35) and the specimens with monocoat (Figure 6-36).

The C-scan results of one of the standard Glare sheets with an artificial delamination created by an ellipse of monocoat are illustrated in Figure 6-36. The surrounding material was evaluated to have a good bonding quality. The area treated with monocoat would be rejected according to the quality specifications although the colour difference with the surrounding material appears to be small in the scan, see Figure 6-36 a. This small difference is due to the fact that the layers within the ellipse are still sticking to each other. The dark spot around the hole indicates full delamination that
occurred during countersinking the hole for the COD instrumentation. Due to the fatigue loading the delamination above the crack edges in the aluminium layers extends towards the edge of the monocoat area.

Figure 6-34, C-scan image of Glare 2/3/2 0.3 specimen with degreased and etched aluminium layers.

Figure 6-35, C-scan image of Glare 2/3/2 0.3 specimen with degreased aluminium layers.
Figure 6-37 illustrates the measured stress versus crack extension curve for all the different specimens. The residual strength results are listed in Table 6-5. Because of the relatively good pre-treatment of the specimens with degreased and etched aluminium layers the behaviour of these specimens is nearly the same as the behaviour of standard Glare 2 3/2 0.3. The behaviour of the material with aluminium layers that were only degreased differs from standard Glare 2 3/2 0.3. Both tested specimens show higher maximum residual strength and a slightly longer crack extension than the standard material. The bonding strength is less than in standard Glare. Therefore the prepreg can easily separate from the aluminium layers, and the fibres can elongate more than in the standard Glare.

The crack initiation stress for the specimen with artificial delamination is significantly lower than the stress at which standard material starts to crack. This is due to the fact that the fibre layers at the crack tip are less effective in supporting the aluminium layers because of the large delaminated area. As a result, the crack bridging of the fibres is less effective and the
aluminium at the crack tip is severely loaded. After a crack extension of $\Delta \theta_{\text{phys}} = 10 \text{ mm}$, a constant ratio between the applied load and the physical crack extension in the outer aluminium layers is observed. This is due to the fact that all fibres between the saw cut tip and the crack tip in the aluminium layers remain intact and thus more fibres are bridging the crack for longer crack extension.

When the crack reaches the edge of the artificial delamination, the ratio between the applied stress and $\Delta \theta_{\text{phys}}$ increases significantly due to the better support of the aluminium layers by the glass layers. Since all the fibres in the wake of the crack are still intact and the delaminated area is rather large, the residual strength of this panel with artificial damage is larger compared to the other specimens, even though the amount of cracked aluminium layers is larger.

The fatigue cycling before the residual strength testing is not expected to have had any effect since the fibres in all specimens were cut over the fatigue delaminated areas.

![Gross stress version crack extension, Glare 2 3/2 0.3 specimens with different pre-treatment aluminium layers.](image)

Figure 6-37, Gross stress version crack extension, Glare 2 3/2 0.3 specimens with different pre-treatment aluminium layers.

After the residual strength tests, the outer aluminium layers of all specimens were removed by an etching process, giving a clear view of the prepreg layer and showing static delamination due to the fatigue tests and fibre failure delamination due to residual strength tests. Two etched specimens are shown; Figure 6-38 shows the degreased and etched specimen while Figure 6-39 shows the specimen with elliptic delamination.
Chapter 6

Figure 6-38 clearly show the difference and size of the fibre failure delamination compared to the fatigue delamination. The size of the fibre failure delamination is clearly larger than for standard material.

Figure 6-38, Glare made with degreased and etched aluminium layers after etching away the outer aluminium layers.

Figure 6-39, Standard Glare with an elliptic delamination after etching away the outer aluminium layers.

Figure 6-39 for the standard material with artificial elliptic delamination illustrates a lot of delamination and visible fibres. The elliptical area above
and below the crack is easily recognised and is completely delaminated. The delamination did not spread any further than this edge.

At the edge of the starter crack fibre splitting occurred, which stopped the crack extension in the fibre layers perpendicular to the loading direction. The result of this mechanism was that the fibres were loaded in tension as if hardly any crack was present. The loading created a very large static delamination over the edge of the original elliptic delamination. At the left side, this is confirmed since the fibre layer is still intact. At the right side the fibre layer fractured and fibre failure delamination is visible.

Poor pre-treatment gives a higher residual strength due to the fact that fibres can elongate over a longer length because the adhesive cannot resist the high shear stresses at the edge of the delamination. In this way, more energy can be stored in the fibres before fracture occurs.

6.4 Residual strength after impact

Through the thickness cracks in Glare will only occur due to impact or in case of an extremely large fatigue crack where sudden unstable fracture of the crack bridging fibre layers might occur as will be discussed in chapter 9.

In case of impact damage, the visible damage for example due to deformation is always larger than the delaminated area. However, it is possible that the actual through the thickness damage is smaller than the statically delaminated area as illustrated in Figure 6-40. As discussed in § 6.3.3, this can have a favourable effect on the residual strength.
Chapter 3 discussed the results of impact tests on CCT specimens with a sharp impactor. These impacted specimens did not show a larger delaminated area than the through the thickness crack and did not result in a residual strength increase compared to a saw cut.

Within this study, tests were carried out for which the impact damage as applied with a round shaped impactor while the specimens carried a load, representative for the fuselage pressure at cruising altitude. Because of this pre-load it is expected that fracture of the fibres due to impact can cause static delamination around the impacted area. The specimen dimensions and the geometry of the impactor are given in Figure 6-41, the test matrix is given in Table 6-6. The impact energy was kept constant and was approximately 97 Joule.

![Figure 6-41, Impact specimen and impactor geometries.](image)

A clear difference was observed between the impact shapes. A large, round shaped opened area can be seen in impacted aluminium, see Figure 6-42a. The aluminium is slanted to all sides of the impacted area. The impact on Glare 2 3/2 0.2 shows a small round opening with slanted aluminium layers and a larger fracture in the direction of the fibres, see Figure 6-42b. The impact in the quasi isotropic material Glare 3 caused a round hole, see Figure 6-42c and because of the cross ply lay up with 67% of the fibres in
loading direction, the impacted area in Glare 4 is a mix between Glare 2 and Glare 3.

Table 6-6, Test matrix impact specimens and residual strength results

<table>
<thead>
<tr>
<th>Material</th>
<th>CCT [MPa]</th>
<th>Impact with pre stress [MPa]</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>0</td>
<td>60</td>
</tr>
<tr>
<td>Al 2024-T3</td>
<td>292</td>
<td>266</td>
</tr>
<tr>
<td>Glare 2 3/2 0.2</td>
<td>412</td>
<td>564</td>
</tr>
<tr>
<td>Glare 2 3/2 0.3</td>
<td>381</td>
<td>561</td>
</tr>
<tr>
<td>Glare 3 3/2 0.2</td>
<td>245</td>
<td>294</td>
</tr>
<tr>
<td>Glare 3 3/2 0.3</td>
<td>261</td>
<td>324</td>
</tr>
<tr>
<td>Glare 4 2/1 0.2</td>
<td>281</td>
<td>389</td>
</tr>
</tbody>
</table>

1) $2a_0 = 30$ mm

Figure 6-42, Images of specimens after impact.
Figure 6-43 shows the measured length of the impact damage perpendicular to the loading direction. No clear influence of the pre-stress can be observed on the damage length; the damage lengths in the specimens with a 120 MPa pre-stress are in general slightly larger than in the specimens without pre-stress but the 60 MPa pre-stressed specimens show no relation. The damage length seems to show a relation with the amount of fibres in loading direction; the larger this amount, the smaller the
damage length. Finally, a larger MVF for a certain Glare type results in smaller damage lengths. However, this can also be due to the thickness since the specimens with a higher MVF are also thicker.

Figure 6-44 shows the residual strength values of the impact tests together with the results of CCT specimens with a crack length that was comparable with the damage length. It is clear that the residual strength of the impact specimens is slightly larger, which is due to the more blunt damage. No influence is observed of the pre-stress.

6.5 Influence aluminium layer thickness and laminate thickness

In the past, most of the residual strength tests were carried out on small, 400 mm wide Glare 2 and Glare 3 material with only three layers of aluminium. For the application of Glare in large aeroplanes, more data were necessary of wide Glare 3 and Glare 4 laminates, thicker laminates and laminates with different aluminium layer thicknesses. This paragraph discusses a large test program that was carried out on Glare 3 and Glare 4 CCT specimens and focuses on the influence of the aluminium layer thickness and the laminate thickness on the residual strength.

6.5.1 Materials, specimen geometry and test procedure

Within this test program a total of 48 flat Glare CCT specimens are tested representing 13 different lay-ups. The test matrix is given in Table 6-7. All Glare types are tested both in L-T direction and in T-L direction. The specimens were 800 mm wide and contained an initial crack length of 200 mm (2a_0 / W = 0.25).

24 specimens have been tested at the Delft University of Technology, DUT, while the remaining 24 specimens were tested at the National Aerospace Laboratory, the NLR.

The crack extension was followed at both sides of the specimen with microscope and a CCD camera. When crack extension occurred, the displacement of the clamps was kept constant and the crack extension was measured after the crack had been stabilized. The CCD camera also recorded the CTOA, which was measured twice after each displacement step; directly when the displacement was stopped and after the crack had been stabilized. After measuring the crack extension and the CTOA the displacement of the clamps was continued.
Table 6-7, Test matrix.

<table>
<thead>
<tr>
<th>Lay-up</th>
<th>Glare 3</th>
<th>Glare 4B</th>
<th>Test location</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Test direction</td>
<td>Test direction</td>
<td></td>
</tr>
<tr>
<td></td>
<td>L-T</td>
<td>T-L</td>
<td>L-T</td>
</tr>
<tr>
<td>3/2 0.3</td>
<td>2×</td>
<td>2×</td>
<td>2×</td>
</tr>
<tr>
<td>4/3 0.4</td>
<td>2×</td>
<td>2×</td>
<td>2×</td>
</tr>
<tr>
<td>4/3 0.5</td>
<td>2×</td>
<td>2×</td>
<td>2×</td>
</tr>
<tr>
<td>6/5 0.4</td>
<td>1×</td>
<td>1×</td>
<td>-</td>
</tr>
<tr>
<td>6/5 0.5</td>
<td>-</td>
<td>-</td>
<td>2×</td>
</tr>
<tr>
<td>7/6 0.4</td>
<td>1×</td>
<td>1×</td>
<td>-</td>
</tr>
<tr>
<td>8/7 0.4</td>
<td>2×</td>
<td>2×</td>
<td>2×</td>
</tr>
<tr>
<td>8/7 0.5</td>
<td>2×</td>
<td>2×</td>
<td>2×</td>
</tr>
</tbody>
</table>

No standard Glare 3 7/6 0.4 material. It contains two outer Alclad layers of 0.3 mm thickness and 5 internal bare aluminium layers with a thickness of 0.4 mm.

6.5.2 Test results

The test results are summarised in Table 6-8 to Table 6-11. The Stress Intensity Factors (SIF’s) presented in these tables are defined as:

\[
K_i = \frac{1}{\cos \left( \frac{\pi a_0}{W} \right)} \sigma_i \sqrt{\pi a_0} \quad \text{eq. 6-1}
\]

\[
K_e = \frac{1}{\cos \left( \frac{\pi a_0}{W} \right)} \sigma_c \sqrt{\pi a_0} \quad \text{eq. 6-2}
\]

\[
K_c = \frac{1}{\cos \left( \frac{\pi a_c}{W} \right)} \sigma_c \sqrt{\pi a_c} \quad \text{eq. 6-3}
\]

Figure 6-45 to Figure 6-48 illustrate the residual strength and different stress intensity values as a function of the MVF. The axis on the left-hand side represents the stress intensity factors while the axis at the right-hand side represents the residual strength.

The results show that the residual strength values for specimens of a comparable Glare variant and loading direction are almost equal, independent of the amount of layers.
Figure 6-45, Residual strength and characterising SIF's for different Glare 3 laminates tested in L-T direction.

Table 6-8, Test results Glare 3, tested in L-T direction.

<table>
<thead>
<tr>
<th>Lay-up and t</th>
<th>MVF</th>
<th>$\sigma_a$ [MPa]</th>
<th>$\sigma_c$ [MPa]</th>
<th>$\Delta a$ [mm]</th>
<th>$K_i$ [MPa m$^{0.5}$]</th>
<th>$K_e$ [MPa m$^{0.5}$]</th>
<th>$K_c$ [MPa m$^{0.5}$]</th>
</tr>
</thead>
<tbody>
<tr>
<td>3/2 0.3 t = 1.45</td>
<td>64.3</td>
<td>108.5</td>
<td>201.7</td>
<td>32.3</td>
<td>63.4</td>
<td>118.0</td>
<td>139.9</td>
</tr>
<tr>
<td>4/3 0.4 t = 2.43</td>
<td>68.1</td>
<td>125.2</td>
<td>204.7</td>
<td>35.1</td>
<td>73.2</td>
<td>119.7</td>
<td>143.9</td>
</tr>
<tr>
<td>4/3 0.5 t = 2.80</td>
<td>72.7</td>
<td>128.3</td>
<td>207.2</td>
<td>34.7</td>
<td>74.6</td>
<td>120.5</td>
<td>144.7</td>
</tr>
<tr>
<td>6/5 0.4 t = 3.70</td>
<td>65.8</td>
<td>122.2</td>
<td>213.0</td>
<td>28.9</td>
<td>71.7</td>
<td>125.0</td>
<td>145.8</td>
</tr>
<tr>
<td>7/6 0.4 t = 4.22</td>
<td>63.4</td>
<td>138.0</td>
<td>208.8</td>
<td>27.8</td>
<td>81.1</td>
<td>122.7</td>
<td>142.3</td>
</tr>
<tr>
<td>8/7 0.4 t = 5.13</td>
<td>64.6</td>
<td>117.5</td>
<td>197.6</td>
<td>36.8</td>
<td>69.0</td>
<td>116.1</td>
<td>141.8</td>
</tr>
<tr>
<td>8/7 0.5 t = 5.84</td>
<td>69.6</td>
<td>116.0</td>
<td>200.6</td>
<td>35.1</td>
<td>67.7</td>
<td>117.2</td>
<td>140.9</td>
</tr>
</tbody>
</table>

1) No standard Glare 3 7/6 0.4 material. It contains two outer Alclad layers of 0.3 mm thickness and 5 internal bare aluminium layers with a thickness of 0.4 mm.
Figure 6-46, Residual strength and characterising SIF's for different Glare 3 laminates tested in T-L direction.

Table 6-9, Test results Glare 3, tested in T-L direction.

<table>
<thead>
<tr>
<th>Lay-up and t [mm]</th>
<th>MVF</th>
<th>$\sigma_i$ [MPa]</th>
<th>$\sigma_c$ [MPa]</th>
<th>$\Delta a$ [mm]</th>
<th>$K_i$ [MPa $\sqrt{m}$]</th>
<th>$K_e$ [MPa $\sqrt{m}$]</th>
<th>$K_c$ [MPa $\sqrt{m}$]</th>
</tr>
</thead>
<tbody>
<tr>
<td>3/2 0.3, t = 1.45</td>
<td>64.3</td>
<td>62.0</td>
<td>186.3</td>
<td>28.2</td>
<td>36.2</td>
<td>108.9</td>
<td>126.6</td>
</tr>
<tr>
<td>4/3 0.4, t = 2.45</td>
<td>68.1</td>
<td>114.4</td>
<td>182.7</td>
<td>31.1</td>
<td>66.9</td>
<td>106.8</td>
<td>126.0</td>
</tr>
<tr>
<td>4/3 0.5, t = 2.80</td>
<td>72.7</td>
<td>121.7</td>
<td>195.7</td>
<td>33.0</td>
<td>71.2</td>
<td>114.5</td>
<td>136.2</td>
</tr>
<tr>
<td>6/5 0.4, t = 3.70</td>
<td>65.8</td>
<td>121.7</td>
<td>193.1</td>
<td>27.9</td>
<td>71.4</td>
<td>113.4</td>
<td>131.5</td>
</tr>
<tr>
<td>7/6 0.4, t = 4.25</td>
<td>63.4</td>
<td>114.0</td>
<td>187.1</td>
<td>36.1</td>
<td>66.7</td>
<td>109.5</td>
<td>132.3</td>
</tr>
<tr>
<td>8/7 0.4, t = 5.14</td>
<td>64.6</td>
<td>115.4</td>
<td>178.9</td>
<td>38.5</td>
<td>67.4</td>
<td>104.6</td>
<td>127.8</td>
</tr>
<tr>
<td>8/7 0.5, t = 5.85</td>
<td>69.6</td>
<td>106.6</td>
<td>191.1</td>
<td>42.3</td>
<td>62.2</td>
<td>111.6</td>
<td>138.9</td>
</tr>
</tbody>
</table>

1) No standard Glare 3 7/6 0.4 material. It contains two outer Alclad layers of 0.3 mm thickness and 5 internal bare aluminium layers with a thickness of 0.4 mm.
Sharp notch design tool for Glare 4B laminates tested in L-T direction.

Table 6-10, Test results Glare 4B, tested in L-T direction.

<table>
<thead>
<tr>
<th>Lay-up and t</th>
<th>MVF</th>
<th>$\sigma_i$ [MPa]</th>
<th>$\sigma_c$ [MPa]</th>
<th>$\Delta a_c$ [mm]</th>
<th>$K_i$ [MPa (\sqrt{m})]</th>
<th>$K_e$ [MPa (\sqrt{m})]</th>
<th>$K_c$ [MPa (\sqrt{m})]</th>
</tr>
</thead>
<tbody>
<tr>
<td>3/2 0.3, t = 1.69</td>
<td>54.5</td>
<td>102.4</td>
<td>181.4</td>
<td>32.9</td>
<td>59.7</td>
<td>105.8</td>
<td>125.9</td>
</tr>
<tr>
<td>4/3 0.4, t = 2.78</td>
<td>58.7</td>
<td>111.9</td>
<td>194.6</td>
<td>35.1</td>
<td>65.2</td>
<td>113.5</td>
<td>136.5</td>
</tr>
<tr>
<td>4/3 0.5, t = 3.20</td>
<td>64.0</td>
<td>97.4</td>
<td>184.3</td>
<td>30.2</td>
<td>57.0</td>
<td>107.8</td>
<td>126.5</td>
</tr>
<tr>
<td>6/5 0.5, t = 5.04</td>
<td>61.5</td>
<td>109.4</td>
<td>177.7</td>
<td>27.1</td>
<td>63.9</td>
<td>103.8</td>
<td>120.0</td>
</tr>
<tr>
<td>8/7 0.4, t = 6.0</td>
<td>54.9</td>
<td>99.2</td>
<td>175.2</td>
<td>43.7</td>
<td>58.1</td>
<td>102.6</td>
<td>128.4</td>
</tr>
<tr>
<td>8/7 0.5, t = 6.74</td>
<td>60.4</td>
<td>102.6</td>
<td>178.6</td>
<td>28.2</td>
<td>59.9</td>
<td>104.4</td>
<td>121.4</td>
</tr>
</tbody>
</table>

Figure 6-47, Residual strength and characterising SIF's for different Glare 4B laminates tested in L-T direction.
Figure 6-48,  Residual strength and characterising SIF’s for different Glare 4B laminates tested in T-L direction.

Table 6-11, Test results Glare 4B, tested in T-L direction.

<table>
<thead>
<tr>
<th>Lay-up and t</th>
<th>MVF</th>
<th>$\sigma_i$ [MPa]</th>
<th>$\sigma_c$ [MPa]</th>
<th>$\Delta\sigma_c$ [MPa]</th>
<th>$K_i$ [MPa $\sqrt{m}$]</th>
<th>$K_e$ [MPa $\sqrt{m}$]</th>
<th>$K_c$ [MPa $\sqrt{m}$]</th>
</tr>
</thead>
<tbody>
<tr>
<td>3/2 0.3 t = 1.70</td>
<td>54.5</td>
<td>118.5</td>
<td>205.7</td>
<td>38.2</td>
<td>69.1</td>
<td>119.9</td>
<td>146.4</td>
</tr>
<tr>
<td>4/3 0.4 t = 2.79</td>
<td>58.7</td>
<td>155.0</td>
<td>205.1</td>
<td>29.2</td>
<td>39.0</td>
<td>119.6</td>
<td>139.8</td>
</tr>
<tr>
<td>4/3 0.5 t = 3.21</td>
<td>64.0</td>
<td>114.7</td>
<td>209.2</td>
<td>36.8</td>
<td>67.1</td>
<td>122.3</td>
<td>148.3</td>
</tr>
<tr>
<td>6/5 0.5 t = 5.05</td>
<td>61.5</td>
<td>119.8</td>
<td>201.8</td>
<td>34.7</td>
<td>70.0</td>
<td>117.9</td>
<td>141.5</td>
</tr>
<tr>
<td>8/7 0.4 t = 6.01</td>
<td>54.9</td>
<td>136.7</td>
<td>204.3</td>
<td>42.3</td>
<td>80.0</td>
<td>119.5</td>
<td>148.7</td>
</tr>
<tr>
<td>8/7 0.5 t = 6.76</td>
<td>60.4</td>
<td>134.5</td>
<td>206.3</td>
<td>32.5</td>
<td>78.7</td>
<td>120.7</td>
<td>143.2</td>
</tr>
</tbody>
</table>

Glare 4B T-L dir. Glare 4B 3/2 0.3 Glare 4B 4/3 0.4 Glare 4B 4/3 0.5 Glare 4B 6/5 0.5 Glare 4B 8/7 0.4 Glare 4B 8/7 0.5
Most of the tests are carried out twice. The difference or scatter between the residual strength results of two comparable tests is remarkably small. The result for $K_c$ show the same trend as $K_e$ and $\sigma_c$ and also hardly any scatter. This means that also the critical crack extension is comparable for all specimens. The $K_i$ values show a somewhat larger but still small amount of scatter, which is normal for crack initiation.

No influence of the laminate thickness on the residual strength can be observed. This corresponds with the idea that the laminate is build up from thin layers and that the material behaviour is determined by the thickness of the separate layers instead of the total thickness or amount of layers. It is expected that no large stresses can develop in thickness direction and that a plane stress condition occurs in the thin aluminium layers, independent of the thickness of the laminate. No clear influence of the aluminium layer has been observed. However, the difference in the investigated range of aluminium layer thicknesses between 0.3 and 0.5 mm is probably too small.

![Graph showing residual strength results](image)

**Figure 6-49** Average residual strength results for different Glare variants. Number between brackets identifies amount of fibres in loading direction.

Figure 6-49 contains the average residual strength results for the different Glare variants. The horizontal lines represent the average residual strength for a certain variant, tested either in L-T or T-L direction. Regarding the amount of scatter, it is difficult to indicate whether the residual strength of a Glare variant increases or decreases for a larger MVF. Assuming that the position of the fibre layers in- and perpendicular to the loading direction is of minor importance, the Glare 3 results illustrate the superior residual
strength behaviour of 2024-T3 in L-T direction compared to 2024-T3 in T-L direction. For the tested configuration, the average residual strength of Glare 3 in L-T direction is 9% higher than in T-L direction. 

The addition of one extra fibre layer in loading direction to the 0/90 prepreg in a Glare 3 T-L specimen results in a Glare 4B T-L variant. Since the results of the Glare 4B specimens in T-L direction are comparable with the Glare 3 results in L-T direction, the addition of this extra fibre layer seems to compensate for the rolling direction of the aluminium. The opposite seems to be true as well; adding an extra fibre layer perpendicular to the loading direction in a Glare 3 L-T specimen results in a Glare 4B L-T specimen with a comparable residual strength as Glare 3 T-L.

If the stress concentration in the prepreg layers is disregarded and the load carrying capability of a fibre layer perpendicular to the loading direction is set to zero, a 0/90/0 prepreg layer like in Glare 4B T-L laminate can carry a 33% higher stress than a 0/90 prepreg layer in Glare 3 T-L (2 x 0º / 3 layers = 2/3 versus 1 x 0º / 2 layers = 1/2). Since the MVF for these Glare 4B laminates is in the order of 65%, the addition of an extra prepreg layer would result in a residual strength increase of approximately 0.35 x 33% = 12%. The average results presented in Figure 6-49 showed a residual strength increase of 8.6% by adding an extra fibre layer to a Glare 3 T-L laminate. In the opposite way by taking a Glare 3 variant tested in L-T direction and adding a fibre layer perpendicular to the loading direction would result in a residual strength decrease of 12%. In reality, an average decrease of 11.3% was measured. These first rough predictions indicate that the assumptions at the beginning of this paragraph are not that bad as a first approximation. The influence of the aluminium rolling direction was disregarded.

Based on the available test results in literature for Glare 3 and 2024-T3 that illustrate that the residual strength of 2024-T3 is slightly higher than Glare 3 in L-T direction and slightly lower in T-L direction (see also chapter 3), it is expected that the contribution of a 0/90 prepreg layer to the residual strength of the laminate is slightly smaller than the contribution of a 2024-T3 aluminium layer in rolling direction. Perpendicular to the rolling direction, the contribution the contribution of a 0/90 prepreg layer to the residual strength of the laminate is expected to be slightly larger than the contribution of a 2024-T3 layer. Based on this experience, the expected trends for the Glare 3 and Glare 4B specimens are illustrated in Figure 6-50.
Unfortunately, none of the charts really support the existence of these relations nor prove them to be wrong. It is expected that the minor scatter in the results in relation to the small range of MVF-values cause this.

![Graph showing residual strength vs MVF](image)

Figure 6-50, Expected trend between residual strength and MVF.

### 6.6 Development design tool

The results of the previous paragraphs illustrated the dependency of the laminate residual strength properties on the fibre- and aluminium material characteristics and the bonding strength of the prepreg with the aluminium layers. The metal layer is responsible for the stable crack extension, yielding of the material and the redistribution of the stress away from the crack tip. The fibre layers are responsible for the large strain hardening of the material and final fracture. Without aluminium layers or in case of bad bonding quality, the fibre layer in front of a crack would split as illustrated in the pre-treatment research and show no stable crack extension but fracture suddenly. Because the Glare laminates illustrate residual strength behaviour that is more comparable with the behaviour of aluminium than for composites, the development of a calculation method is based on methods that are applicable for aluminium. Two methods are discussed; the CTOA approach in § 6.6.1 and the $K_0$-curve approach in § 6.6.2.
§ 6.6.3 describes the development of a calculation method to predict the residual strength for arbitrary Glare variants in L-T or T-L direction based on the expected trend in Figure 6-50 and the $K_{cr}$-curve approach. The influence of the bond strength is not implemented in the design tool.

### 6.6.1 CTOA approach

In the extensive research program described in § 6.5 the critical CTOA has been measured during several residual strength experiments. The CTOA fracture criterion assumes that the angle maintains a constant value during stable crack growth. The strength of this criterion lies in the fact that it is a very local parameter and is therefore easily applied in structures. If a critical CTOA exists and can be used for predictions, the method only needs an accurate finite element model of the region under investigation and that is able to incorporate the influence of the surrounding structure (frames, stiffeners, rivets, etc) on the stress field around the investigated crack.

Seshadri et al.\textsuperscript{23} measured a constant critical CTOA value for 2024-T3 of 5° and found that finite element simulations using the STAGS shell code and an angle of 4.6° in L-T direction or 4.8° in T-L direction produced quite accurate results of crack extension and failure loads\textsuperscript{24}.

Figure 6-51 to Figure 6-54 show the measured CTOA values for the different experiments carried out on Glare 3 and Glare 4B specimens. Two different measuring techniques of the critical CTOA's were applied based on the test location. In both cases, the critical CTOA was measured with the software package Impress, discussed in chapter 4. However, the critical CTOA's in the thinner specimens (3/2 and 4/3 lay-ups) were measured during the test while the critical CTOA's of the thicker laminates were measured after the test. In the latter case, images were captured and stored during the test. The time between fracture and storage is in the order of 1 to 2 seconds while the time between fracture and the online measurement was in the order of 10 seconds. Therefore, the CTOA results of these tests are more closely related to the fracturing process.

All figures show that the CTOA goes to a more or less constant but large bandwidth for larger crack length between 4 and 8°. Only the results for Glare 4B T-L show a smaller bandwidth. With the presence of the fluctuation or scatter in the critical CTOA results and the discussion in the previous paragraph of the different measuring techniques, it is hard to see if there is an influence of the thickness of the laminate or the thickness of the aluminium layers. Although the online measurements of the critical CTOA's
for thinner laminates are within the bandwidth of the results for the thicker laminates, they are in general somewhat larger. This is well illustrated in the graph for Glare 4B L-T.

Figure 6-51, Measured CTOA's as a function of the physical crack extension, Glare 3 L-T.

Figure 6-52, Measured CTOA's as a function of the physical crack extension, Glare 3 T-L.
Based on the fracture mechanism for larger crack length that is dominated by fracture of the fibres, it is expected that the critical CTOA of the Glare laminates should be equal or smaller than the critical CTOA for 2024-T3
reported by Seshadri but certainly not larger. These smaller angles are not clearly observed. However, it must be considered that the critical CTOA is measured over a distance of 1 mm from the crack tip where the broken fibres try to open the crack. In this way, the critical CTOA for Glare would decrease closer towards the crack tip.

To investigate the applicability of critical CTOA values to predict crack growth and failure in Glare panels, a first analysis has been carried out with the general-purpose finite element program ABAQUS. The Glare used in the investigation was Glare 3 3/2 0.4 with a thickness of 1.7 mm and tested in T-L direction. For the FE model, the material is assumed to be homogeneous and orthotropic. The model is build with CPS8R plane stress shell elements, which represent the total laminate instead of single layers of aluminium or prepreg. Most of the applied material properties, given in Table 6-12 originate from test results discussed in chapter 9. The Poisson’s ratio and shear modulus are taken from literature25.

<table>
<thead>
<tr>
<th>$E_{1,2}$ [MPa]</th>
<th>$G_{1,2}$ [MPa]</th>
<th>$\nu_{1,2}$</th>
<th>$\sigma_{0,0}$ [MPa]</th>
<th>$\varepsilon_{0,0}$ [%]</th>
<th>$\sigma_{\text{ult}}$ [MPa]</th>
<th>$\varepsilon_{\text{ult}}$ [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>58500</td>
<td>16150</td>
<td>0.27</td>
<td>150</td>
<td>0.25</td>
<td>603</td>
<td>4.6</td>
</tr>
</tbody>
</table>

Plasticity is modelled with the von Mises yield criterion, associated flow rule and isotropic hardening. A first verification of the finite element modelling is done by modelling a uni-axial dog bone specimen, determine the tensile stress strain curve and comparison of the results with an experimental stress strain curve. The calculated and experimental results were comparable25.

Because of symmetry, only one quarter of the CCT specimen is modelled with fine discretization in the area of the crack tip. In this region, the node distances are 1 mm. An illustration of the boundary conditions and the applied mesh is given in Figure 6-55. The convergence of the calculations has been checked by refining the mesh in the region surrounding the crack.

Figure 6-56 shows the finite element results for different critical CTOA values, varying from 6 to 8 degrees. The applied force is plotted for different crack extensions. As can be seen in this figure, this difference of 2 degrees resulted in a predicted difference in residual strength of 30 kN or 21% while the band width of the results for the critical CTOA is even 4 degrees. Thus, this method seems to be very sensitive for differences in the CTOA. The
critical CTOA was 7.5º, which is within the band of possible CTOA’s, but at the upper side.

![Figure 6-55](image.png)

Figure 6-55, Boundary conditions and mesh for finite element calculations.

![Figure 6-56](image.png)

Figure 6-56, Applied force versus physical crack extension, calculated for different critical CTOA values, Material Glare 3 3/2 0.4.

An additional study on the use of the critical CTOA on the same material but with local glass fibre reinforcement strips in loading direction has been
carried out. The specimen considered, contained three of these reinforcement strips; one in the centre of the sheet that was cut by the crack and two in front of the crack that were still intact. More information about the experiment is given in chapter 9. To predict the residual strength of this panel, the FE calculations resulted in a critical CTOA of 8.5º. This means that the broken centre strip should have increased the critical CTOA. This is possible based on the definition of the CTOA, 1 mm behind the crack tip. However, as the CTOA should be a material constant it cannot be used in this way.

The CTOA method might have potential as a describing material parameter and needs further investigation towards an applicable CTOA definition for Glare and a sensitivity analysis of the CTOA for residual strength predictions. However, in this study the attention is mainly focussed on the $K_R$-curve approach since this approach already showed to be a material in chapter 3. This is discussed in the next paragraph.

### 6.6.2 $K_R$-curve approach

Figure 6-57 to Figure 6-64 presents the experimental $K_R$-curves for Glare 3 L-T, Glare 3 T-L, Glare 4B L-T and Glare 4B T-L obtained within the large research program described in § 6.5. Both the $K_R$-curves based on the Irwin correction and the compliance correction are presented for a specific Glare variant and loading direction. It is remarkable how small the differences are between the $K_R$-curves of the different lay-ups and aluminium thicknesses.

The $K_R$-curves based on the Irwin plastic zone correction illustrate that the initial part of the curves up to crack initiation are the same for all laminates. This can cause a problem if too small specimens are used to obtain these curves, which was discussed in chapter 3. However, for the 800 mm wide specimens, NSY occurred close to or beyond final fracture for all Glare variants. This is illustrated in Figure 6-65 for the Glare 3 specimens tested in L-T direction but representative for all specimens tested. Therefore, most of the stable crack extension is incorporated in the remaining part of the curve.

The $K_R$-curves based on the Compliance correction show somewhat more scatter, especially in the beginning, but are comparable as well for the different laminates. Initially, the “Irwin” curves are somewhat below the “compliance” curves since the effective crack length according to Irwin starts to extend when the loading is applied. However, after this initial difference, the curves become comparable.
Chapter 6

**$K_{R}$-curves Glare 3 L-T**

![Graph showing $K_{R}$-curves for Glare 3 laminates tested in L-T direction.](image)

**Figure 6-57**, $K_{R}$-curves based on *Irwin plasticity correction* for different Glare 3 laminates, tested in L-T direction.

![Graph showing $K_{R}$-curves for Glare 3 laminates tested in L-T direction.](image)

**Figure 6-58**, $K_{R}$-curves based on *Compliance plasticity correction* for different Glare 3 laminates, tested in L-T direction.
Sharp notch design tool for Glare

**K_R-curves Glare 3 T-L**

![Graph of K_R-curves Glare 3 T-L](image)

Figure 6-59, $K_R$ -curves based on *Irwin plasticity correction* for different Glare 3 laminates, tested in T-L direction.

![Graph of K_R-curves Glare 3 T-L](image)

Figure 6-60, $K_R$ -curves based on *Compliance plasticity correction* for different Glare 3 laminates, tested in T-L direction.
**K_R-curves Glare 4B L-T**

Figure 6-61, $K_R$ -curves based on *Irwin plasticity correction* for different Glare 4B laminates, tested in L-T direction.

Figure 6-62, $K_R$ -curves based on *Compliance plasticity correction* for different Glare 4B laminates, tested in L-T direction.
Sharp notch design tool for Glare

$K_{R}$-curves Glare 4B T-L

Figure 6-63, $K_{R}$ -curves based on *Irwin plasticity correction* for different Glare 4B laminates, tested in T-L direction.

Figure 6-64, $K_{R}$ -curves based on *Compliance plasticity correction* for different Glare 4B laminates, tested in T-L direction.
In Figure 6-64, the curve for a specimen with buckling is presented as well. Although A.B. guides were applied, they were not fully effective in avoiding the occurrence of an out of plane displacement of the crack edges. The influence on the $K_R$-curve is illustrated by the larger increase of effective crack length initially. This is caused by the faster increase of the COD from the onset of buckling. For larger crack extension, the curve tends to return to the non-buckled curves. This can be explained since the A.B. guides become more effective in sustaining any further out of plane displacement with larger crack length.

![Graph showing average net section stress as a function of physical crack extension for different Glare laminates tested in L-T direction.](image)

Figure 6-65. Average net section stress as a function of the physical crack extension for different Glare laminates, tested in L-T direction.

As far data are available, they have been compared with the $K_R$-curves presented in this paragraph. This was possible for example for Glare 3 3/2 0.3 L-T. Comparing Figure 6-58 with Figure 3-44 illustrates that the curves presented in this paragraph for 800 mm wide CCT specimens and $2a_0 = 200$ mm estimate with the curves for 200 and 400 mm wide CCT specimens with several different initial crack lengths. This proves again that the $K_R$-curve can be used as a material parameter for Glare laminates.

Figure 6-66 and Figure 6-67 show the $K_R$-curves obtained with the research on Glare 3 like laminates with different aluminium alloys. This research was described in § 6.3.1. The results based on the Irwin plasticity correction illustrate a clear difference between the different alloys. However, these results are once again misleading. Because the yield stress of the different aluminium alloys is different, the plastic zone correction differs as well. For
a comparable gross stress, the plastic zone size for a material with a higher yield stress is smaller. As long as no crack extension occurs, the $K_{IR}$-curves based on the Irwin correction for a material with a higher yield stress must be more to the left. Comparing the different laminate yield stresses presented in Table 6-3 with the curves of Figure 6-66 illustrates this. As soon as physical crack extension takes place, the situation changes. This is illustrated well for the curves of 7075-T6 and 7475-T6, which suddenly show an extensive effective crack extension and cross the other curves. Because the specimens were only 400 mm wide, most curves could only be derived to a very small crack extension due to the occurrence of NSY. Therefore, the curves in Figure 6-66 are not representative.

![Graph showing $K_{IR}$-curves](image)

Figure 6-66, $K_{IR}$-curves based on *Irwin plasticity correction* for Glare 3 3/2 0.3 like laminates with different aluminium alloys laminates, tested in L-T direction.

Figure 6-67 show the $K_{IR}$-curves obtained with *Compliance plasticity correction*. Now, the difference between the curves is much smaller. Initially, the curve for 2024-T3 seems to be somewhat lower than the other curves. However, taking into account the amount of scatter for small effective crack extension illustrated in Figure 6-58, this seems to be within the scatter band. After some effective crack extension, the curve for 2024-T3 exceeds the curves for the other materials, which means that this combination of aluminium alloy and glass fibres represents the highest residual strength. The lowest residual strength curve is obtained for Glare with 7075-T6.
Figure 6-67, $K_R$ -curves based on Compliance plasticity correction for Glare 3 3/2 0.3 like laminates with different aluminium alloys laminates, tested in L-T direction.

6.6.3 Development prediction tool

As illustrated in chapter 3 and once again in the previous paragraph, the $K_R$ - or crack growth resistance curve of a specific Glare laminate is able to describe the laminate residual strength properties. This curve is a material "parameter" since it is independent of the geometry. However, every Glare variant has its own $K_R$ - curve that needs to be determined experimentally. Therefore, it would be easier if the $K_R$ - curve of a specific laminate could be based upon the constituent properties.

Based on the experimental results obtained during this study, it can be concluded that the residual strength or more general, the $K_R$-curve of a Fibre Metal laminate is a function of:

1. the amount of aluminium in the laminate, represented by the MVF,
2. the material properties of the metal layers,
3. the rolling direction of the metal sheets compared to the loading direction,
4. the amount of fibres in the laminate in loading direction,
5. the material properties of the fibre layers, and
6. the interfaces between the different layers
while the fibres perpendicular to the loading direction are expected not to add any strength in this direction.

A Fibre Volume Fraction comparable with the MVF, see eq. 1.1, can represent the amount of fibres in loading direction. This fraction will be noted as $FVF_{LD}$. In this definition only the thickness of the fibre layers in loading direction (LD) are taken into account, which explains the subscript.

Figure 6-68 contains $K_R$ – curves based on the compliance correction for a wide variety of Glare laminates in L-T direction all based on the same aluminium alloy 2024-T3. According to the list given above, the residual strength ranking for these laminates is based only on the MVF and the $FVF_{LD}$ (nr. 1 and 4), because the laminates have the same fibre- and pre-treated aluminium layers and are all tested in rolling direction. Figure 6-68 also contains a reference curve for 2024-T3 L-T and a table with the MVF and $FVF_{LD}$ values for the different laminates.

![Figure 6-68, $K_R$ - curves in L-T direction for different Glare variants and 2024-T3, based on compliance plasticity correction.](image)

To obtain a relation between the MVF, the $FVF_{LD}$ and $\sigma_{res}$, experimental $\sigma_{res}$ values are collected for several laminates obtained on 800 mm wide CCT panels with $2a/W = 0.25$. As far as data are available, this is also done for laminates tested in T-L direction. These $\sigma_{res}$ values are plotted in Figure 6-69 as a function of $FVF_{LD}$ only. The $FVF_{LD}$ values are presented as a fraction $[0,1]$ not as a percentage. The results on the left axis are for 2024-T3 with $FVF_{LD} = 0$. For increasing $FVF_{LD}$, $\sigma_{res}$ in L-T direction seems to decrease or remain equal first due to the fact that the amount of fibres...
perpendicular to the loading direction, which do not contribute to the residual strength, is relatively large. This is subsequently followed by an increase of $\sigma_{\text{res}}$ up to a Glare 2 laminate with all fibres in loading direction. The trend is comparable for the different aluminium rolling directions but $\sigma_{\text{res}}$ in L-T direction is higher than in T-L direction.

![Figure 6-69](image)

Figure 6-69, Relation between experimental residual strength results and $FVF_{L-D}$.

Based on these results, a new parameter $T$ is introduced representing the effective load-carrying fraction in the laminate. It is defined as:

$$T = \alpha \cdot FVF_{L-D} + MVF$$

where $\alpha$ is a constant that represents the contribution of the fibre layers to the residual strength in comparison to the contribution of the metal layers. $\alpha$ depends on the properties of the aluminium alloy, the properties of the fibres and the aluminium rolling direction. In this definition, $T$ can vary from a value below 1 (but above 0) up to $\alpha$ for a material with fibres in loading direction only. $T$ is presented as a summation of the different constituents contribution. A ratio like $FVF_{L-D}/MVF$ might be more expected. However, such a ratio results for example in comparable values for the Glare 3 and Glare 4B in L-T ratio while both the $FVF_{L-D}$ and the MVF are different.

The residual strength results are plotted as a function of $T$. The value of $\alpha$ is determined such that the Glare and aluminium residuals strength data for a specific test direction are represented by one line. By trial and error is
obtained that $\alpha = 1.84$ for L-T. Although not enough data are available for the T-L direction, the first approximation for $\alpha$ is 2.1 in this direction. The results are illustrated in Figure 6-70. The curve fits were derived in the form

$$\sigma_{\text{res}} = a \cdot T^2 + b \cdot T + c$$

where $\sigma_{\text{res}}$ is the residual strength of a Glare laminate with a value $T$. $W = 800$ mm and $2a_0 = 200$ mm. The curve fit constants $a$, $b$ and $c$ that were obtained are presented in Table 6-13. Within the prediction tool factor is necessary that relates the residual strength of a specific Glare laminate to the residual strength of the aluminium alloy ($T=0$). Therefore, a new variable $M(T)$ is introduced:

$$M(T) = \left( \frac{a \cdot T^2 + b \cdot T + c}{a + b + c} \right)$$

where

The determined values are in accordance to several experiences:

- The lower $\alpha$ for the laminate tested in rolling direction compared to a laminate tested perpendicular to the rolling direction means that the influence of the aluminium layers becomes more important in the value for $T$. That means that the aluminium layers are
contributing more to the residual strength of the laminate in rolling direction than perpendicular to the rolling direction.

- When $\alpha = 2$, the contribution of a 0/90 prepreg layer (Glare 3) to the residual strength is equal to the contribution of the aluminium within the laminate since only 50% of the 0/90 prepreg layer is adding strength. A value of $\alpha$ smaller than 2 means that the 0/90 prepreg layer adds less strength to the laminate than an aluminium layer. The determined value for $\alpha$ in loading direction of 1.84 corresponds to test results that showed that the residual strength of a Glare 3 laminate in L-T direction is lower than for 2024-T3. The opposite is true in T-L direction.

Based on the curve fit equation that represents the relation between $\sigma_{\text{res}}$ and $T$, a residual strength multiplication factor $M$ can be obtained for the residual strength ratio between a laminate and the aluminium alloy.

An example is given for a Glare 4B 4/3 0.4 laminate tested in L-T direction. The thickness of the laminate is 2.625 mm. The MVF and $F_{VFLD}$ are equal to:

$$\text{MVF} = \frac{4 \cdot 0.4\text{mm}}{2.725\text{mm}} = 0.59$$

$$\text{and } F_{VFLD} = \frac{3 \cdot 0.125\text{mm}}{2.725\text{mm}} = 0.14$$

and $T = 1.84 \cdot 0.14 + 0.59 = 0.85$

Based on the curve fit given in Table 6-13,

$$M(T = 0.85) = \frac{164 \cdot 0.85^2 - 125 \cdot 0.85 + 171}{164 - 125 + 171} = 0.87$$

This factor is used to multiply the standard $K_R$ curve of the aluminium alloy in the laminate with the same rolling direction. For the present results, this was 2024-T3:

$$K_R(T) = M(T) \cdot K_{R, 2024-T3} \quad \text{eq. 6-2}$$

To be precise, a $K_R$ curve of a wide aluminium sheet with the same thickness as the single aluminium layers should be used for this purpose. However, a large tests program that was carried out on different thicknesses between 0.8 to 2.0 mm, 2024-T3 CCT specimens illustrated that the difference in $K_R$ curves was within the scatter. More important, the
shape of the curves was identical, which means that use of a different but identically shaped $K_R$ curve is adjusted by the factor $\alpha$.

Table 6-13, Prediction tool.

<table>
<thead>
<tr>
<th>Material:</th>
<th>$\alpha$</th>
<th>a</th>
<th>b</th>
<th>c</th>
<th>Validated range $T$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare L-T (2024-T3)</td>
<td>1.84</td>
<td>164</td>
<td>-125</td>
<td>171</td>
<td>0.82 - 1.3</td>
</tr>
<tr>
<td>Glare T-L (2024-T3)</td>
<td>2.1</td>
<td>307</td>
<td>-561</td>
<td>441</td>
<td>1.02-1.18</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>$K_R$ curve fit</th>
<th>k</th>
<th>l</th>
<th>m</th>
<th>n</th>
<th>Source</th>
</tr>
</thead>
<tbody>
<tr>
<td>2024-T3 L-T (1.6mm)</td>
<td>16.12</td>
<td>-1.06</td>
<td>27.53</td>
<td>0.612</td>
<td>DERA 27</td>
</tr>
<tr>
<td>2024-T3 T-L (1.6mm)</td>
<td>18.08</td>
<td>-0.51</td>
<td>21.49</td>
<td>0.520</td>
<td>DERA</td>
</tr>
</tbody>
</table>

Figure 6-71, Relation between residual strength and $T$ for L-T direction.

The 2024-T3 $K_R$ curves used in these first predictions are based on 2000mm wide, 1.6 mm thick sheets and were derived with the compliance plasticity correction27. The obtained $K_R$ curves for different laminates in this way are plotted in Figure 6-71, together with the experimental $K_R$ curves. It is clear that this approach can be used to predict the $K_R$ curves for different laminates.
The prediction model is summarized in Figure 6-72. It is based on the assumption that \( \alpha \) is known for the aluminium constituent. When a new aluminium alloy is introduced, CCT residual strength tests with a comparable geometry must be carried out for different lay-ups that cover a large range of possible \( T \)-values. With this data a new \( \alpha \) is derived that is related to the new aluminium alloy.

---

**Figure 6-72, Residual strength prediction model.**
The derived prediction model is only validated for Glare variants that are based on 2024-T3 alloys and are loaded under tension in either L-T or T-L direction. Additional research is necessary to extend this prediction model for different loading conditions.

6.7 CONCLUSIONS

A new phenomenological model for fracture of Glare laminates with through the thickness cracks is presented in this chapter. It was known that the interfaces between the aluminium layers and the fibre rich layers in front of a uniaxially loaded crack are loaded with shear stresses. The new model assumes that these shear stresses are not only the result of plasticity but also of crack blunting in the aluminium layers. Although the shear stresses due to plasticity move with crack extension and increased loading, the shear stresses due to blunting reduce with extending crack length. Superposition of these effects results in a decrease of the amount of static delamination for larger crack extensions.

Static delaminations are expected to release the fibre layers from following the large strain concentrations of the aluminium layers at the crack tip. Because the ultimate strain of the fibres is significantly less than for aluminium, failure of the fibres is delayed by this kind of delamination. As a result, the aluminium layers control crack initiation and initial crack growth but the fibre layers will dominate larger crack extensions and fracture.

Semi-destructive- and non-destructive inspections on the specimens that were kept under a constant load confirmed the creation of static delamination before dynamic delamination. The etching method showed that static delamination advanced before the crack tip. Fracture of the fibre layers occurred in bundles, randomly in the close front of the crack tip. It was also observed that the crack extensions in the aluminium layers are initially longer than in the fibre layers. However, after some crack extension and increased load, the cracks in the prepreg layers become longer.

Influence constituent properties
An extensive investigation has been carried out to determine the influence of the constituent properties on the residual strength and to compare these results with the expectations based on the proposed fracture model. A distinction is made between the crack initiation process and the crack growth / fracture process.
Crack initiation
The crack initiation stress of a laminate that is statically loaded increases when an aluminium alloy is used with a lower yield stress. The use of a stiffer fibre prepreg advances crack initiation.

Crack growth / fracture
The residual strength of a laminate is directly related to the strain hardening capacity of the laminate and thus of the aluminium layers in relation to the prepreg; a larger strain hardening increases the residual strength. A reduction of the failure strain and strength of fibres decreases residual strength of a Glare laminate.

It is observed that stable crack extension takes place mostly at loads close to fracture.

Influence of aluminium pre-treatment
To investigate the effect of delamination on the residual strength, the bond strength between the prepreg and aluminium layer has been influenced by applying different pre-treatments to the aluminium layers. A ranking of the bond strength was based on both Lap-shear and Roller peel tests, which gave the same ranking. Fatigue tests illustrated that a better bond strength results in lower crack growth rates and lesser fatigue delamination. On the contrary, the residual strength of a laminate increases with lower bond strength.

Impact
Impact tests are carried out with a blunt impactor on specimens that were pre-loaded. It was concluded that the pre-loading had no influence on the residual strength capabilities of the materials.

Laminate- and aluminium layer thickness
As expected, no influence of the laminate thickness on the residual strength is observed. A slight influence of the aluminium layer thickness is seen as all the laminates show a relation with the MVF. However, for the investigated aluminium layer thicknesses between 0.3 and 0.5 mm, this influence is small.

For the development of a calculation method, both the critical CTOA criterion and the $K_{IC}$ curves approach have been investigated. Although a more or less constant critical CTOA was measured for a large amount of tests on Glare 3 and Glare 4B specimens, the bandwidth of the results seemed to be too large to be applicable. This method needs further
investigation since it can be a very powerful criterion for the analysis of structures.

A large test program illustrated the applicability of the $K_R$-curve approach as a material parameter. Both methods, the Irwin and the compliance method, gave comparable results. A calculation method is derived that is based on the MVF and the FVF in loading direction of a specific laminate. Based on the amount of load carrying layers and a derived ratio between the different layers, a multiplication factor is obtained which is to be multiplied with a standard $K_R$-curve for 2024-T3 to obtain the laminate curve. This works well for curves obtained with the compliance method. To investigate this for the Irwin method, not enough data are available. The derived prediction model is only validated for Glare variants that are based on 2024-T3 alloys and are loaded under tension in either L-T or T-L direction. Additional research is necessary to extend this prediction model for different loading conditions.
References


Sharp notch design tool for Glare


[18] Schwarmann, L, Letter from Dr. Schwarmann (Deutsche Airbus) to Dr. Roebroeks (SLC), Library Structural Laminates Company, Do 202, The Netherlands, 1991


[27] Schwarmann, L., Letter from Dr. Schwarmann (Deutsche Airbus) to Dr Roebroeks (SLC) and Mr. Vermeeren, Library Structural Laminates Company, D0202, The Netherlands, 1991.
Part 3:
Applications
7 BLUNT AND SHARP NOTCH BEHAVIOUR OF SPLICED LAMINATES

7.1 Introduction

In general, as wide as possible sheets are preferred as skin material in a fuselage structure to reduce the amount of necessary joints. This is profitable for the weight, production time and production costs. For the application of Glare, the length of the available aluminium sheets and UD prepreg are sufficient. However, the maximum available sheet width for thin aluminium sheets between 0.2 and 0.5 mm is approximately 1.65m. The manufacturers can provide wider sheets but this would result in a large scrap increase during the rolling process to reach the required accuracy. This will result in a large price increase for the aeroplane manufacturer.

The splicing concept, see Figure 7-1, opens the possibility to increase the maximum sheet sizes for Glare considerably. In this concept, metal sheets are interrupted in the laminate while the fibre layers bridge these splices. With this concept, the sheet dimensions are limited by the autoclave size as evidently illustrated in Figure 7-2.

![Splices (filled with resin) Metal layers Glass fibre layers](image)

Figure 7-1, Illustration of the splicing concept.

Besides the advantages of the splicing concept mentioned above, it also offers a reduction of the inspection time during the operational use of the aeroplane, because of the reduction of the amount of rivets, which act as stress raisers in a structure and are areas prone to fatigue damage.
Chapter 7

Figure 7-2, Single curved Glare fuselage panel with different structural details like doublers and splices. Size: 14 m × 3.5 m (courtesy Fokker Aerostructures B.V.)

The first splice concept illustrated in Figure 7-1, was introduced in 1993 and the design has been continuously investigated and modified since. The first structure that contained splices, was presented at the 1993 Paris Air show; a full-scale 4.8m × 1.7m A330/A340 fuselage side panel made of a spliced Glare skin with stringers, frames, doublers and windows, see Figure 7-3. After the show, this panel was fatigue tested in the A330/A340 barrel set-up at DaimlerChrysler Aerospace Airbus and showed outstanding results in terms of crack propagation.

This chapter discusses the blunt and sharp notch behaviour of spliced Glare sheets. Since the splice configuration has been continuously improved, several different splices were investigated. It occurred several times that the research of a certain splice configuration had not been finished before the introduction of a new splice configuration. Therefore, § 7.2 will give an overview of the splice development before the research towards the different splices is discussed. § 7.3 presents the first splice investigation on the residual strength properties of simple splices. §7.4 presents a residual strength study that was carried out on spliced Glare material covered with doublers. § 7.5 presents an extended experimental program that investigated the static strength, fatigue behaviour, blunt-notch-
and residual strength of a splicing concept with internal doublers. This research determined the existence of different splice details and investigated this detail separately in order to predict the behaviour of an arbitrary splice that is build with these details. The conclusions of this chapter are given in § 7.6.

Figure 7-3, First spliced Glare structure presented at Paris Airshow in 1993.

7.2 Splice development

The aim of the splice research was to develop a splice that needs no or hardly any special attention neither within the design and manufacturing process nor during maintaining an aeroplane. Last but not least, the production of the splice itself must be cost effective to compete with other materials. These items can be translated in the following requirements for an ideal splice:
• The material properties like ultimate-, yield- and blunt-notch strength and but also the outstanding fatigue behaviour should not be reduced.
• The durability of the material must be maintained.
• The weight increase of the Glare sheet due to splices must be limited.
• The production process must be simple and cheap.
• The outer skin must be flat while the inner side of the spliced sheet should not interfere with the locations of stiffeners, frames, doublers and joints.
• The operational advantages of a monolithic aluminium skin must be maintained. These are for example the application of a cladded outer surface layer and the possibility of paint removal.

The remaining text of this paragraph discusses the development of the splice concept over the past years in order to fulfil these requirements.

Simple splice concept
The first splice design was a so-called “simple splice” in which the edges of the aluminium sheets were positioned opposite to each other, see Figure 7-1. In this concept, the effective load-carrying cross-section in the splice area is reduced, which causes strength reduction. Another disadvantage is the rather easy ingress of moisture that that can reach the fibre layers through the adhesive filling in the splice gap and reduce its properties. Therefore, the outer splices need to be protected for example by covering them with doublers. From an aerodynamic standpoint, the outside doublers should be as thin as possible and located in longitudinal direction and not in circumferential direction. § 7.3 presents the first splice investigation towards the residual strength properties of simple splices.

Splice covered with external doubler
The first splices covered with a doubler consisted of an extra 0/90 fibre prepreg layer and an extra aluminium layer, see Figure 7-4 a). The prepreg layer functioned also as adhesive. However, the delamination resistance of this doubler concept was too low¹ and during static loading perpendicular to the splice, the doubler delaminated completely from the spliced skin material. Therefore, the second splice-doubler concept used standard adhesive film instead of prepreg to bond a Glare 3 2/1 0.2 doubler over the splice. This is illustrated in Figure 7-4 b. It was applied in the A330/A340 Glare side panel and gave satisfying results. § 7.4 presents a residual strength study that was carried out on spliced Glare material covered with doublers.
Blunt and sharp notch behaviour of spliced laminates

a) First concept:  

- Spliced base sheet  
- 0.3 mm 2024-T3 doubler  
- 0/90 glass prepreg  

b) A330/A340 Glare side panel  

- Glare 3 2/1 0.2 doubler  
- Adhesive

---

Fibre direction transverse to splice direction (Fibres bridge splice)  
Fibre direction parallel to splice direction

---

Figure 7-4, First splice-doubler concepts.

Both splice-doubler designs contained single sided “thick” doublers, while double-sided “thin” doublers are superior with respect to delamination, durability and fatigue crack initiation. Both ways of adding doublers have some disadvantages:

- the surface is no longer flat which is unfavourable from an aerodynamic point of view for the outside, and
- the addition of doublers cannot be done during curing of the skin material but needs an additional costly curing cycle.

**Splice with internal doublers**

These disadvantages were solved with the introduction of the Self-Forming Technique (SFT). This method takes advantage of the low bending stiffness of the individual layers of the uncured laminate. They will form to any single or slightly double curved mould geometry under the standard curing pressure for the laminate and this geometry will freeze during the cure cycle of the adhesive in the laminate. The concept is illustrated in Figure 7-5 where two aluminium doublers are bonded with adhesive to the spliced Glare laminate. At the location of the outside doubler, the aluminium layers are curved to obtain a flat shape and the adhesive fills the gap between the doubler and the outer aluminium layer.

The concept illustrated in Figure 7-5 still contains some disadvantages:

- For corrosion protection and to allow the airlines to have a “polished skin aeroplane”, the outside aluminium layer will almost certain contain a single sided clad layer. Applying a clad layer within this
concept results in an outside doubler that is bonded to the single sided clad layer. This is known to reduce the durability of the bondline (bondline corrosion). The aeroplane industry will hesitate to allow the use of clad layers in bondlines that are of importance for the integrity of the primary structure.

- Inspection of the aeroplane requires paint removal of the fuselage, several times during the lifetime of an aeroplane. The existing mechanical or chemical paint removal techniques may damage the adhesive fillets on the outside of the skin.
- Unpainted like in the case of a polished skin aeroplane, the adhesive fillets are clearly visible. Using uncoloured adhesive can solve this but will certainly become visible during the operational use of the airplane due to environmental effects and/or polishing operations.

In the splice concept presented in Figure 7-6 the outside doubler is transferred into the laminate. As a consequence, the clad layer of the outer aluminium layers is no longer in the bond line and the flow out of the adhesive can be limited to the 0.1 - 0.5 mm gap between the spliced outer aluminium layers.

The external doubler at the inside of the material is either replaced by a second internal doubler or an overlap of one of the inside aluminium layers which also covers the internal splice. Also the overlapping aluminium layers are bonded with adhesive to the next aluminium layer. In both solutions every splice within the laminate is covered with an extra aluminium layer. §
7.5 presents an extended experimental program that investigated the static strength, fatigue behaviour, blunt-notch- and residual strength of this splicing concept.

Figure 7-6, Splice with internal – and overlapping doubler.

It has been considered to combine longitudinal splices with circumferential splices in order to increase the residual strength properties in both directions. Prototypes with this idea were also manufactured. However, the residual strength research illustrated the limited crack arresting capabilities of single splices what made the application of splices in two directions unnecessary. It was chosen to splice only in longitudinal direction allowing the stringers to run lengthwise unhampered by thickness steps. Because the frames will be connected to the skin with separate clips, these are less sensitive for thickness steps.

In 1999, the German Air force (GAF) decided to modify one of their A310 passenger aeroplanes into a freighter version and asked DCAA to design the implementation of a large cargo door. With the permission of the GAF and the German Airworthiness authorities, DCAA used this opportunity to include a large spliced Glare panel in the redesign of the fuselage, see Figure 7-7.
Initially, the A310 panel contained splices with internal doublers but it turned out during the manufacturing process that the accurate positioning of the spliced aluminium layers with gaps between 0.1 and 0.5 mm only, caused problems. Therefore, a new splice was introduced that contained only overlaps, see Figure 7-8. Within this concept, the gap- and overlap length are larger and it turned out that the necessary positioning accuracy could vary with plus or minus 5 mm

The research of the blunt and sharp notch behaviour of this latter splice concept with overlaps has not been investigated in this study and is not discussed.
7.3 Residual strength investigations for simple splices

The first residual strength experiments on spliced panels within this study were carried out in 1994 on simple spliced panels\(^5\).\(^6\).\(^7\). These experiments needed to address the concern whether a sheet, simply spliced in longitudinal direction and containing a crack in the same direction and area would reduce the residual strength in circumferential direction, see Figure 7-9 a). This research is presented and discussed in § 7.3.1. Subsequently, the simple splice behaviour is investigated when positioned perpendicular to and in front of the crack. This research is described in § 7.3.2.

![Diagram of longitudinal simple splice with crack at "critical" location and derived test specimen geometry.](image)
7.3.1 Cracks parallel to splice

The first residual strength experiments were carried out on flat 400 mm wide CCT-panels with splices in all three aluminium layers and two different \(2a_0\)’s, see Figure 7-9 b). The tested material was Glare 3 3/2 0.3 material with aluminium 7475-T761. The cracks were positioned in the sheet at the location and direction of the centre layer splice. For the experiments the splice distance \(s_d\), defined as the distance between splices in two adjacent aluminium layers, varied. Additional tests were performed on non-spliced CCT-panels as a reference. The results of the experimental program are illustrated in Figure 7-10. A different behaviour occurs between the panels with \(s_d = 0\) and \(s_d > 0\). This is discussed separately.

**Splice distance \(s_d > 0\)**

Compared to the unspliced reference panels, the spliced panels illustrate a decrease of the residual strength. For \(2a_0/W = 1/4\), this decrease was less than 4.5% but increased to a reduction of almost 9% for \(2a_0/W = 1/3\). The reductions are equal for all three different applied \(s_d\) values. The reductions are attributed to the reduced effective load carrying cross section.

**Splice distance \(s_d = 0\)**

In these specimens, all three splices in the aluminium layer coalesce with the initial crack. This actually represents a situation where all aluminium layers are cracked and the load has to be carried by the intact fibres in the

![Figure 7-10, Residual strength results of panels with splices perpendicular to the loading direction.](image-url)
net-section, see Figure 7-11. In contrast with the specimens with \( s_d > 0 \), the specimens exhibited no stable crack extension. The residual strength based on the gross cross section was somewhat higher compared to the reference panels although the net section was significantly decreased (residual strength improvement of 3.5% for \( 2a_0/W = 0.25 \) and 2.8% for \( 2a_0/W = 0.33 \)).

![Figure 7-11, Situation for \( s_d = 0 \).](image)

After the tests, the outer aluminium layers were etched away and the fibre layers illustrated a considerable amount of fibre failure delamination indicating a large storage of elastic energy in the fibre layers before failure. It seems that the specimens behaved more like two parallel tensile tests on fibre prepregs. Since none of the aluminium layers bridges the crack, the stress concentration at the crack tip will be lower as discussed in chapter 6. This resulted in a somewhat higher residual strength.

### 7.3.2 Cracks perpendicular to splice

A second test series has been performed out to investigate the residual strength behaviour of longitudinal splices in combination with cracks in circumferential direction, see Figure 7-9 a). The experiments focussed on the residual strength of a cracked panel in the presence of splices in front of the crack. Figure 7-12 illustrates the specimen geometry and the variables; \( s_s \) and the splice pitch \( s_p \). \( s_p \) is defined as the distance between two splices in the same layer while \( s_s \) is the distance between splices in different layers assuming that the splices in all layers are grouped.

The experimental residual strength results compared to a non-spliced specimen are illustrated in Figure 7-13. It illustrates that splices located in front of a crack tend to increase the residual strength but that this effect decreases rapidly with the distance between the original crack tip and the
first splice. Remarkable are the results for \( s_d \); a larger \( s_d \) should have a positive influence when the first splice is located 10 mm in front of the crack and a negative influence if this distance is 17.5 mm. It is expected that these differences are within the scatter of the residual strength results and that \( s_d \) has no influence on the residual strength.

**Figure 7-12, Geometry of test panel with splices perpendicular to the crack.**

**Figure 7-13, Residual strength results of CCT panels with non-overlapping splices in front of the initial crack tips.**

The residual strength improvements can be understood from the point of view that the crack needs to reinitiate into non-cracked aluminium layers. Depending on the distance between the crack tip and the first splice, this splice is able to arrest the crack in its layer during stable crack growth. As a
Blunt and sharp notch behaviour of spliced laminates

Consequence, crack growth in the other layers is retarded. This is illustrated
in Figure 7-14 where the applied gross stress is plotted as a function of the
physical crack extension. The standard definition of the physical crack
extension is used here; being the average crack extension in both outer
aluminium layers and at both crack tips. None of the specimens showed
any stable crack extension after the crack reinitiated in a new aluminium
layer.

Figure 7-14, Illustration of the crack stopping effect of simple splices in front
of the crack.

When the crack reaches the splice, a large delamination is created between
the spliced aluminium layer and the fibre layers due to the large plastic
zone in the aluminium layer. This is illustrated in Figure 7-15 a) where a
spliced outer layer is shown just before failure and after unloading. During
the test, the other side of the crack failed. After chemically etching the outer
aluminium layer away, the static delamination below of the fibre layer
becomes visible while the fibres are still intact, see Figure 7-15 b). This
delaminated zone decreases the stress intensity of the fibre layers at the
 crack tip and explains the higher residual strength.

Residual strength tests on 800 mm wide comparable Glare 3 3/2 0.3
specimens and splices in front of the crack tip with \( a_0 = 25.4 \) mm gave
comparable results. A splice located 50mm in front of a 200 mm long initial
crack was able to arrest the crack significantly and gave a residual strength
increase of 8%. When the splice was located 100mm in front of the crack
tip, the residual strength increase was only 1.6%, which lies within the
gen-eral scatter band of residual strength results.

Material:
- Glare 3 3/2 0.3 L-T based on 7475-T761
- with splices in front of initial crack.
- CCT specimen, W=400mm, 2a=100mm
7.4 Simple splices covered with doublers

As discussed in §7.2, splices need to be reinforced and protected against moisture ingress. Both aspects could be obtained either with a smart splice design or by bonding a thin doubler on top of the outside splice gap. If cracks occur perpendicular to the splices and covering doublers, the doublers can also act as crack stoppers. However, since the doublers will be thin the crack arresting possibilities are limited. Two residual strength investigation have been carried out regarding these external doublers:

- § 7.4.1 discusses a residual strength program with all aluminium layers in the Glare laminate spliced below one single doubler, while
- § 7.4.2 presents a residual strength investigation for specimens in which only one aluminium layer was spliced below a single doubler.
7.4.1 All layers spliced below doubler

The first experiments of splices in combination with doublers were carried out on edge cracked Glare 3 3/2 0.3 specimens. The reason for not using standard CCT specimens was to limit the amount of necessary material. The specimen dimensions are illustrated in Figure 7-16.

![Figure 7-16](image_url)

All specimens were loaded in the rolling direction and also the rolling direction of the doublers was oriented in loading direction. The crack stopper was a 3.2 mm thick doubler of aluminium 2024-T3 that was bonded to the Glare 3 sheet with AF 163. Besides two reference panels without splices, all specimens contained splices that were located in the area below the doubler. Three different splice distances were tested; 0, 7.5 and 20 mm. The intention was to locate the centre of the splice at the centre of the doubler area. However, after production it was observed that the splices in all panels shifted somewhat in the direction of the initial crack. The test matrix is given in Table 7-1.

Two identical specimens could be tested at the same time what helped also to avoid in plane bending. This is illustrated in Figure 7-17. Out of plane bending was avoided with A.B. guides. The crack extension was measured with a microscope, at the specimen side without doublers. The doublers were equipped with strain gauges. The tests were carried out by controlling the displacement of the clamping edges, and consequently the crack extension beyond the maximum load could be measured.
Table 7-1, Test matrix and results of simple spliced specimens with doublers.

| Skin material: | Giare 3 3/2 0.3 L-T, t = 1.4 mm |
| Doubler:       | Al 2024-T3, t= 3.2 mm & Wcs = 30 mm |
| Specimen geometry: | W = 300 mm, Edge crack, \(a_0 = 85\) mm |

<table>
<thead>
<tr>
<th>Nr of specimens</th>
<th>Splice distance</th>
<th>Residual strength [MPa]</th>
<th>Increase [%]</th>
</tr>
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<tbody>
<tr>
<td>2</td>
<td>No splice</td>
<td>234</td>
<td>0</td>
</tr>
<tr>
<td>2</td>
<td>0</td>
<td>266</td>
<td>13.7</td>
</tr>
<tr>
<td>2</td>
<td>7.5</td>
<td>256</td>
<td>9.4</td>
</tr>
<tr>
<td>2</td>
<td>20</td>
<td>250</td>
<td>6.8</td>
</tr>
</tbody>
</table>

Figure 7-17, Test set-up, doublers are located at backside of specimens.

In every test, the initial crack extended straight in the direction perpendicular to the external load, also after the doubler was passed. The reference panels with doublers but without splices showed only a minor crack stopping effect of the aluminium doubler, see Figure 7-18. The crack in these specimens became unstable after an average stable crack extension of 25 mm, so at a physical crack length of 110 mm, and a gross stress of 227 MPa. This was before the crack reached the doubler. When the crack reached the vicinity of the doubler, the doubler was able to arrest
the unstable crack and the gross stress could be increased with 3% to 234 MPa before final failure occurred.

Figure 7-18, Test results doubler panels with and without splices.

This crack stopping effect of the doubler was also observed for the spliced specimens. When the crack in these panels reached the first splice, a delamination zone was created around the crack and the cracked aluminium layers were pulled away from the intact aluminium sheet after the splice. This is illustrated in Figure 7-19. A large plastic zone is created in the spliced aluminium layer in front of the crack tip. Only after an extensive load increase the crack is reinitiated in the new layers at one of the edges of the plastic zone. With the onset of crack initiation, the maximum load was reached except for the specimens with \( s_d = 20 \) mm, see Figure 7-18. These specimens were investigated after the tests and it turned out that the splice area in these specimens was shifted towards the direction of the crack and that the first splice was actually located in front of the crack stopper. The first splice increased the maximum load considerably and the second splice was able to add some extra strength. Fracture of the doublers occurred for the panels without splices and the panels with \( s_d = 7.5 \) mm. All doublers were plastically deformed in the net section before failure of the specimens.

The panels with \( s_d = 0 \) mm showed the largest crack stopping capability, which is in agreement with the results of the test program presented in § 7.3.2. However, in the present test series the crack stopping effect decreased when increasing \( s_d \) which is different from the previous test results. Figure 7-18 gives the impression that the influence of the doubler
was limited while the influence of the splice was significant. However, the small crack stopping effect of the doubler might be enough to let the splice function as an effective crack stopping method.

Figure 7-19, Picture of fracture path in panels with splices and doublers.

Application of $s_d = 0$ may result in buckling problems when loaded in compression. It is also questionable what the splice will do when it is located further away from the crack tip. In a real fuselage structure, splicing each layer underneath a crack stopper and stiffening element would result in approximately 200 mm wide sheets in longitudinal direction and 500 mm wide sheets in circumferential direction. This is not applicable and therefore, splicing of a single aluminium layer underneath a doubler needs investigation.

7.4.2 Single splice below doubler

Because of the reasons discussed at the end of § 7.4.1, a new residual strength research has been carried out, comparable with previous test programs but with a splicing concept that is more applicable to real structures. The experiments have been carried out on 400 mm wide CCT specimens with $2a_0/W = 0.25$. The test matrices are given in Table 7-2 and Table 7-3. Because of a lack of laminates with 0.3 mm thick aluminium layers at the time of testing, most of the tests are carried out on Glare 3 3/2 0.2 material in the L-T direction. Besides 0.2 mm thick aluminium crack stoppers, also Glare 2 2/1 0.2 doublers have been investigated. These are
not only thicker (0.65 mm) but also significantly stronger. For the tested configurations, the doubler edges closest to the crack lie 15 mm in front of the initial crack and the splices are located 27.5 mm in front of the initial crack tips.

Table 7-2, Test matrix extended splice-doubler program Glare 3 3/2 0.2 L-T.

<table>
<thead>
<tr>
<th>Nr.</th>
<th>Spliced</th>
<th>Doubler</th>
<th>Geometry</th>
</tr>
</thead>
<tbody>
<tr>
<td>1-1</td>
<td>no</td>
<td>no</td>
<td></td>
</tr>
<tr>
<td>1-2</td>
<td>s_d = 155 mm</td>
<td>no</td>
<td></td>
</tr>
<tr>
<td>1-3</td>
<td>no</td>
<td>Al. 2024-T3, t = 0.2 mm, W_d = 25 mm</td>
<td></td>
</tr>
<tr>
<td>1-4</td>
<td>s_d = 155 mm</td>
<td>Al. 2024-T3, t = 0.2 mm, W_d = 25 mm</td>
<td></td>
</tr>
<tr>
<td>1-5</td>
<td>s_d = 155 mm</td>
<td>Glare 2 2/1 0.2, t = 0.65 mm, W_d = 25 mm</td>
<td></td>
</tr>
</tbody>
</table>

The residual strength results for the Glare 3 3/2 0.2 specimens are illustrated in Figure 7-20 and presented in comparison to a reference panel without splices and doublers. In contrast to the results discussed in § 7.4.1 where all the aluminium layers in the laminate were spliced in the same region, a single splice seems not to be able to arrest crack growth (compare the spliced specimen with the reference panel (+0.1%) and the spliced- and non-spliced specimens with doublers (-0.2%)). The relation between the gross stress and the crack extension plotted in Figure 7-22 illustrates that the splices have some crack stopping effect but only after the crack became unstable and the maximum load was reached.

In this test series, the doublers had a much larger influence on the residual strength but these were also located much closer to the initial crack tip. The largest residual strength increase is obtained with the stronger and thicker Glare 2 doubler. The aluminium doublers cracked along with the crack extension in the panel when the crack grew underneath the doubler. In contrast to this, the Glare doublers remained intact and the skin fractured. After the tests, all specimens were checked ultrasonically for delaminations.
Some minor delaminations were found between the doublers and the Glare skins and the delaminations of the Glare 2 doublers were somewhat larger compared to the aluminium doublers.

Figure 7-20, Test results Glare 3 3/2 0.3 L-T with splices and doublers.

Table 7-3, Test matrix extended splice-doubler program Glare 3 3/2 0.3 L-T.

<table>
<thead>
<tr>
<th>Nr.</th>
<th>Spliced</th>
<th>Doubler</th>
<th>Geometry</th>
</tr>
</thead>
<tbody>
<tr>
<td>2-1</td>
<td>$S_d = 155$ mm</td>
<td>no</td>
<td></td>
</tr>
<tr>
<td>2-2</td>
<td>no</td>
<td>Glare 2 2/1 0.2</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>$t = 0.65$ mm</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>$W_d = 25$ mm</td>
<td></td>
</tr>
<tr>
<td>2-3</td>
<td>$S_d = 155$ mm</td>
<td>Glare 2 2/1 0.2</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>$t = 0.65$ mm</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>$W_d = 25$ mm</td>
<td></td>
</tr>
</tbody>
</table>

Figure 7-21 illustrates the residual strength results for the Glare 3 laminates with 0.3mm aluminium layers compared with the laminates with 0.2mm aluminium layers. Again, the influence of the splice can be neglected while the influence of the Glare 2 doubler is significant. This is well illustrated by Figure 7-22 where the crack extension is plotted versus the gross stress. Although the residual strength of the Glare 3 3/2 0.2 material is lower
compared to Glare 3 3/2 0.3 (8.5%), see Figure 7-21, the use of the Glare 2 doubler compensates this completely. The load on the crack edges redistributed to the doubler and the ultimate load of the Glare doubler determined the residual strength. The influence of the splice is somewhat visible in Figure 7-22 and Figure 7-23 but comes too late to add any crack arresting capabilities.
After investigating the crack extension in the spliced specimens without doublers, the conclusion could be drawn that the splice in the outer layer is more effective to stop the crack than the splice in the centre layer. This due to fact that the intact centre aluminium layer is loaded by shear loads from two sides. The outside layer is loaded only at one side and this favours the occurrence of delamination. Due to delamination, high local stresses are redistributed along the delaminated aluminium layer and crack initiation is delayed. This larger delamination zone around the outside splice compared to the inside splice was proven with ultrasonic testing after the tests.

7.5 Splice configuration with internal doublers

The splice concept based on the SFT contains many different details, which can be divided in more or less three main details, see Figure 7-24:

- **the external doubler**, where an aluminium layer ends at the outside of the laminate,
- **the internal doubler**, where an aluminium layer ends within the laminate and
- **the simple splice with external doubler**, investigated in § 7.4
The internal splice differs from the internal doubler detail since the fibre-layers are not draped over the edge of the ending aluminium layers. The details identified with A differ slightly from detail A since the fibre-layers are deviating from the aluminium at two locations (A) in the laminate instead of one (A').

A = internal doubler configuration
B = external doubler configuration
C = internal splice

- = Aluminium
= Adhesive
= 0º-fibre layer
= 90º-fibre layer

Figure 7-24, Illustration of different details in splice with internal doublers.

In the present study towards the behaviour of this kind of splices the behaviour of the details has been investigated first and is presented in § 7.5.1 and § 7.5.2. Several static and dynamic tests were carried out on these details to investigate their failure mechanisms and weakest points. Subsequently, § 7.5.3 presents work that investigated the influence of neighbouring details on a specific splice detail in realistic splices. If there is no interaction, different splices can be designed based on the behaviour of the individual details. This section also discusses an investigation where cracks and holes were applied in the spliced specimens at the weakest points to investigate the blunt notch- and residual strength properties.

7.5.1 External doubler investigation

To investigate the behaviour of an external doubler under static and dynamic loading, a test program has been carried out on the specimens illustrated in Figure 7-25. A relatively thick 0.5 mm aluminium doubler is bonded with adhesive to a standard Glare 3 5/4 0.4 laminate to obtain a maximum stress concentration at the location of the doubler ending. The specimens contained tabs, designed in such a way that the load introduction itself caused no bending. Within the program, two different
adhesive layer thicknesses were applied to bond the doubler to the laminate since the autoclave cycle has the tendency to reduce the bondline thickness at the free end of the doubler. The small test matrix is given in Table 7-4.

Figure 7-25, External doubler specimen geometry.

Table 7-4, Test matrix external doubler research.

<table>
<thead>
<tr>
<th>Material</th>
<th>Experiments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3 5/4 0.4 with 0.5 mm 2024-T3 doubler (two different adhesive layer thicknesses)</td>
<td>3 Tensile test</td>
</tr>
</tbody>
</table>

Static tensile test results

The average ultimate strength of 3 Glare 3 5/4 0.4 specimens with outside doubler was 709 MPa, which is comparable to the static strength of standard Glare 3 5/4 0.3 material without doubler. During the tensile tests, the development of delamination was observed by microscope at the specimen sides. The following mechanisms were observed:

- At a gross stress of 275 MPa (39% of $\sigma_{ult}$) the doubler head separated from the adhesive fillet, see Figure 7-26 a).
- Between 300 and 350 MPa, delamination initiated between the doubler and the outer aluminium layer, see Figure 7-26 b).
- From 350 MPa up to failure, this delamination extended linearly with the applied gross stress. At the onset of fracture, the delamination extended over 12 mm.

It was observed that the delamination initiation occurred slightly later for a thicker adhesive thickness while the delamination growth was comparable.
Fatigue test results
During the fatigue experiments, the external doubler specimens were loaded with a CA loading spectrum. The loading consisted of two sequences:

- From 0 to 180 kcycles: $\sigma_{\text{max}} = 120 \text{ MPa}, R = 0.05 \text{ and } f = 10 \text{ Hz}$
- From 180 to 270 kcycles: $\sigma_{\text{max}} = 150 \text{ MPa}, R = 0.05 \text{ and } f = 10 \text{ Hz}$

Between 10 and 30 kcycles the doubler head released from the fillet in the same way as in the static test, see Figure 7-26 a). However, now the crack or delamination grew in a step like manner under 45° through the adhesive to the outer aluminium layer of the Glare laminate. During the second loading sequence at 150 MPa, after the initial 180 kcycles with $\sigma_{\text{max}} = 120$ MPa, a fatigue crack grew into the first aluminium layer of the laminate, see Figure 7-27a). This aluminium layer failed after 240 kcycles. Subsequently, delamination took place between the broken aluminium layer and the next fibre layer with fibres in loading direction, see Figure 7-27b). Large out of plane deflections were observed between the laminate and the broken aluminium layer. This illustrates that rather large peel stresses were acting on the adhesive.

Figure 7-26, Delamination mechanisms observed during static testing.

Figure 7-27, Delamination and fracture mechanisms observed during fatigue testing.
Summary
The external doubler does not reduce the static properties but at higher loads the doubler delaminated from the laminate. The delaminated area was a linear function of the applied load. During fatigue loading at $\sigma_{\text{max}} = 120$ MPa, the adhesive fillet released from the doubler head and a crack grew in the adhesive to the laminate. No delamination occurred. At an elevated stress of 150 MPa, the outer aluminium layer of the laminate cracked at the location where the cracked adhesive reached the aluminium layer. This cracked aluminium layer delaminated from the next load carrying layer during further fatigue loading.

7.5.2 Internal doubler investigation
The investigation towards internal doublers has been combined with an investigation of the behaviour of a ply-drop off. A ply-drop off is a reduction of the amount of layers in a Glare laminate, for example from a 4/3 lay-up to a 3/2 lay-up. Consequently, it includes an internal end of a prepreg layer, see Figure 7-28 a). Therefore, the internally ending aluminium layers in the test program are bonded to the outer aluminium layer with both prepreg and adhesive instead of only adhesive. The interrupted fibre layers ended well before the ending of the aluminium doubler and the gaps were filled with adhesive to obtain a better delamination resistance.

![Figure 7-28](image-url)

Figure 7-28, Illustration of ply-drop off.
The internal doubler and/or ply-drop-off have some advantages over the external doubler or ply-drop of:

- the load from the ending layer can be redistributed to two neighbouring and continuing layers instead of one,
- the bending moment reduces since the interrupted layers are moved towards the neutral axis of the laminate, and
- no moisture can reach the fibre layer since it is covered by the outside- and continuing aluminium layer.

Three different experiments have been carried out; tensile tests, blunt notch tests and fatigue tests. The general specimen geometry is illustrated in Figure 7-28 b). The specimens are symmetric and contain two ply-drop offs. This has the advantage that if one fails, the other will be in the situation just before failure and can be inspected afterwards. Three different thicknesses for the internal aluminium doubler were applied and tested; 0.3, 0.4 and 0.5 mm. The basic Glare 3 4/3 laminate contained aluminium layers of 0.4 mm.

**Tensile tests**

Table 7-5 contains the average measured tensile strength values for the different doubler laminates.

<table>
<thead>
<tr>
<th>Doubler thickness (mm)</th>
<th>0.3</th>
<th>0.4</th>
<th>0.5</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nr. of specimens</td>
<td>2</td>
<td>1</td>
<td>2</td>
</tr>
<tr>
<td>Static strength (MPa)</td>
<td>613</td>
<td>514</td>
<td>459</td>
</tr>
<tr>
<td>Reduction (^1) (%)</td>
<td>4</td>
<td>20</td>
<td>28</td>
</tr>
</tbody>
</table>

\(^1\) Reduction compared to standard Glare 3 4/3 0.4 L (\(\sigma_{ult} = 640\) MPa)

During the tensile test, a local grid was bonded to one of the outer aluminium layers, which was bended over the doubler head. With the imaging system and software package Impress that were developed within this study, see chapter 4) it was possible to measure the strain distribution in this grid and thus of the outer aluminium layer at the location of the internal doubler edge. The results are plotted in Figure 7-29 for three different external loads. The vertical axis represents the strain and therefore the level or height of the surface is an indication of the strain. The strains are presented as \(\Delta l/l_0\). The \(x\) and \(y\) axis represent the labelled locations of the strain measurement. As a help to the reader a schematised specimen lay-up is plotted in the figure that estimates with the positions along the y-
Figure 7-29 a) shows the strain distribution at an applied stress of 130 MPa. The strains are still small (between 0 and 0.5%) and the figure illustrates the scatter or inaccuracy of the method. The right side of the figure represents the thicker Glare 3 5/4 0.4 laminate and the left side represents Glare 3 4/3 0.4 after ply drop-off. Approximately at this stress level of 130 MPa, the adhesive fillet released from the doubler edge.

a) Strain distribution at 130 MPa
Blunt and sharp notch behaviour of spliced laminates

b) Strain distribution at 380 MPa

c) Strain distribution at 510 MPa

Figure 7-29, Strain distribution internal doubler\textsuperscript{11}. 

\textsuperscript{11} Reference to the source of the figure.
At higher loads, see Figure 7-29 b), the difference in stiffness (EA) becomes apparent and causes higher strains in the thinner laminate. At a gross stress between 340 and 380 MPa, delamination is initiated between the doubler and the continuing 0º fibre layer. This delamination grows until fracture of the specimen.

Figure 7-29 c) illustrates the occurrence of local necking at large strains, at the doubler end. This is the location where the specimens fail. A thicker aluminium doubler decreases the static strength. This is expected because of the larger step that the covering layer must bridge. The reduction is rather severe; from 4% for the 0.3 mm doubler to 28% for the 0.5 mm doubler.

**Fatigue tests**
The CA loading spectrum for the fatigue tests program is illustrated in Figure 7-30. The specimens were first tested for 180 kcycles at $\sigma_{\text{max}} = 120$ MPa. Subsequently, $\sigma_{\text{max}}$ was increased to 150 MPa for another 90 kcycles. Finally, $\sigma_{\text{max}}$ was increased to 200 MPa for 130 kcycles to obtain a total of 400 kcycles. The stress ratio was 0.05 and the frequency 10 Hz.

![Fatigue loading spectrum](image)

During the first 180 kcycles no damage occurred. Even after 270 kcycles tested at the raised stress level no damage to the specimens was observed by microscope. After about 20 kcycles at $\sigma_{\text{max}} = 200$ MPa, the adhesive fillet of one specimen released locally from the edge of the inside doubler and was completely released form edge to edge after 60 kcycles. The other 4 specimens did not show any other damage. After 400 kcycles, the
specimens were inspected by chemically etching away the outer aluminium layers. Again, no signs of damage were observed. The internal doubler configuration seems to be insensitive for fatigue loading.

**Blunt notch tests**

Blunt notch tests have been carried out on identical internal doubler specimens. The blunt notches with a diameter of 4.9 mm were drilled at the location where the internal doubler ends in the laminate, identified as the weakest location for static loading. Figure 7-31 illustrated the blunt notch results for different doubler thicknesses and compared with the un-notched tensile strength results.

![Blunt notch results internal doubler](image)

Figure 7-31, Blunt notch results internal doubler.

The thickness of the doubler has a clear influence on the static strength of specimens with inside doublers. This influence is not apparent for the blunt notch strength of specimens with inside doublers. The notch is the most critical parameter in this configuration. Any influence of the difference in doubler thickness has disappeared. The notch dominates the reduction in strength of the specimen. No difference was observed between the blunt notch strength of a basic laminate or with an internal splice.

Figure 7-32 illustrates the strain distribution around the notch just before failure. Again, the higher strains are visible in the thinner laminate but also
the strain increase due to the presence of the blunt notch. The grid used in this test differed from the one used in Figure 7-29; this one was drawn manually on the specimen and the distance between the cross-points was approximately 1 mm instead of 0.1 mm. For that reason, the local necking effect is not visible in this strain distribution.

Figure 7-32, 3D strain distribution around the notch on the side of the doubler right before failure (460 MPa)\(^1\).

Summary
The static strength of the internal doubler detail reduced from 4% for a 0.3 mm thick doubler up to 28% for a 0.5 mm doubler. Whether this is acceptable depends on the importance of the static strength in the design allowables. Fracture is initiated in the covering aluminium layer at the location of the internal doubler ending that shows local necking at high loads. The fatigue properties of this detail are not influenced within a normal lifetime of an airplane. The static blunt notch properties are not influenced by the presence of an internal doubler ending.
7.5.3 Splice with internal doublers

With the knowledge of the static and dynamic splice detail behaviour separately, a large test program has been carried out on specimens with the splice configuration illustrated in Figure 7-33. This represents a realistic splice configuration and contains the splice details discussed before. The experimental program contained tensile tests, fatigue tests, blunt notch tests, and residual strength tests. The base-line materials were Glare 3 4/3 0.4 and Glare 3 3/2 0.4.

Figure 7-33, Splice configuration splice with internal doublers.

Tensile test

The tensile tests were carried out on dog bone specimens and contained a splice perpendicular to the loading direction. The Glare 3 4/3 0.4 tensile specimens were tested in L and T direction and originated from the sheets with the residual strength specimens since these tensile data are used for the residual strength analyses. Figure 7-34 illustrates the location of the tensile specimens with the splice in T direction. The unspliced tensile specimens cut from the sheets are used as reference. The Glare 3 3/2 0.4 laminate was only tested in L direction.

The stress-strain curves for the Glare 3 4/3 0.4 specimens are illustrated in Figure 7-35 and are based on the laminate thickness outside the spliced area. The curves show that the Young’s moduli are comparable with the reference laminates. The yield stress for the spliced specimens tested in the T direction is higher compared to the reference panels. The same trend was observed for the spliced Glare 3 3/2 0.4 specimens. The yield stress for the spliced Glare 3 4/3 0.4 laminates tested in L direction are equal to the values for the non-spliced specimens.
The static strength of the Glare 3 4/3 0.4 laminates is reduced with 15% and 25% for the specimens tested in T and L-direction respectively. It is expected that the larger reduction in L-direction is caused by a lay-up mistake during production, which resulted in a gap (x) of 10 mm instead of 0
Blunt and sharp notch behaviour of spliced laminates

mm between the doubler heads, see Figure 7-33. This strength reduction is not dramatic since the ultimate load is not determined by the ultimate stress of the laminate but more likely by the blunt notch strength. The Glare 3 3/2 0.4 spliced specimens with a correct lay-up also illustrated a 20% ultimate strength reduction.

During the tensile test, the internal doubler ends were released from the adhesive fillets and subsequently delamination occurred between the doublers and the first fibre layer oriented in loading direction, see point 1 in Figure 7-36. Delaminations were also observed below the external splices (point 2). In the Glare 3 4/3 0.4 specimens, fracture was initiated in the covering aluminium layer at the end of the doubler head (point 3) All these mechanisms are comparable with the results for the internal and external doublers.

The specimens with the larger gap x showed the same delamination mechanisms but fracture was initiated at point 4. Since these specimens were tested in the L direction, the load carrying fibre layers are shifted to the centre of the laminate. Due to the lay-up error in these specimens, the next splice was shifted to the doubler end. Consequently y in Figure 7-36 was reduced to approximately 10 mm. Delamination from the doubler head and the centre splice grew to each other and caused an early failure of the laminate.

In the Glare 3 3/2 0.4 specimens, not the outer- but the inner aluminium layer fractured first at location 5. Subsequent fracture of the laminate occurred at a lower gross stress at location 4.

Fatigue tests
Two CA fatigue tests were carried out on Glare 3 3/2 0.4 laminates. The specimens were fatigue loaded for 270 kcycles with $\sigma_{\text{max}} = 120$ MPa, $R = 0.05$ and $f = 10$ Hz. After 5 kcycles, the adhesive fillet in the gap at point 2
in Figure 7-36 released from the edge of the outer aluminium layers. During the remaining of the tests, only some extension was observed into the underlying adhesive layer. Further, no more damage was found. This mechanism was also observed during the external doubler research in § 7.5.1.

After this fatigue loading, the specimens were statically tested. The specimens behaved in an identical manner as the tensile specimens discussed above, resulting in no reduction of the yield- and ultimate strength.

Blunt notch tests
The blunt notch tests were carried out on Glare 3 3/2 0.4 material. The notches with a diameter of 4.9 mm were positioned at the most critical locations, determined during the previous tensile tests. Two different locations were selected and illustrated in Figure 7-37.

In an additional test program, the blunt notch of position 1 and the location of the outside splice have been filled with NAS 1097 DD 6 countersunk rivets that were installed with a squeeze force of 24 kN. Both filled holes were located in one specimen as illustrated in Figure 7-37. This specimen has been fatigue tested for 180 kcycles with a CA fatigue spectrum with $\sigma_{\max} = 120$ MPa, $R = 0.05$ and $f = 10$Hz.
The countersunk head of the rivets has been placed in an aluminium strip, which was riveted to the specimen because creating a countersunk hole into the laminate would imply extra damage. Only the influence of the filled hole is regarded. The aluminium strip does not influence the test results, because the rivets do not transfer any load.

The blunt notch results are comparable with the blunt notch strength for a Glare 3 3/2 0.4 laminate. This is the same result as presented in § 7.5.2 for the internal doubler configuration. Since the internal doubler configuration was critical for static loading, the results are expected. Fracture occurred in the net section where the notch was located.

During fatigue testing of the riveted specimen, no crack initiation occurred at the hole or at any other location during the 180 kcycles. Additionally, the specimen was static loaded to failure and the strength and failure mechanism was comparable as for the blunt notch specimens.

Residual strength tests
The residual strength program was split in two test programs:

- experiments on 400 mm wide Glare 3 4/3 0.4 CCT specimens and
- experiments on 800 mm wide Glare 3 3/2 0.4 specimens.

400 mm wide Glare 3 4/3 0.4 specimens
Two 400 mm wide CCT specimens have been tested with the intention to give a first impression of the influence of the splice with internal doublers on the residual strength properties of the laminate. The specimen geometries are illustrated in Figure 7-38.

The first CCT specimen, see Figure 7-38a) contained a splice perpendicular to the rolling- and loading direction. Due to a lay-up mistake this splice contained a gap of 10 mm, see also § 7.5.3. The crack was placed at the failure location determined in the tensile test program. The initial crack length was 100 mm. This test geometry is comparable with the first residual strength test that was carried out on simple splices, discussed in 7.3.1, that showed in general a residual strength reduction.

The second CCT specimen, see Figure 7-38 a), contained a splice in rolling- and loading direction, located in the centre of the specimen. The idea behind this test is that a splice with overlap or internal doubler adds
some stiffness to the specimen and this extra material is cracked, it will tend to open the crack and possibly reduce the residual strength. To obtain a fully cracked splice area, the initial crack was 133 mm.

Figure 7-38, CCT geometries for 400 mm wide spliced Glare 3 4/3 0.4 L-T specimens.

Figure 7-39, $K_I$ curves based on compliance correction for spliced panels.
Blunt and sharp notch behaviour of spliced laminates

Because the initial crack lengths differ and no residual strength data are available for 400 mm wide unspliced Glare 3 4/3 0.4 laminates, the $K_R$-curves will be compared. This is illustrated in Figure 7-39.

Based on the $K_R$-curve results, the presence of a splice in loading direction or perpendicular to the loading direction has no influence on the residual strength. The local area increase has not been taken in to account in the determination of the curves. It can be concluded that the splice with internal doublers is does not reduce the residual strength compared of the material that surrounds the splice, neither for cracks in the direction of the splice nor perpendicular to the splice.

800 mm wide Glare 3 3/2 0.3 L-T specimens
Additionally to the previous residual strength program, tests have been carried out on larger 800 mm wide specimens, see Figure 7-40. The material was Glare 3 3/2 0.4, tested in rolling direction.

Figure 7-40, CCT geometries for 800 mm wide spliced Glare 3 3/2 0.3 L-T specimens.
The splices were identical to the splice configuration for the 400 mm wide specimens but in all splices the gap width $x$ between the two doublers was set to 5 mm. One specimen contained a centre splice identical as in the previous tests while the other specimen contained two splices in front of the crack tips. Both panels contained an initial crack of 200 mm.

Figure 7-41 shows the measured crack extension as a function of the gross stress for both panels and some unspliced Glare 3 3/2 0.3 reference panels. Unfortunately no reference data were available for Glare 3 3/2 0.4 specimens. The residual strength of the panel with centre splice is comparable with the reference panels. The panel with splices in front of the crack shows a higher residual strength. This is attributed to the local doublers in the splice and not to the presence of the splices.

Figure 7-41, Crack extension versus applied gross stress for 800 mm wide Glare 3 3/2 0.4 specimens with internal splices.

### 7.6 Conclusions

**General**
The splicing concept for Glare allows the production of large sheets. The sheet size is no longer limited by the available aluminium sheet width from the aluminium supplier but by the dimensions of the available autoclave. The application of large sheets as skin material for fuselage structures offers several advantages like a weight decrease, lower assembly costs.
and reduced inspection and maintenance costs. The splicing concept has been developed over the last years from

- a simple splice configuration without doublers, to
- a simple splice configuration covered with external doublers, to
- a splice configuration with internal doubler and
- a splice configuration with overlaps

The driving parameters in this development were

- No significant reduction of material properties like ultimate strength, yield strength, blunt notch strength and fatigue behaviour.
- Maintaining the durability of the material.
- Minimum weight increase,
- Minimum impact on the production process, and
- A flat outer skin and no interference of the inner skin with the locations of stiffeners, frames, doublers and joints.

The splicing configuration is now a mature solution with no stiffness, yield strength, blunt-notch and residual strength reduction and the ingress of moisture is almost eliminated.

Based on the research on the different splice configurations, the following conclusions can be drawn.

**Simple splice**

In case of a simple splice, the remaining unspliced layers of the laminate bridge the spliced aluminium layers. The laminate is not reinforced and the effective load carrying cross section over a splice is reduced. This reduces the static properties of the laminate perpendicular to the splice. The splice configuration is described by the splice distance and the splice pitch.

Two crack scenarios have been investigated for the residual strength:

1. Splice with crack perpendicular to the loading direction, and
2. Splices in loading direction, positioned in front of the crack that was oriented perpendicular to the loading direction.

Crack situation 1: In this situation, the effect on the residual strength depends on the splice distances between the splices in the different layers; a splice distance of zero has a positive influence while a larger splice distance reduces the residual strength. However, the application of a zero splice distance is not realistic since it represents the situation that all the aluminium layers are spliced at the same location. This results in a large
effective cross section reduction and reduces the static tensile and compression properties significantly.

Crack situation 2: If the splices are located in front of an existing crack, they add crack arresting capabilities to the Glare sheet and increase the residual strength. The residual strength increase was equal for the investigated splice distances investigated but decreases if the distance between the crack tip and the first splice gap increases.

Summarized, the simple splice will reduce the static and residual strength properties of the glare sheet. Additionally, this splice concept is very sensitive for moisture ingress.

Splice with external doubler
To improve the static, tensile and durability properties of the simple splice, it can be covered with an aluminium doubler. This research has investigated several different concepts:

Bonding of the aluminium doubler with prepreg layers resulted in poor delamination behaviour under static loading perpendicular to the doubler. This was solved by using AF-163 only without fibres.

Two splice concepts with external doublers have been investigated; one aluminium layer spliced below one doubler or all aluminium layers spliced below one doubler (full splice). The doublers and splices were oriented in loading direction and positioned in front of the crack. When the splices in the different aluminium layers are moved from each other and placed under different stiffening elements ($s_d > 150$ mm), no crack stopping effect was observed anymore. This in contrast to the full splice that resulted in a significant residual strength increase while the crack stopping effect of the 3.2 mm thick aluminium doubler was negligible. Applying a Glare 2 doubler turned out to be significantly more effective in increasing the crack arrest capability than applying an aluminium doubler.

Although the splice with external doubler concept solved the reduction of the static properties as far as they were investigated, this configuration has the large disadvantage that the skin is no longer flat at the outside.

Splice based on SFT technique including internal doublers
The production of a splice configuration with a flat outer skin and good durability properties is solved with the introduction of the Self Forming
Technique (SFT). Within this concept the splice configuration can contain internal doublers and/or overlapping aluminium layers.

To allow a more general design of splice configurations of these kind, several splice details have been isolated from this configuration:

- the external doubler, where an aluminium layer ends at the outside of the laminate,
- the internal doubler, where an aluminium layer ends within the laminate and
- the simple splice with external doubler

A research has been carried out to investigate the behaviour of these details separately, followed by a research to investigate their interaction in a real splice.

The behaviour of the simple splice with external doubler was described above.

The external doubler did not reduce the static properties but at higher loads the doubler delaminated from the laminate. The delaminated area was a linear function of the applied load. During fatigue loading for 180 kcycles at $\sigma_{\text{max}} = 120$ MPa, the adhesive fillet released from the doubler head and a crack grew in the adhesive to the laminate. No delamination occurred. At an elevated stress of 150 MPa, the outer aluminium layer of the laminate cracked at the location where the cracked adhesive reached the aluminium layer. This occurred after an additional amount of 60 kcycles and the cracked aluminium layer delaminated from the next load-carrying layer during further fatigue loading. Therefore it is concluded that this splice detail is not critical for the normal use in an aeroplane fuselage.

The static strength of the internal doubler detail reduced from 4% for a 0.3 mm thick doubler up to 28% for a 0.5 mm doubler. Fracture is initiated in the covering aluminium layer at the location of the internal doubler ending that shows local necking at high loads. The fatigue properties of this detail are not influenced within a normal lifetime of an airplane. Also the static blunt notch properties are not influenced by the presence of an internal doubler ending.

The research on the total splice based on SFT technique including internal doublers showed that the properties of this splice design can well be described and predicted based on the static and dynamic behaviour of the
different splice details. The stiffness, yield stress, blunt notch strength, fatigue properties and remaining strength after fatigue of a splice with and without filled holes are not decreased compared to non-spliced reference laminates. Also the residual strength of a spliced panel with a through crack at the most critical locations is not decreased. The only property that decreased over the splice is the static strength, which is no critical design parameter for the ultimate load.

Based on these results, it can be concluded that a spliced panel based on the Self-Forming Technique with internal doublers does not influence the design allowables and adds extra advantages to the design of a large capacity aeroplane in Glare.
References


8 RESIDUAL STRENGTH IN COMBINATION WITH RIVET HOLES AND FATIGUE DAMAGE

8.1 Introduction

As discussed in chapter 2, an aeroplane is designed to sustain certain damage either due to fatigue loading, corrosion or impact of discrete sources such as from high-energy engine fragments during disintegration. The damage capability or tolerance is verified in full-scale tests. Figure 8-1 illustrates a schematic full-scale barrel that can be cyclically loaded with internal pressure, tension or compression, bending and torsion, comparable with in service loading conditions.

Figure 8-1, Barrel test set-up DCAA, Hamburg.

To investigate the damage tolerance capability, the full-scale tests are equipped with artificial damages like scratches and manufacturing defects. Investigation of fuselage damage due to an engine disk fragment is
simulated with a harpoon blade that is fired into the pressurised cabin. These tests can also provide information about continued operation of the aeroplane beyond the DSG, which increases the possibility that WFD will occur in the pressure cabin and degrades the damage capability\(^1\).

MSD in combination with a lead crack is, and has been investigated extensively for aluminium structures after the well-known Aloha accident. It is likely that the behaviour of a Glare skin with MSD is different. The large lead crack in aluminium skin structures is generated through link up of many existing fatigue cracks in the same rivet row. In case of a Glare skin, fatigue cracks will only occur in aluminium layers and the crack growth is much slower than in aluminium because of the intact fibres that bridge the crack. It is questionable whether the occurrence of MSD in Glare will turn into WFD in the operational lifetime of an aeroplane. To investigate the occurrence of MSD in Glare, a small initial test program was carried out to determine the fatigue behaviour of Glare in case of open- and filled holes. This research is described in \(\S\) 8.2 and contains a discussion about the likelihood of the occurrence of MSD and WFD in Glare.

A different damage scenario for Glare is the occurrence of a Foreign Object damage (FOD) that creates a through the thickness crack. The most critical location for a FOD to occur is at the location of a rivet row, for example in a single rivet row that connects a stringer or clip to the skin or in multiple rivet rows, as they exist in butt- or lap joints. Although the history is different, the cracked situation is somewhat comparable with MSD in an aluminium skin after link-up of some of the fatigue cracks. The probability that this damage scenario occurs, \textit{FOD within a rivet row}, is so small that it is no requirement for any aeroplane\(^2\).

However, based on several questions from the aerospace industry, this scenario has been investigated within this research. Therefore, residual strength tests on Glare were performed simulating a FOD in rows of open holes, filled holes and lap joints. All configurations were regarded both with and without MSD. The purpose of the test program was to investigate the residual strength reduction for this situation. To be able to predict the residual strength in the presence of WFD, \(\S\) 8.3 presents and discusses several prediction methods that are available for aluminium. \(\S\) 8.4 presents the research carried out for Glare and validates the prediction methods discussed in \(\S\) 8.3 with the experimental results.

This chapter finishes with a summary and conclusions, given in \(\S\) 8.5.
8.2 Widespread Fatigue Damage

To investigate the influence of MSD in front of a FOD on the residual strength, first the occurrence of MSD in Glare is investigated and discussed in this paragraph.

8.2.1 Open holes in Glare versus aluminium

Roebroeks\textsuperscript{3} compared the fatigue behaviour of multiple hole specimens of 1.38 mm thick Glare 3-3/2-0.3 sheets with 1.25 mm Alclad 2024-T3 sheets. Two kinds of specimen configurations were tested; a 3 hole and a 7 hole specimen. The specimen geometry and test practice are given in Figure 8-2.

\begin{figure}[h]
\centering
\includegraphics[width=0.5\textwidth]{specimen.png}
\caption{Specimen geometry and test practice\textsuperscript{3}.}
\end{figure}

The Glare specimens failed only after all the aluminium layers were cracked by fatigue loading over the full specimen width. At this point the fibres were still intact and were carrying the load for another several thousand cycles. Test results showed that the fatigue life of Glare (42000 cycles) is almost twice as long as for the 2024-T3 specimens (21600 cycles). For monolithic aluminium 80-90 % of the fatigue life consist of initiation of the micro crack and micro crack growth. In Glare, crack initiation occurs earlier due to

- the residual stresses after curing that result in a tensile stress in the aluminium layers at temperatures below the curing temperature, and
- the higher stiffness of the aluminium layers compared to the prepreg layers which attracts load.
However, as soon as fatigue cracks are present in the aluminium layers, the fibres start to bridge the crack. As a result, the largest part of the fatigue life in Glare consists of the growth of the macro crack.

8.2.2 Glare with open holes versus filled holes

To obtain more information about the occurrence and importance of MSD in Glare, a fatigue test program was carried out on small coupon specimens with either a row of open holes or a row of holes filled with rivets. Some of the specimens were equipped with small initial flaws at both sides of the holes to represent manufacturing defects. The test matrix with the test program and the specimen geometry are given in Table 8-1.

Table 8-1, Test matrix- and procedure for coupon tests program.

| Fatigue loading: CA, $\sigma_{\text{max}} = 120$ MPa, $R = 0.05$ and $f = 10$ Hz |
|---|---|
| Specimen dimensions: | Rivet data: |
| Width = 160 mm | • NAS 1097 DD (countersunk) |
| Length = 300 mm | • $\varnothing = 5.6$ mm |
| $\varnothing$ holes = 5.7 mm | • Pitch = 28 mm |
| tested in T direction | • Squeeze force = 24 kN |

| Geometry: Only holes, no artificial damages | Open holes | Filled holes$^{1)}$
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare3-3/2-0.4</td>
<td>ID = O</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Geometry: Holes contained one or more initial flaws. These flaws were through-the-thickness flaws with a length of 0.5 mm. They were always positioned at both sides of a hole.</th>
<th>Glare3-3-2-0.3</th>
<th>Glare3-3-2-0.4</th>
</tr>
</thead>
<tbody>
<tr>
<td>ID = O-sf</td>
<td>ID = F-sf</td>
<td></td>
</tr>
</tbody>
</table>

| Specimen ID: O = open hole, F = filled hole, sf = single flaw and mf = multiple flaw |

$^{1)}$Extra aluminium strip prevents presence of countersunk in Glare material
Residual strength in combination with rivet holes and fatigue damage

During testing, crack initiation (defined as the moment that the first crack growth was observed) occurred at crack lengths between 0.1 and 0.3 mm. Crack length for initiation and growth was recorded with a CCD camera.

Open hole specimens
Specimen O with open holes without initial flaws illustrated crack initiation in all holes after roughly 30 kcycles, see Figure 8-3. The crack initiation was found to be a function of the distance from the centre; the hole in the centre indicated with nr. 3 showed first crack initiation followed by holes 2 and 4, one pitch further to the specimen edge. After 98 kcycles, the crack extension in the centre hole was grown up to 10 mm. Up to a crack extension of 6 mm, the crack growth rate was constant and equal to 0.13 mm per kcycle. Beyond this crack extension, the crack growth rate increased, expected to be mainly due to crack interaction.

Figure 8-3, Crack growth data for Glare 3 specimen with open holes.

The specimen with initial flaws at both sides of the centre hole (specimen O-sf) showed crack initiation from the centre hole after a few load cycles. Crack initiation in the other holes without initial flaws occurred between 20 and 40 kcycles. The crack growth rate was comparable with the previous test with open holes only; 0.122 mm per kcycle. The crack initiation and crack growth will be discussed in more detail.
Chapter 8

Crack initiation
The stresses in the aluminium layers depend on the residual stresses due to the curing cycle and stiffness differences between the aluminium layers and the prepreg. If these are taken into account, the applied fatigue gross stresses of $\sigma_{\text{max}} = 120$ and $\sigma_{\text{min}} = 6$ MPa result in aluminium stresses of 160 MPa and 23 MPa respectively. In reality this high maximum stress will create plasticity at the notch edge ($K_t = 3$) what reduces the maximum stress. Not taking into account plasticity results in an $R$ change from 0.05 to 0.14. To obtain the average stresses in the net section, the gross stresses have to be multiplied with $p/(p-d) = 28/(28-5.6) = 1.25$ and are equal to 200 MPa and 29 MPa respectively.

Roebroeks\textsuperscript{5} calculated that the stress concentration factor $K_t$ for an infinite Glare 3 sheet with a round hole is equal to 3.07, see also eq. 3-4. This is equal to an increase of 2\% compared to the isotropic $K_t$ for this situation. The fact that multiple holes are present is accounted for in the equation of Heywood (eq. 3-7) by using the effective width or rivet pitch:

$$K_t = \frac{W}{W - d} \left(2 + \left(1 - \frac{d}{W}\right)^3\right)$$  \hspace{1cm} \text{eq. 8-1}

where $d =$ diameter hole = 5.6 mm and $W =$ specimen width or rivet pitch = 28 mm

Multiplying $K_t = 3.14$ according to eq. 8-1 with the correction for anisotropy (2\% $\equiv x 1.02$) results in $K_t = 3.2$.

The calculated average net stresses for the aluminium layers ($\sigma_{\text{max}} = 200$ and $\sigma_{\text{min}} = 29$ MPa) and the $K_t$-value can be used to predict the amount of cycles for initiation, using handbook data\textsuperscript{6}. This is illustrated in Figure 8-4 for a diagram with a $K_t$-value of 3.1. This $K_t$ is somewhat lower than the real value and will result in a prediction of a too large amount cycles before crack initiation. The prediction with Figure 8-4 results in an amount of load cycles before crack initiation of 40 kcycles, which is 30\% too high. Taking into account the too low $K_t$, the reduction of the maximum stress around the notch due to plasticity and the normal scatter, this method seems to give rather good prediction results.
Crack growth

The crack growth rate for the open hole specimens is comparable for all holes. Figure 8-3 also contains crack growth data from Kieboom\(^7\), obtained on single hole 50 mm wide specimens, which is slightly wider than two rivet pitches. The results for crack initiation and the constant crack growth after initiation are comparable. No influence of the width is present at that time.

For larger fatigue crack lengths, there is a clear increase of the crack growth rate visible. This is due to the presence of the other holes and indicates the onset of WFD. At this time the average crack originating from the hole is 6 mm. Including the hole diameter, the total crack length = 6 + 5.6 + 6 = 17.6 mm and with a pitch of 28 mm, only 10.4 mm of the aluminium material is still intact between two open holes. The interaction influence of WFD can be compared with the influence of the width for a specimen with a single hole. If the multiple hole specimens are compared with single hole specimens with an effective width equal to the rivet pitch, the relative crack length in this case is in the order of 17.6 / 28 = 0.63. This is illustrated in Figure 8-5. Results for fatigue crack growth of Alderliesten\(^8\), see Figure 8-6, show that the crack growth rate is initially constant until the relative crack length is more or less equal to 0.60. After that, the crack growth rate increases due to the presence of the specimen boundary.
Figure 8-5, Comparison MSD specimen with CCT specimen.

Figure 8-6, Crack growth data for finite width specimen.

Filled holes
When the holes are filled with squeezed rivets, the crack initiation and crack growth process changes significantly. Specimen F without initial flaws was fatigue cycled for 350 kcycles under the same conditions as the open hole specimen. Subsequently, a tear down inspection was carried out to find cracks in the holes or at the mating surface. None were found either visually or with an optical microscope. Specimen F-sf with two 0.5 mm long starter notches at both sides of the centre hole and specimen F-mf with 0.5 mm starter notches at all holes were fatigued for 180 kcycles. Both specimens showed no sign of crack initiation from these initial flaws during a tear down inspection.

Thus it can be concluded that open holes in Glare are sensitive for fatigue crack initiation clearly within the lifetime of an aircraft and that the presence of initial flaws advances initiation. However, when the holes are filled with
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Rivets, no crack initiation is found after at least 180 kcycles, even in the presence of initial flaws.

Based on these results, it can be concluded that the occurrence of WFD in Glare at the location of a row of rivets is not likely to occur during the lifetime of an aircraft, not even if all the holes contain initial flaws. Thus, the situation that MSD results in a large lead crack due to link-up like for aluminium is not realistic for the tested Glare configurations. The tests do not give any results for the situation that the rivets transfer loads in the direction of the external load like for example in a lap- or butt joint. This is discussed in 8.2.3.

8.2.3 Lap joints

In the case of a riveted joint, the rivets have to transfer load. There are three stress-raising contributions that affect the fatigue life of a rivet row, which are illustrated in Figure 8-7:

- a) Stress concentration caused by the by pass loads,
- b) Stress concentration on the hole edge caused by pin loading of the rivet, and
- c) Secondary bending.

![Figure 8-7, Illustration of stress concentrations due to a riveted joint.](image)

In case of an unloaded rivet, the last two contributions are not present. The stress concentration in case of a loaded rivet will therefore be higher.
Furthermore, fretting of the mating surfaces is the source for fatigue nuclei. These nuclei will appear in the area of the higher stress concentration.

Where unloaded rivets under fatigue loading did not cause any fatigue cracks, lap joints under fatigue have shown to induce fatigue cracks. Muller\(^9\) carried out fatigue tests on a series of Glare 3-3/2-0.3 riveted lap joints with three rows of rivets. The applied squeeze force for rivet installation was 22 kN and the lap joints were fatigue tested at $\sigma_{\text{max}} = 125$ MPa and $R=0.05$.

Illustrative results of crack growth measurements in the riveted lap joints are shown in Figure 8-8. The results confirm that crack growth in a Glare lap joint starts as part-through cracking. The first fatigue crack initiated in the aluminium layer at the mating surface. Crack initiation in Glare is not troublesome since most of the fatigue life consists of extremely slow crack growth and crack initiation is only a small portion of the total fatigue life. After a crack extension of approximately 4 mm and 420 kcycles, the crack growth in the mating aluminium layer increased. At that time the net section of this layer was reduced with more than 50%. The increased crack growth rate is still less than in monolithic aluminium joints.

After 600 kcycles, crack initiation occurred in the second layer. This initiation only started after complete link up of all the cracks in the first layer.

![Figure 8-8](image)

**Figure 8-8**  Crack growth in a Glare lap joint in the countersunk sheet at the upper rivet row\(^9\). Layer numbering as indicated and crack length is measured from hole edge.
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In aluminium joints the crack will start most likely as a corner crack and grow both in the direction perpendicular to the loading direction and through the thickness of the aluminium sheet.

For Glare the situation is completely different. For crack growth through the thickness, the crack must re-initiate in the subsequent layers while the intact fibres bridge the crack. Therefore the crack free life is larger in these subsequent layers while the intact layers delay the crack growth.

The above results show that crack initiation and growth from loaded rivets in a Glare lap joint can occur during the DSG of an airplane. However, fatigue cracks in all aluminium layers and link up of these cracks is not likely to occur at all, due to the long crack free life of the subsequent layers and the extreme slow crack growth.

The situation might differ if the lead cracks are caused by FOD instead of MSD. FOD can cause a large, through-the-thickness crack. The influence of MSD on a FOD will be discussed in the following paragraph.

8.3 Prediction methods

This paragraph discusses several analytical methods described in literature, to predict the failure load of aluminium panels with a lead crack and MSD. Due to the presence of holes, with or without fatigue cracks in front of a large lead crack, the stress intensity at the crack tip of the lead crack is changed. This is discussed in § 8.3.1. The paragraphs 8.3.2 to 8.3.4 present and discuss several failure criteria used for aluminium.

8.3.1 Stress intensity for large crack in row of rivet holes

For the interaction between a lead crack and a neighbouring MSD cracks several stress intensity factor solutions are available. In this thesis, the solutions of Yokobori are used, given in several stress intensity handbooks. This solution should be exact and is valid for a situation with two collinear through-the-thickness cracks of unequal length. Figure 8-9 illustrates the two collinear cracks configuration, where the smaller crack is presented as a hole with two cracks. Within the remainder of this chapter, the Yokobori solution for two cracks is assumed to be representative also for one crack and a crack that originates from a hole. For Glare this is slightly supported by the results presented in chapter 2 that illustrated the
fast decreases of the notch strength as a function of $K_t$. If $2b = d$, only a hole without fatigue damage is present.

**Figure 8-9, Notations for two, unequal through the thickness cracks.**

The stress intensity increase at the lead crack tip 1 according to Yokobori is given by:

$$K_{1-\text{MSD}} = \beta_1 \cdot K_1$$

with

$$\beta_1 = \sqrt{1 + \frac{2b}{t}} \left( 1 + \frac{1}{2a} \right) \frac{K(k) - E(k)}{E(k)}$$

**eq. 8-1**

where:

- $K_1$ = Stress intensity at crack tip 1 without presence of 2$^{nd}$ crack.
- $\beta_1$ = correction factor for presence of second crack.
- $a$ = half lead crack length.
- $b$ = half MSD crack length.
- $t$ = ligament length.
- $K(k)$ = Complete elliptical integral of the first kind
- $E(k)$ = Complete elliptical integral of the second kind.

With the following additional relations:

$$k = 2 \sqrt{\frac{ab}{(2a + t)(2b + t)}}$$

and

$$K(k) = \frac{\pi}{2} \left( 1 - k^2 \sin^2 \phi \right)^{\frac{1}{2}} d\phi$$

$$E(k) = \frac{\pi}{2} \left( 1 - k^2 \sin^2 \phi \right)^{\frac{3}{2}} d\phi$$
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The integrals are a function of k only, which is a geometry parameter. The two integrals were calculated using a first-degree approximation giving an appropriate accuracy. Figure 8-10 illustrates the factor \( \beta \) as a function of the distance \( t \) for a certain lead crack and MSD crack configuration. It is clear that the influence of a small MSD crack in front of a lead crack has a stress raising effect. However, this effect becomes only significant if the distance between the two is smaller than half a regular rivet pitch \( p \).

Figure 8-10, Correction factor for presence second collinear crack.

### 8.3.2 Net ligament loss or net section yield criterion

The net ligament loss criterion predicts failure based on the amount of material available in the net section to carry the external load\(^{15}\). For a monolithic aluminium structure without extensive strain hardening, this would mean that the failure load is a function of the aluminium yield strength and the amount and sizes of the flaws present. In the case of Glare with its remarkable strain hardening, failure would be based on a value between the yield stress and the ultimate strength of the laminate. The net section width of the specimen with a lead crack and WFD is defined as:

\[
W_{net} = W - 2a - n \cdot d - \sum_{i=1}^{n} (a_{ri} + a_{ri}) \quad \text{eq. 8-2}
\]

where:  
\( W = \) gross width  
\( 2a = \) lead crack length  
\( n = \) amount of holes  
\( d = \) hole diameter
af = initial flaw size plus fatigue crack extension (l = left, r = right)

With this definition of \( W_{\text{net}} \), the residual strength according to the net ligament loss criterion becomes:

\[
P_{\text{net}} = \alpha \cdot \sigma_y W_{\text{net}} \quad \text{where} \quad \alpha = 1 \quad \text{for aluminium and} \quad 1 \leq \alpha < \frac{\sigma_{\text{ult}}}{\sigma_y} \quad \text{for Glare}
\]

Based on the results presented in § 3.6.1, it is expected that this criterion is very much dependent on the geometry of the specimen. Results obtained by Cherry\textsuperscript{15} confirm this. For narrow 229 mm wide specimens of 2024-T3, the residual strength was under predicted by 9% while the residual strength of 381 mm width was over predicted by 7%. However, to compare the residual strength of specimens with the same overall geometry, the use of the net section width can be more illustrative than the use of the gross width. This is illustrated in §8.4.

### 8.3.3 Ligament yield criterion

Figure 8-11 illustrates the Feddersen residual strength criterion for CCT specimens as a function of the physical crack length\textsuperscript{16}. The curved line is determined by the critical stress intensity factor \( K_c = \sigma \sqrt{\pi a} \). Based on experimental results for aluminium alloys, Feddersen observed that \( K_c \) predicts the residual strength as long as it remains below \( 2/3 \sigma_y \) and the physical crack length is smaller than \( W/3 \). For stresses above \( 2/3 \sigma_y \) and cracks larger than \( W/3 \), the experimental results followed tangent lines from \( \sigma_y \) on the vertical axis and \( W \) on the horizontal axis to the \( K_c \) curve.

The straight lines in this figure that connect \( \sigma_y \) with \( W \) represent lines of NSY for a specific width. For aluminium materials that hardly show any strain hardening, these lines represent failure due to general yielding of the material. Only in that region where the \( K_c \)-curve lies below the NSY line, failure is described by \( K_c \).

In the case of multiple cracks (or MSD), the effective panel width is significantly reduced to the crack pitch, \( W_1 \). According to the Feddersen this would change the failure mechanism from fracture to NSY. Therefore, it appears reasonable to assume that link-up of a lead crack with an MSD crack in an aluminium sheet would occur when the ligament stress between the lead crack tip and the MSD crack reaches the typical yield strength of the material\textsuperscript{1} This is illustrated in Figure 8-12.
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$$\sigma_y$$

$$2/3 \sigma_y$$

$$\sigma_y$$

$$\sigma_y$$

$$W_1$$

$$W/3$$

$$W$$

Physical crack length, $$2a$$

Figure 8-11, Feddersen residual strength diagram\textsuperscript{16}.

$$r_y(1) = \text{lead crack plastic zone diameter}$$

$$r_y(2) = \text{MSD plastic zone diameter}$$

Criterion for link up: $$r_y(1) + r_y(2) = p - d - \Delta a_1 - \Delta a_2$$

Figure 8-12, Illustration of ligament yield criterion.

A rough way to determine the plastic zone size is to use the approach of Irwin, or Dugdale assuming that the stresses at the crack tip cannot exceed $$\sigma_y$$ of the material. Under these boundary conditions the plastic zone diameter is equal to:
where:
a_{\text{eff}} = \text{Effective crack length} \ (= a_{\text{phys}} + r_y)

\sigma_y = \text{Yield stress, and}

\sigma = \text{Remote stress}

Using the Irwin plastic zone size gives $\alpha = 1/\pi$ while Dugdale calculates $\alpha = \pi/8$. A small comment must be made here. Several authors\textsuperscript{1,15} use the first order approximation of Irwin to calculate the size of the yield plastic zone. However, it is well described in several fracture mechanics textbooks\textsuperscript{17,18} that the Irwin plastic zone size is twice that obtained with the first approximation where stress redistribution is not taken into account.

8.3.4 $K_R$-curve approach

The $K_R$-curve approach is somewhat similar to the $K$-apparent ($K_{\text{app}}$) approach, which will be discussed first.

The $K_{\text{app}}$ approach uses the plane stress critical K value ($K_{\text{app}}$) of a CCT specimen without holes or MSD damage. This value is obtained experimentally on a panel with the same geometry and size of the centre crack as the geometry and length of the lead crack in the MSD panel that is investigated. Failure is predicted when the $K$-value at the lead crack, including crack interaction effects due to the presence of MSD, equals $K_{\text{app}}$ for the CCT specimen without MSD. Because $K_{\text{app}}$ is no material constant, it needs to be determined for every geometry and crack configuration.

This is solved by not using $K_{\text{app}}$ but the $K_R$-curve of a material, which is in principle nothing else than a collection of $K_{\text{app}}$-values as a function of the effective crack length.

With the equations presented in § 8.3.1 and the geometric values of crack and ligament lengths, a modified crack driving force, $K_{G(MSD)}$ can be determined as:

$$K_{G(MSD)} = \beta L K_G$$

\text{eq. 8-4}
where the correction factor $\beta_1$ for the presence of an MSD crack in front of the lead crack must be calculated for different values of $a$, see eq. 8-1. The modified crack driving force is illustrated in Figure 8-13. $\beta_1$ is initially constant. When the lead- or MSD crack starts to extend, $t$ decreases, which results in new values for $k$, $K(k)$, $E(k)$ and an increasing $\beta_1$ from point B. Due to the presence of an MSD crack, the crack driving force does not cross the $K_R$ curve at point A but creates link-up at point C. At point D, the crack extension covers the full ligament and a "new" crack driving force must be calculated based on a larger lead crack length (initial lead crack plus link-up) and the same gross stress. Based on this "new" $K_G$ curve, it follows whether the link-up at point C results in fracture or is followed by another part of stable crack extension.

![Figure 8-13, Influence MSD on crack driving force ($K_G$).](image)

The correction factor $\beta_1$ is calculated using the physical crack length while the $K_G$-values are based on the effective crack length. The calculation procedure to determine a relation between $\beta_1$ and $\Delta a_{phy}$ for each geometric situation (lead crack length, ligament size and MSD crack) is schematised in Appendix A. Because the physical crack length is obtained from the effective crack length by calculating the Irwin plastic zone size (see chapter 3) the $K_R$ curves used for this criterion must be obtained with the Irwin plastic zone size correction as well.

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8.4 FOD with MSD

This paragraph discusses a test program carried out to investigate the residual strength of a through the thickness crack or FOD in a row of holes. § 8.4.1 presents experiments of a FOD in a flat sheet with a row of open holes. The configuration is to some extent comparable with the coupon tests discussed in § 8.2. Paragraph 8.4.2 discusses the residual strength if the holes are no longer open but are filled with rivets. Finally, § 8.4.3 presents the results of a FOD in one of the rivet rows of a lap joint. All panels were 800 mm wide, which allows a direct comparison with other residual strength data presented in this thesis.

8.4.1 FOD in row of open holes

Two CCT like specimens with a row of holes and a 200 mm long centre crack in line with the holes were statically loaded to fracture. The exact configurations and test conditions are shown in Table 8-2.

Initially, specimen nr. 2 contained only a row of holes. Every hole was provided with 0.25 mm long initial flaws through the thickness at both sides. These flaws were made by saw. The specimen was fatigue loaded before a 200 mm long centre crack was sawn in the specimen and the residual strength was determined.

Initial fatigue loading of specimen 2
The fatigue loading was stopped after 22 kcycles. At this time, the initial saw cuts originating from the 7 holes located in the centre reached a crack length of approximately 3 mm. The fatigue crack extension was maximum for the holes in the centre of the specimen and decreased towards the edges. The average crack growth was in the order of 0.15 mm / kcycle and was comparable for all holes.

Residual strength test on specimen 1 and 2
Both specimens were equipped with AB-guides. During static loading, both specimens displayed crack extension originating from the lead crack. Figure 8-14 illustrates the crack extension of the lead crack of both specimens as a function of the remote stress. At a certain load, the cracks link up to the first hole in front of the cracks. In the specimen that was not pre-fatigued, the crack had to reinitiate at the opposite hole edges what needed a significant load increase. The crack in this specimen was still stable after link up with the second hole but the load was comparable with the load at final fracture. This data point between the second and the third hole is not plotted.
Table 8-2, Test matrix- and procedure for open hole FOD tests program.

<table>
<thead>
<tr>
<th>Material: Glare 3 4/3 0.5 T-L</th>
</tr>
</thead>
</table>

Specimen 1

- Width = 800 mm
- Length = 1040 mm
- 11 holes in front of crack per side
- Ø holes = 5.7 mm
- Pitch = 28 mm (= 5 x 5.6)

Initial saw cut

All dimensions are in mm

Test: Residual strength test only

Specimen 2 contains additional through the thickness saw cuts at all holes except for the two edge holes:

- Ø 5.7 mm
- 0.25 mm

Test: Fatigue test, followed by a residual strength test.

CA fatigue loading:

- $\sigma_{\text{max}} = 110$ MPa
- $R = 0.1$
- $f = 2.5$ Hz

Overall specimen dimensions are identical as specimen 1. The 200 mm long saw cut was applied after the fatigue testing.

Figure 8-14 also shows a reference residual strength test with a 200 mm centre crack but without holes. It is clear that this specimen has the highest residual strength when based on the gross stress. The residual strength reduction due to the holes in front of the lead crack was equal to 8.8% while the presence of MSD resulted in an extra reduction of 26.9%. The residual strength results are given in Table 8-3.
Figure 8-14, Crack extension for Glare 3 4/3 0.5 T-L specimens with open holes and lead crack.

For a comparison, it is more realistic to calculate the remote stress as the maximum load carried by the net section, thus to use the net ligament loss width definition presented in eq. 8-2 in § 8.3.2. The fatigue cracks have been modelled as if the fibres were broken as well. In reality, this underestimates the net section area of the specimens with MSD and decreases the calculated net stress. Figure 8-15 illustrates the crack extension as a function of the net stress up to link-up to the second hole. The net link up stress from the lead crack to the first hole is comparable for both specimens with holes, with and without MSD. Also the behaviour after link-up is comparable. The crack in the specimen with MSD is larger after link-up due to the already existing fatigue crack. However, for further crack growth, the load must be increased in a comparable way. Based on the net section, the residual strength of the specimen with holes is only 3% larger than for the specimen with holes and MSD and 6% larger than the reference CCT specimen, see Table 8-3. This increase is attributed to the crack stopping effect of the holes and the incorrect calculation of the net cross section of the MSD specimen because of the intact fibres. It seems that the net ligament loss criterion can be used to predict the residual strength in case of holes and MSD. However, as explained in § 8.3.2, this criterion can only be used to do residual strength predictions for comparable specimen geometries.
Residual strength in combination with rivet holes and fatigue damage

Figure 8-15, Crack extension for Glare 3 4/3 0.5 T-L specimens as a function of the net stress (open holes and lead crack).

The yield stress of the Glare 3 4/3 0.5 material in T direction at 0.2% remaining strain is 283 MPa while the yield stress at the onset of yielding (0.0% remaining strain) is approximately 240 MPa. This latter stress corresponds to the net link-up stresses from the lead crack to the first hole with or without MSD, presented in Figure 8-15.

Table 8-3, Residual strength results for specimens with lead crack in row of open holes. The values between the brackets illustrate the difference with the reference test.

<table>
<thead>
<tr>
<th>Specimen: Glare 3 4/3 0.5 T-L</th>
<th>Residual strength [MPa]</th>
<th>Net. stress(^7) [MPa]</th>
</tr>
</thead>
<tbody>
<tr>
<td>1, holes</td>
<td>178.3 (-8.8%)</td>
<td>317.6 (+6%)</td>
</tr>
<tr>
<td>2. holes + MSD</td>
<td>125.8 (-26.9%)</td>
<td>307.5 (+3%)</td>
</tr>
<tr>
<td>3. reference</td>
<td>195.6</td>
<td>299.7</td>
</tr>
</tbody>
</table>

\(^7\) Based on intact ligament

To support the observations and the development of a calculation method, the specimens were equipped with strain gages in the net section between the holes. Figure 8-16 and Figure 8-17 show the measured strain distribution in loading direction in the specimens with and without MSD.

It is clear that the highest strains occur close to the tip of the lead crack. The yield strain of the Glare material at the onset of yielding is marked with the vertically dotted line. If the strain of a certain strain gage is at the right
side of this dotted line, the material is locally plastic. A strain gage is broken when the strain gage readings no longer increase or decrease with the load, resulting in a horizontal line.

Figure 8-16, Measured strain distribution in net section of specimen 1 with holes. Material Glare 3 4/3 0.5 T-L

Figure 8-17, Measured strain distribution in net section of specimen 2 with holes and WFD.

At the onset of crack initiation from the lead crack, the region between the lead crack and the first hole is far in the plastic region (gage 1 and 2) for both specimens. The region between the first and the second hole is only
Residual strength in combination with rivet holes and fatigue damage

partially plastic (gage 3, 4 and 5) at the hole closest to the lead crack. At first link up from the lead crack to the neighbouring hole also this latter area is fully plastic. At final fracture, the strain at the edges of the specimen is still in the elastic region.

Figure 8-18 illustrates the unidirectional stress strain curve for the Glare 3 4/3 0.5 material together with the situations at strain gage 1 and 2 at crack initiation and link-up for specimen 1 with holes only. It looks as if failure takes place at a constant value of 3% strain. Because the strain gages cover a relatively large area, this 3% of strain is an average. It is well possible that the maximum strain below the strain gage is equal to 4.5%; the ultimate strain of the fibres.

Figure 8-18, Unidirectional stress-strain curve for Glare 3 4/3 0.5 material in T direction

Based on the previous results, the use of the yield stress in the ligament yield criterion, see § 8.3.3, will not give satisfying results. However, it is investigated if there is a unique flow stress between the yield- and ultimate strength of the material that predicts the different link up situations. The results, based on eq. 8-3 are presented in Figure 8-19. Within these calculations, the stable crack extension before link-up is taken into account and the K is calculated for both the lead crack and the MSD crack or hole with the Yokobori solution given in eq. 8-1.
The results illustrate that this criterion gives a larger flow stress for the first link-up compared to the second link-up. The average link-up stress is 574 MPa and the maximum difference is 9.4%. This average flow stress is equal to 94% of the ultimate strength or 2.4 times the yield stress for 0.0% remaining strain. The difference between the flow stress for 1st and 2nd lay up is only 2%. It was expected that the flow stress for specimen 2 with holes and MSD should be somewhat lower since the holes contain cracks. However, this is only observed for the second link up. To know whether this criterion holds for different specimen dimensions, etc. the two specimens of this test program are not sufficient.

Figure 8-20 shows the $K_{GR}$-curve of Glare 3 4/3 0.5 material together with different $K_G$-curves for specimen 1 with holes only. The $K_G$-curves represent several typical loads and initial crack lengths that originate from Figure 8-14.

According to the results given in Figure 8-14, crack initiation took place at a gross stress of 112 MPa. The corresponding $K_G$ curve is given and intersects the $K_{GR}$-curve at point A. To obtain further crack extension, the gross stress needed to be increased.

Link-up from the lead crack occurred at a gross stress of 136 MPa and is represented by point B on the $K_{GR}$-curve. The unstable crack extension was stopped when the crack reached the first hole. At this time, the crack had to reinitiate again and the new initial crack length was equal to $100 + 14.8 =$
114.8 mm. This is illustrated in Figure 8-20 as well. The $K_G$ curve for this link up stress and new initial crack length crosses the $K_R$-curve at point C. This means that the crack extension was stable while moving along the $K_R$ curve from A to B while the crack became unstable from B to C.

Figure 8-20, $K_G$ - curves for several different situations in specimen 1 with open holes.

At point C, the crack became stable again; the gross stress needed to be increased to obtain further crack growth. Link up from the first to the second hole took place at a gross stress of 174 MPa, represented by the intersection point D. Although the specimen was able to withstand this second link-up, only a minor stress increase was enough to obtain unstable crack extension. The $K_G$ curve for the second link up stress and the second new initial crack length remains above the $K_R$-curve, which indicated an unstable situation for this combination of initial crack length and applied stress.

Figure 8-21 illustrates those $K_G$-curves with and without a correction for the presence of the hole in front of the lead crack, which predict first and second link-up. The raising of the $K_G$ curve represents the influence of the hole ahead of the lead crack. A significant increase occurs when the ligament decreases due to crack extension. The figure also shows the $K_G$ curves without the presence of holes, which are labelled with "No MSD".
Figure 8-21, Predictions link-up and fracture for specimen with lead crack in row of holes.

According to these corrected $K_G$-curves, first link-up was expected to occur at a gross stress of 154 MPa, what is 11% too high compared with the experimental results. Second link-up was predicted to occur at a gross stress of 172 MPa what is 3.4% too low. The figure illustrates that the gross stress must decrease to let the $K_G$ curve for the new crack length cross the $K_R$-curve. Thus, the stress predicted for the second link-up is also the residual strength prediction. This estimates remarkably well with the experimental results.

The predictions for specimen 2 with a lead crack in a row of holes with MSD are illustrated in Figure 8-22. The predictions show that the specimen is close to fracture after second link-up but the gross stress could still be slightly increased. The corrected $K_G$ curves are calculated as if the MSD cracks are present in both the aluminium- and prepreg layers. Within the $K_G$-curve approach, only the presence of one single rivet hole is taken into account, not the presence of the full row of rivet holes.

Table 8-4 shows the predictions based on the net ligament loss criteria and the $K_G$ approach for 1st link-up and fracture. All predictions for 1st link-up are within 11% while the predictions for fracture are within 9%. The preference for the prediction method is for the $K_G$ approach since the predictions are based on material parameters ($K_R$-curve). The determined flow stress is actually nothing else than a curve fit.
Residual strength in combination with rivet holes and fatigue damage

Figure 8-22, Prediction for link-up and fracture for specimen 2 with lead crack in row of holes with MSD.

Table 8-4, Residual strength predictions for specimens with lead crack in row of open holes.

<table>
<thead>
<tr>
<th>specimen</th>
<th>NLL (^{11}) (Flow stress = 574 MPa)</th>
<th>(K_G) approach</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3/4 0.5 T-L</td>
<td>1st link-up</td>
<td>Fracture</td>
</tr>
<tr>
<td>1, holes</td>
<td>5.0%</td>
<td>-2.4%</td>
</tr>
<tr>
<td>2, holes + MSD</td>
<td>6.7%</td>
<td>-9.3%</td>
</tr>
</tbody>
</table>

\(^{11}\) Net Ligament Loss criteria

The results of using the corrected crack driving force (\(K_G\)) approach on Glare specimens with a lead crack and open holes, with and without MSD are promising in predicting link-up and final failure.

8.4.2 **FOD in row with filled holes**

After the open hole test program presented in § 8.4.1 the program was extended towards a comparable but more realistic test program where the holes are no longer open but filled with rivets. The flat specimens in this research represent fuselage skin panels with a stringer riveted to the skin that are loaded in circumferential direction, see Figure 8-23. Compared to the program discussed in § 8.4.1 the tested material was changed from Glare 3 to Glare 4B with the rolling direction of the aluminium sheets in longitudinal direction and 2/3 of the fibres in circumferential direction. The rivets that connect the stringer to the skin are not loaded in circumferential...
direction and are, therefore, represented by an aluminium strip in the specimens. In the specimens, the protruded rivet heads are placed in the aluminium stringer part instead of in the Glare skin material. Although this is not realistic, it simplifies the search for a prediction method since the influence of the countersunk head does not have to be taken into account.

Figure 8-23, Specimen definition with respect to fuselage.

The test matrix and program is briefly presented in Table 8-5. Three different CCT panels have been tested:

1. a specimen with a 210 mm centre crack within a rivet row,
2. a specimen with a 220 mm centre crack within a rivet row that contains initial flaws and fatigue crack extension, and
3. a specimen with a 200 mm centre crack as reference test.

*Initial fatigue loading*
Initially specimen 1 was fatigue tested with a row of rivets only. At that time, the specimen contained neither initial flaws nor a lead crack. After 180 kcycles, the rivets were removed for a tear down inspection and the holes showed no signs of crack initiation.
Residual strength in combination with rivet holes and fatigue damage

Table 8-5, Test matrix- and procedure for filled hole FOD tests program.

<table>
<thead>
<tr>
<th>Geometry:</th>
<th>Rivet data:</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specimen 1</td>
<td>• NAS 1097 DD (countersunk)</td>
</tr>
<tr>
<td></td>
<td>• Ø = 4.8 mm (hole Ø = 4.9 mm)</td>
</tr>
<tr>
<td></td>
<td>• Pitch = 24 mm (= 5 x 4.8 )</td>
</tr>
<tr>
<td></td>
<td>• Squeeze force = 17 kN</td>
</tr>
<tr>
<td>Specimen dimensions</td>
<td>CA fatigue loading:</td>
</tr>
<tr>
<td>• Width = 800 mm</td>
<td>• $\sigma_{\text{max}}$ = 110 MPa</td>
</tr>
<tr>
<td>• Length = 1040 mm</td>
<td>• $R = 0.05$</td>
</tr>
<tr>
<td></td>
<td>• $f = 2.5$ Hz</td>
</tr>
<tr>
<td></td>
<td>Initial saw cut</td>
</tr>
<tr>
<td></td>
<td>15</td>
</tr>
<tr>
<td></td>
<td>24</td>
</tr>
<tr>
<td>All dimensions in mm</td>
<td>Test: Fatigue and residual strength test</td>
</tr>
</tbody>
</table>

Specimen 2 contains additional through the thickness saw cuts at all holes except at the two edge holes:

<table>
<thead>
<tr>
<th>Test: Residual strength test after fatigue loading</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specimen 2 contains additional through the thickness saw cuts at all holes except at the two edge holes:</td>
</tr>
</tbody>
</table>

Subsequently, initial flaws with a length of 1.5 mm were sawn in the 5 central holes and the specimen was reassembled with rivets. The starter cracks were larger than the squeezed rivet heads and crack extension could be measured visually. During the next 180 kcycles, fatigue crack growth was observed from these initial flaws up to 4.2 mm of crack extension. Initially, the crack growth was rather slow in the order of 0.01 mm/kcycle. However, after 100 kcycles the crack growth rate became constant and was equal to 0.022 mm/kcycle. This is much slower compared
to the results for open holes (± 0.15 mm/kcycle). No cracks were observed at the locations of the other holes. After fatigue testing, a 210 mm long lead crack was milled in the specimen removing the damage due to the initial flaws and the fatigue loading.

Specimen 2 was equipped with 1.5 mm long initial flaws at all rivet holes except for the two holes at the specimen edges. Fatigue crack growth was visible during the 180 kcycles of fatigue loading and the maximum crack lengths were in the order of 6.5 mm. The average crack growth rate was in the order of 0.027 mm/kcycle. After the fatigue loading, a lead crack of 220 mm was milled over the centre rivets and crack extension. The distance between the lead crack tip and the first MSD crack was only 2.7 mm.

The reference specimen was not fatigue loaded.

*Residual strength tests*

Figure 8-24 illustrates the crack extension as a function of the gross stress of three different specimens. The figure illustrates the location of the first two rivets, their closing heads, the initial flaws and the fatigue crack extension. Because the initial lead crack lengths differ, the location of the rivets relative to the lead crack tip is not the same for the specimens. Based on the $K_r$-curve for Glare 4B 4/3 0.5 T-L, the residual strength predictions for a 205 and 210 mm long initial crack are 200 and 196 MPa respectively. The test results and predictions are listed in Table 8-6.

Figure 8-24, Physical crack extension for Glare 4B 4/3 0.5 T-L specimens with filled holes and lead crack.
Residual strength in combination with rivet holes and fatigue damage

Specimen 1 with a lead crack and holes but no initial flaws showed stable crack extension starting from the lead crack. After link up between the lead crack and the first hole, the stable crack growth continued. The crack linked up with the second hole and the panel fractured almost directly after this event along the rivet row. The residual strength reduction compared to the prediction for a panel with a 210 mm long initial crack only given above is 6.3%.

In specimen 2, the intact ligament between the lead crack and the MSD crack tip originating from the first hole was only 2.7 mm. During static loading, not the lead crack but the fatigue crack from the first hole extended in the direction of the lead crack. Just before link up between the lead crack and the rivet hole, the lead crack started to extend and created link-up. Identical fracture behaviour was observed between the first and the second rivet; the fatigue crack at the second hole extended towards the first rivet but link-up was created from the extended lead crack. After this link-up, the load could not be increased anymore and the panel fractured. The residual strength reduction compared with the prediction for the reference panel is 33%. This is significant.

Figure 8-25, Crack extension for Glare 4B 4/3 0.5 T-L specimens as a function of the net stress (filled holes and lead crack)

Figure 8-25 shows the same results but now for the net section stresses, obtained with eq. 8-2. In the calculation for the net section, the fatigue cracks have been accounted for as if the fibres were broken. Therefore, the actual net section stress for the specimen with MSD is somewhat smaller.
Again, the link-up stresses from the lead crack to the first hole are comparable for the two specimens with a rivet row. Also the stresses at which 2nd link-up and fracture occurs are comparable. The differences between the net section stresses for the rivet row specimens and the predicted values for cracks only are very large; specimen 1 gives a 27% higher residual strength and specimen 2 a 36% higher residual strength. These results are also listed in Table 8-6. It seems as if the net ligament loss criteria is not applicable for the rivet row specimens.

Table 8-6, Residual strength results for specimens with lead crack in rivet row. The values between the brackets illustrate the difference with predictions based on the reference test.

<table>
<thead>
<tr>
<th>Specimen: Glare 3 4/3 0.5 T-L</th>
<th>Residual strength [MPa]</th>
<th>Net. stress(^1) [MPa]</th>
</tr>
</thead>
<tbody>
<tr>
<td>1, rivets (2a₀ = 210 mm)</td>
<td>185.7 (-6.3%)</td>
<td>375.3 (26.8%)</td>
</tr>
<tr>
<td>2. rivets + MSD (2a₀ = 220 mm)</td>
<td>131.3 (-33%)</td>
<td>393.7 (35.8%)</td>
</tr>
<tr>
<td>3. reference</td>
<td></td>
<td></td>
</tr>
<tr>
<td>prediction 2a₀ = 210 mm</td>
<td>200</td>
<td>296</td>
</tr>
<tr>
<td>prediction 2a₀ = 220 mm</td>
<td>196</td>
<td>290</td>
</tr>
</tbody>
</table>

\(^1\) based on intact ligament

Also the use of the flow stress as illustrated in Figure 8-26 is not useful to do predictions; specimens one shows large differences between 1st and 2nd link-up, while specimen 2 gives a very high flow stress for the 2nd link-up. 1st link-up gave no realistic results because of the very small initial ligament.

Figure 8-26, Calculated flow stresses for specimens with rivet rows.
Figure 8-27 shows the $K_G$-curves with and without a correction for the presence of a rivet without fatigue cracks in front of the lead crack, which predicts first and second link-up. 1st link-up was predicted at a gross stress of 178 MPa, what is 4.7% too high compared with the experimental results. Second link-up was predicted at a gross stress of 179 MPa, what is 3.2% too low. According to the figure, the gross stress must be decreased to cross the $K_R$-curve after 2nd link-up. The link-up stress predicted for the 2nd link-up is therefore also the residual strength prediction.

Figure 8-27, Predictions link-up and fracture for specimen with lead crack in row of rivets.

Figure 8-28 shows a similar figure for specimen 2 with a row of rivets and MSD. In this figure the corrected $K_G$-curve for the prediction of the first link-up has been omitted. Due to the relatively large fatigue cracks, the ligament has become rather small and results in large, non-realistic correction factors for the crack driving force $K_G$. After first link-up the ligament between the new, larger lead crack and the first crack in front of this lead crack is larger, resulting in more realistic $K_G$ curves.

However, also the prediction for the second link-up (predicted to occur at 84 MPa) is with 36.8% difference far too low. It seems that the assumption that the fibres are broken in the wake of the fatigue crack gives a far too conservative prediction for this geometry.
Figure 8-28, Predictions link-up and fracture for specimen with lead crack in row of rivets with MSD.

Table 8-7 shows the predictions based on the $K_G$-curve approach. The prediction for 1st link-up is within 4.7%. The prediction for fracture is about 3.6%. The results for specimen 2 should be neglected, where the fatigue cracks have reduced the cross section enormously.

The $K_G$-curve approach has shown to predict the gross stresses for 1st link-up and fracture very well. The prediction method is valid for the situations of open and filled holes without MSD. In case the MSD cracks have reduced the cross section too much, no realistic predictions will be found from the $K_G$-approach.

Table 8-7, Residual strength predictions for specimens with lead crack in row of filled holes.

<table>
<thead>
<tr>
<th>specimen</th>
<th>$K_G$ approach</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 4 4/3 0.4 T-L</td>
<td>1st link-up</td>
</tr>
<tr>
<td></td>
<td>Fracture</td>
</tr>
<tr>
<td>1. rivets</td>
<td>4.7%</td>
</tr>
<tr>
<td>2. rivets + MSD</td>
<td>--</td>
</tr>
</tbody>
</table>

8.4.3 FOD in lap joint

The previous test program on rivet rows simulated the occurrence of FOD at the location of a riveted stringer in longitudinal direction. The following test program also investigated cracks in longitudinal direction in a rivet row
but now at the location of a lap joint where two sheets have to be connected in circumferential direction. The situation differs from the previous program since:

- the rivets in these tests have to transfer load and,
- secondary bending takes place due to this load transfer.

Two lap joint specimens of the same material as the rivet row specimens, Glare 4B 4/3 0.5 T-L, have been made, see Table 8-8.

**Table 8-8, Test matrix and procedure for lap joint specimens with FOD.**

<table>
<thead>
<tr>
<th>Specimen geometry:</th>
<th>Material: Glare 4B 4/3 0.5 T-L</th>
</tr>
</thead>
<tbody>
<tr>
<td>800 mm</td>
<td>Rivet data:</td>
</tr>
<tr>
<td>210 mm</td>
<td>• NAS 1097 DD (countersunk)</td>
</tr>
<tr>
<td>1140 mm</td>
<td>• Ø = 4.8 mm (hole Ø = 4.9 mm)</td>
</tr>
<tr>
<td>All dimensions in mm</td>
<td>• Pitch = 24 mm</td>
</tr>
<tr>
<td></td>
<td>• Squeeze force = 17 kN</td>
</tr>
<tr>
<td>Initial saw cut</td>
<td>CA fatigue loading:</td>
</tr>
<tr>
<td>15 mm</td>
<td>• $\sigma_{\text{max}} = 90$ MPa,</td>
</tr>
<tr>
<td></td>
<td>• $R = 0.05$ and</td>
</tr>
<tr>
<td></td>
<td>• $f = 3.5$ Hz</td>
</tr>
<tr>
<td></td>
<td>• no initial flaws</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Test procedure</th>
</tr>
</thead>
<tbody>
<tr>
<td>Specimen 1:</td>
</tr>
<tr>
<td>1. CA fatigue loading for 60 kcycles</td>
</tr>
<tr>
<td>2. Creation of FOD and additional residual strength test</td>
</tr>
<tr>
<td>Specimen 2:</td>
</tr>
<tr>
<td>1. CA fatigue loading for 180 kcycles</td>
</tr>
<tr>
<td>2. Creation of FOD and additional residual strength test</td>
</tr>
</tbody>
</table>

Specimen nr. 1 has first been fatigue tested for 60 kcycles and specimen nr. 2 for 180 kcycles. During fatigue loading, the specimens contained no
centre crack. Usually, the thickness at the location of a lap joint is locally increased to reduce the load. For that reason, the maximum applied fatigue load was reduced to 90 MPa compared to the previous tests. None of the holes contained initial flaws before the fatigue and/or residual strength testing. The damage present in the specimen nr 2 was created by fatigue loading only. During fatigue loading, the fatigue damage was monitored with Eddy Current equipment.

After fatigue testing, a centre saw cut of 210 mm length was created in both specimens. This saw cut was located in the non-critical lower rivet row, where fatigue cracks will grow in the inner sheet without countersunk holes. In this way, the influence of the protruded rivet head or knife edge effect could be disregarded in the analysis.

Initial fatigue loading
Several times during the fatigue loading, the specimens were inspected with an Eddy Current apparatus for fatigue cracks. After 60 kcycles, this equipment showed no crack development in the lap joint area of specimen 1 and the fatigue loading was stopped. A tear down inspection confirmed that no cracks were present.

Specimen nr. 2 was fatigued for 180 kcycles. At this time, the Eddy Current equipment gave a clear indication of fatigue crack growth in the lap joint area but no quantitative results. It was decided to wait with the tear down inspection until after the residual strength test in order to avoid any disturbing effects due to rivet re-installation. After the residual strength tests, large fatigue cracks were found in the outer rivet row without lead crack. The average fatigue crack length emanating from each rivet hole was 6.5 mm. Thus the average total fatigue crack length was equal to $2 \times 6.5 \text{ mm} + \varnothing 4.9 \text{ mm} = 17.9 \text{ mm}$ resulting in a 75% reduction of the net section. The fatigue crack length was only measured at the mating surface. The outside layers showed no sign of any fatigue crack initiation. The total fatigue crack extension at the mating surface was measured and equal to

- a total length of 250 mm excluding the 210 mm long centre crack in the bottom rivet row and
- a length of 400 mm in the critical upper row

Residual strength test
Figure 8-29 illustrates the crack extension as a function for both lap-joint specimens. Surprisingly, the curves and the residual strength for the two
different specimens are identical. During loading, crack initiation took place at the lead crack tip and the crack grew stable in the direction to the first rivet. When the ligament was reduced with approximately 60%, the load could not be increased anymore, indicating unstable fracture. However, because the test was carried out displacement controlled, some crack length measurements could be done in this unstable region. After the first rivet, the gross stress could be raised only slightly more before link-up took place to the second rivet. At that moment, the maximum load could not be increased anymore. At lower loads and larger crack lengths, the rivets in the rivet row with saw cut sheared off and caused final fracture.

![Figure 8-29, Gross stress versus crack extension for lap-joint specimens.](image)

Based on these results, it can be concluded that the static strength of a lap joint was not influenced by the presence of the fatigue damage in the mating aluminium layers. Since the specimens were already beyond the critical situation when the rivets sheared off, an explanation must be given why the specimens showed a comparable stress versus crack extension and comparable residual strength. It is expected that the fatigue cracks in the single mating aluminium layer do not influence the fracture behaviour of the other, un-cracked aluminium and fibre layers significantly especially since it is a relatively small damage which is also surrounded by a fatigue delamination that will prevent high local strain concentrations in the neighbouring fibre layer.

Figure 8-29 also shows the test results of a reference tests and the test with filled holes discussed in 8.4.2. The loading of the rivets and secondary
bending causes the difference in residual strength. The specimens showed a remarkable amount of buckling above the crack tip especially in the ending sheet in the lap-joint. This could not be prevented with the use of AB guides, see Figure 8-30. However, at the crack tip the buckling reduced significantly and was therefore not expected to have a large influence.

Figure 8-30, Illustration of buckling in lap-joint.

Because of the occurrence of secondary bending and the pin loading of the rivets, it was not tried to predict the residual strength for these lap joints.

8.5 Summary and conclusions

This chapter described the residual strength behaviour of a large through the thickness crack with the presence of holes with or without fatigue damage in front of the crack. From a geometrical point of view, this situation seems comparable with the presence of MSD in an aluminium sheet, where a large lead crack is created by link up of several small MSD cracks.

However, the occurrence of MSD due to fatigue loading is different for Glare and aluminium. MSD due to fatigue loading in aluminium exists of
through the thickness cracks while MSD in Glare only occurs in the aluminium layers since the fibres in the wake of the crack remain intact.

A small test program on Glare and aluminium specimen with open holes that were fatigue loaded illustrated that Glare specimens could sustain almost twice as much fatigue cycles before failure. The crack initiation period of Glare was significantly shorter but the crack growth period was much longer.

Fatigue tests on Glare 3 specimens with a row of open holes showed crack initiation after 30 kcycles for $\sigma_{\text{max}} = 120 \text{ MPa}$ and $R=0.05$. In specimens where the centre hole contained two 0.5 mm long initial flaws (i.e. saw cuts), the crack initiated almost directly from these flaws. However, the crack initiation in the other holes was not influenced by the presence of these flaws or the fatigue crack extension at those locations. The amount of cycles before crack initiation was predicted rather accurately with the use of fatigue diagrams for aluminium by taking into account the residual stresses in the laminate after curing and the stiffness differences between the different layers. The crack growth rate was in the order of 0.13 mm per 1000 fatigue cycles independent of the presence of initial flaws. Interaction between the MSD cracks became visible only when 60% of the ligament was fatigue cracked. From that moment, the crack growth rates increased.

Since rows of open holes, are not common in aircraft structures, the test program also contained specimens with a row of holes filled with rivets. Fatigue loading under the same circumstances as for the open holes showed no fatigue crack initiation after 350 kcycles. This was confirmed with a tear down inspection. Comparable with the open hole program, one specimen was carried out with 0.5 mm long initial flaws originating from the centre hole while an other specimen contained initial flaws of this size at every hole. After filling these holes with rivets like in a normal production process, the specimens were fatigued with 180 kcycles and showed no crack initiation or crack growth either. Based on these results, it can be concluded that the occurrence of MSD in Glare at the location of a row of rivets with the main loading perpendicular to this row, is not likely to occur during the lifetime of an aircraft, not even if all the holes contain initial flaws.

The previous tests did not give any results for the situation that the rivets transfer loads in the direction of the external load like for example in a lap- or butt joint. Under these circumstances, there are two additional stress raising contributions that affect the fatigue life of a rivet row; a stress concentration caused by pin loading of the rivet and secondary bending.
Additionally, fretting of the mating surfaces can cause crack nuclei and decrease the crack free life. Müller found crack growth in Glare lap joints within the DSG of an aircraft but only as a part through crack. Fatigue cracks initiated in the single aluminium layer at the mating surface. The crack growth rate in this aluminium layer increased after 420 kcycles. At that time the crack extension was approximately 4 mm what corresponds to a net section reduction of more than 50%. After 600 kcycles and complete link up of the cracks in the first aluminium layer, crack initiation took place in the second layer. From these results it can be concluded that MSD will occur in a Glare lap joint, but it will not result in increased crack growth rates or link-up within the DSG of an aircraft.

It has been investigated whether this changes if "lead cracks" occur in the area with MSD caused by FOD instead of MSD. FOD can cause a large, through the thickness crack. Therefore a test program is carried out to investigate the residual strength of a FOD in a row of holes, a rivet row and a lap joint. Every test configuration also contained specimens with initial manufacturing flaws that were additionally fatigued. To predict the residual strength for the open hole and rivet row specimens, several prediction tools that are developed for aluminium have been evaluated. These are the net ligament loss criterion, the ligament yield criterion and the $K_{app}$ criterion. This latter criterion has been modified to a $K_G$ criterion.

**FOD in row of open holes**

Three 800 mm wide Glare 3 4/3 0.5 L-T CCT specimens have been statically tested:

1. A specimen with a 200 mm long centre crack in a row of open holes.
2. A specimen with a row of open holes with 0.25 mm long initial flaws at both sides of every hole. This specimen was fatigue loaded for 22 kcycles what resulted in fatigue cracks with a length of 3 mm. Additionally a 200 mm long saw cut was applied in the specimen at the row of open holes.
3. A reference panel with a centre crack of 200 mm.

The residual strength results have been presented in two ways; as gross stresses and as net section stresses where the maximum load is divided by the area of the intact ligament. Based on the gross stress, the presence of a row of holes (specimen 1) gave a reduction in residual strength of 9% compared to the reference test (specimen 3). The presence of MSD (specimen 2) resulted in a decrease of 27% compared to the reference. If
the results are based on the net section stress, the situation differs. Now the specimen with holes only (specimen 1) gave the highest residual strength, followed by the specimen with MSD (-3%) and the reference specimen (-6%).

With the prediction tools, a flow stress was determined at which link-up or failure will occur. This flow stress is very close to the ultimate strength (95%) and predicts failure and link-up within 10%. However, this flow stress is in reality nothing else than an empirical parameter. Its applicability needs to be verified in a much larger test program.

Predictions of link-up and fracture based on the $K_{cr}$-curve of the material and a corrected $K_{Gc}$-curve for the geometry resulted in predictions that are within 11% for first link-up and 7% for fracture. Since these results are based on geometry and material only, they are very promising for the use as a prediction tool.

**FOD in row with filled holes**

In this program, three 800 mm wide Glare 4B 4/3 0.5 L-T CCT specimens have been statically tested:

1. A specimen with a row of rivets that was fatigued for 180 kcycles and showed no sign of cracks afterwards during a tear down inspection. Subsequently, the 5 centre holes were carried out with 1.5 mm long initial flaws, the rivets were installed again and the specimen was fatigue cycled for another 180 kcycles. The crack growth rate was very slow (0.022 mm/kcycle) and the maximum observed fatigue crack including initial flaw was 4.5 mm. Additionally, a centre crack of 210 mm was sawn in the specimen, which removed all the present initial flaws and fatigue crack extensions.

2. A specimen with a row of rivets and 1.5 mm long initial flaws at both sides of every rivet was fatigue loaded for 180 kcycles. This resulted in maximum fatigue cracks including the initial flaws of 6.5 mm. Before the residual strength test, a centre crack of 220 mm was sawn in the specimen, which removed the initial flaws and fatigue crack extensions at this location. The distance between the lead crack tip and the first MSD crack was only 2.7 mm.

3. A reference panel with a centre crack of 200 mm.

Based on the gross stress, the presence of a rivet row (specimen 1) gave a reduction in residual strength of 6% compared to a specimen with a lead
crack only (specimen 3). The presence of MSD in the rivet row (specimen 2) resulted in a decrease of 33% compared to a specimen with a lead crack only. Based on the net section stress, the specimens with rivets gave approximately 30% higher residual strength results than the reference test. Thus the net ligament loss criterion is not accurate for these specimens. Also the flow stress criterion was found to be of no use.

Predictions based on the $K_\sigma$ approach worked well for the specimens with rivets only. The predictions for first and second link-up were within 5% from the test results. In case of fatigue damage, this approach predicted a far too low residual strength (37%).

Lap joints
Two lap joints have been tested, made of the same material as the filled hole specimens

1. A lap-joint specimen that has been fatigue tested for 60 kcycles. This specimen showed no fatigue cracks in a tear down inspection. After re-installing the rivets, a 210 mm long saw cut was applied.

2. An identical specimen as specimen 1, that was fatigued for 180 kcycles. After these fatigue cycles, no cracks were present at the outside layers but inspection with Eddy Current techniques showed clearly fatigue cracks at the mating surface. After fatigue loading, also this specimen received a 210 mm long sawn central crack.

Both saw cuts were applied in the non-critical rivet row. Surprisingly, the residual strength behaviour of the lap-joint specimens was identical. The presence of fatigue damage made no difference. It is expected that the fatigue cracks in the single mating aluminium layer do not influence the fracture behaviour of the other, un-cracked aluminium and fibre layers significantly especially since it is a relatively small damage which is also surrounded by a fatigue delamination that will prevent high local strain concentrations in the neighbouring fibre layer.
Residual strength in combination with rivet holes and fatigue damage

References


9 RESIDUAL STRENGTH OF GLARE WITH "CRACK STOPPERS"

9.1 Introduction

As described in chapter 2, there is a tendency in the civil aircraft industry to introduce the two-bay crack criterion as a requirement for a newly designed aircraft. Airbus Industries requires that the A380 fuselage will be capable to resist a two-bay crack while Boeing claims that the 777 already fulfils the two-bay crack requirement.

If this crack is caused by FOD, the two-bay crack requirement can become a critical design parameter for the application of Glare as fuselage skin material. For this kind of damage the fibres in the wake of the crack are broken, which reduces the residual strength of the material significantly compared with a fatigue crack where the fibres remain intact. As a consequence, the weight benefits due to the excellent fatigue behaviour might not be used to their full extent. This is especially the case for cracks in longitudinal direction where the frames are spaced with a large internal distance and are often flexibly connected to the skin, see Figure 9-1. Consequently, the crack arresting capabilities of the frames in case of a two-bay crack are limited and the residual strength is largely depending on the residual strength capabilities of the skin material1.

Figure 9-1, Illustration of a two-bay longitudinal crack in a fuselage.

There are several possible solutions to improve the residual strength in the circumferential direction. The research presented in this chapter was
Chapter 9

confined to add crack arresting capabilities to the Glare skin material itself
by modifying the material locally or bonding strips or doublers to the basic
skin that function as crack stoppers. A comprehensive overview of the crack
stopping capabilities of local stiffening elements will be given in § 9.2. This
overview includes the expectations towards the simultaneous combination
of splices and doublers as presented in chapter 7.

In § 9.3, the computer code P-cracks is used to calculate the crack growth
behaviour of Glare skins with doublers, and the results are compared with
the experimental results obtained in chapter 7. Subsequently, a parametric
study with this computer code is presented to determine the influence of
doubler and skin characteristics on the effectiveness of doublers as crack
stoppers.

Within the framework of the Glare Technology Research Program,
DaimlerChrysler Aerospace Airbus (DCAA) performed several full-scale
damage tolerance experiments on Glare fuselage panels from autumn 1999
to spring 2000. These single curved panels contained several artificial
damages. The panels were first fatigue loaded by internal pressurisation
representing flight cycles. During these flight cycles the crack growth of the
different damages was monitored. Subsequently, all damage was repaired
except for a two-bay crack with which the structure was subsequently tested
on its maximum load carrying capacity.

The used Glare skin material was first tested without crack stopping
elements. The test results illustrated that the two-bay crack would indeed
become a critical design parameter for this skin material. § 9.4 and § 9.5
describe a study that was carried out after this test with the aim to design
effective residual strength improvements for the skin materials. The purpose
of this research was to show effects on the residual strength rather than to
predict the residual strength of the stiffened and curved full-scale structure.
The investigation was limited to the addition of extra fibre layers or titanium
crack stoppers. The full-scale tests including the design modifications are
described in § 9.6. Finally, some conclusions are presented in § 9.7.

9.2 Crack arrest due to local stiffening elements

JAR 25.903 considers the hazards of uncontained engine failure and
states²:
"Practical design measures to minimise the risk of catastrophic damage may include for example possible redundant design of crack stoppers to limit the dynamic propagation of tears which have been caused by debris impact"

A crack stopper is nothing else than a strip, especially designed to prevent an existing crack to grow to any catastrophic size. The function of the crack stopper is to take over part of the load of the damaged skin, thus reducing the stress and the stress intensity factor at the crack tip in the skin. Since the crack stoppers cause the same in-plane effects on the skin as the application of stringers and frames in a conventional fuselage structure, it is useful to briefly discuss the theoretical background of the residual strength of stiffened panels with the use of the residual strength diagram presented in Figure 9-2.

Figure 9-2, Residual strength diagram for a panel with crack stoppers.

For an unstiffened flat panel with a central crack, the residual strength as a function of the initial crack length is represented with curve 1. If the panel is equipped with stiffening elements like stringers, frame or doublers, they add stiffness and tend to decrease the stress at the crack tip by load transfer from sheet to stiffening element. The residual strength for the skin is now represented with curve 2. The maximum tip stress reduction occurs when the crack extends slightly beyond the centre line of the stiffening element. Failure of this element is represented by curve 3. To illustrate the effect of the stiffening element, different initial crack lengths are discussed.
If the initial crack is small \((a_0)\), the stress field at the crack tip is hardly influenced by the presence of the stiffening element. The stress at the onset of unstable crack extension (point A) will be the same as for an unstiffened panel with the same dimensions. When the unstable growing crack reaches the stiffening element (point B), the load concentration in the stiffened element will be so high that it fails without stopping the unstable crack growth.

For an initial crack with length \(a_1\), unstable crack growth occurs when the gross stress is equal to \(\sigma_1\) (point C). However, for this gross stress the stiffening element is able to reduce the peak stress at the crack tip while the crack approaches the stiffening element. For this combination of skin- and stiffening element stiffness and the effectiveness of the connection, the stiffening element is able to stop the unstable crack growth (point D). After this crack arrest, the gross stress can be further increased to \(\sigma_1\) at which the stiffening element fails (point E). If the initial crack extends almost from one stiffening element to the other \((a_2)\), the stiffening element is very effective in reducing the peak stress at the crack tip. For this initial crack length the crack will grow in a stable manner until it reaches the stiffening element.

If the stiffening element would have had a larger cross section or strength, the point of stiffening element failure (point E) shifts to the maximum of curve 2 where skin failure is critical. In the same way, a less effective stiffening element would result in a lower residual strength or maybe no crack arrest at all. This also holds for a worse connection between the stiffening element and the skin that decreases the effectiveness of the stiffening element.

As discussed in chapter 7, splices need to be reinforced and protected against moisture ingress what can be obtained either with a smart splice design or by bonding a thin doubler on top of the outside splice. The experimental results in this chapter illustrated that both splices and doublers can add crack arresting capabilities to the material or structure. However, since the doublers will be thin, the crack arresting possibilities are limited and the capabilities of the splice decreases fast with increasing distance between the crack tip and the splice location. Possibly the combination of splices and crack stoppers at the same location can have a significant crack stopping effect. This is illustrated in Figure 9-3.
Residual strength of Glare with "crack stoppers"

Figure 9-3, Illustration of possible crack stopping effect of splice in combination with a doubler.

In chapter 7, two different residual strength tests programs were presented on panels with splices that were connected with doublers. The first program was discussed in § 7.4.1 and contained fully spliced Glare 3 3/2 0.3 specimens with 3.2 mm thick bonded 2024-T3 doublers. Figure 9-4 illustrates the measured crack extension as a function of the applied load for these panels. It can be observed that the crack arresting capabilities of the splices in these panels was in the order of 10%. For the situation where the first splice closest to the crack tip was positioned before the doubler, no crack stopping influence of the doubler is observed. For the panels where the first splice lies underneath the doubler a small crack stopping effect of the doubler is observed. This effect was larger if the first splice was positioned further away from the doubler edge. It appears that these specimens illustrate the effect that was described above and plotted in Figure 9-3.

The second program discussed in chapter 7 was carried out on Glare 3 specimens that contained one single splice below a 0.2 mm thick 2024-T3 or 0.65 mm thick Glare 2 doubler. This was described in chapter 7 as a more realistic panel configuration.
Compared to the fully spliced specimens, the doublers on the single spliced specimens were closer located to the initial crack. Their crack stopping effect was considerable, even from the 0.2 mm thick 2024-T3 doubler. However, the splices showed hardly any crack stopping effect and only after the crack became unstable and the maximum load was reached. Consequently, the conclusion was drawn that a single splice is not able to arrest crack growth. The relation is comparable with the crack growth for an initial crack $a_2$ in Figure 9-2 where the crack grows in a stable manner until the crack stopper fails.

9.3 Parametric study with P-Cracks

The influence of external doublers on the residual strength is also analysed with the computer code P-Cracks, developed at the Delft University of Technology. This computer code is based on the displacement compatibility method and includes elastic-plastic behaviour of the skin and doublers as well as rivet flexibility.

Based on the data of the skin, fastener and doubler materials and the geometry, P-Cracks calculates the Mode I stress intensity factor $K_I$, the rivet loads and the strain in the doubler for a specific crack length and applied load, see Figure 9-5.
Residual strength of Glare with "crack stoppers"
One of the advantages of P-Cracks is the capability to deal with plasticity of the doubler. If the doubler becomes plastic, the crack stopping effect of the doubler reduces significantly. The output of P-cracks is either scaled to the applied load what is only possible as long as the materials behave elastic or the applied load must be close to the real load. This means that the calculated $K_I$ for the skin with P-cracks depends on the applied load. If this load is too low, the doublers remain elastic and the crack stopping effect is overestimated. Actually, P-Cracks should be used as a design verification tool; to determine if a certain panel design is able to carry a certain crack length for the allowable load. When it is used to determine fracture curves, the input load should be close to the final failure load.

![Image of fracture curves and experimental results for Glare 3 panels with Glare 2 doublers.](image)

**Figure 9-6.** Calculated fracture curves and experimental results for Glare 3 panels with Glare 2 doublers.

For the Glare 3 panels with 2024-T3 or Glare 2 doublers described in § 9.2, P-Cracks has been used to calculate failure of the different panels. The results are presented in Figure 9-6 and the input parameters are given in Table 9-1. The predicted failure curves are rather close to the experimental results. Some differences are observed and will be discussed.

P-Cracks assumes that the doubler is located at the doubler centre line where the rivets attach the doubler to the skin. Assuming a small rivet pitch, small rivet flexibility and a high yield- and ultimate strength simulates bonding of the doubler. As a result of this approach, the intact doubler bridges the skin crack only when the crack tip has passed the doubler centre line what is illustrated in Figure 9-6. In reality, the bonded doubler will bridge the crack at the moment that the crack tip passes the doubler edge...
Residual strength of Glare with "crack stoppers"

and the maximum crack stopping effect is shifted towards this doubler edge. This is illustrated by the experimental results in Figure 9-6.

The curves also predict failure of the doublers while in reality the doublers remained in tact and the skin fractured. This difference can be due to the static delamination between the doubler and the skin, observed during the tests. This reduces the crack stopping effect of the doubler and the strain in the doubler.

Table 9-1, Input data P-Cracks

<table>
<thead>
<tr>
<th>Skin</th>
<th>E [GPa]</th>
<th>t [mm]</th>
<th>R -curve²</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3 3/2 0.2</td>
<td>53.5</td>
<td>1.1</td>
<td>24.2</td>
</tr>
<tr>
<td>Glare 3 3/2 0.3</td>
<td>57.5</td>
<td>1.4</td>
<td>35.2</td>
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</table>

<table>
<thead>
<tr>
<th>Doubler</th>
<th>E [GPa]</th>
<th>t [mm]</th>
<th>W [mm]</th>
<th>εc [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 2 2/1 0.2</td>
<td>68.0</td>
<td>0.65</td>
<td>25</td>
<td>4.6</td>
</tr>
</tbody>
</table>

With P-cracks, a parametric study has been performed⁶ to gain a better understanding of the effects of the design choices on the residual strength of stiffened panels. The study has been done for 2024-T3 and Glare type skins with either 2024-T3, Glare 1 or Glare 2 doublers. The following parameters have been varied and compared with the weight consequences:

- Elastic versus elastic-plastic calculations,
- Doubler stiffness, yield-, ultimate- and blunt notch strength,
- Rivet pitch
- Skin thickness

The influence of the crack stopping effect of a bonded doubler compared to a riveted doubler is compensated by calculating the $K_i$ values for different crack lengths $a_k$ below the doubler and using the average value $K_i^*$ for the skin fracture curve:

$$K_i^*[a_j] = \frac{\sum_{j=1}^{i=n} K_i(a_k)}{n+1} \quad \text{eq. 9-3}$$

where $a_{jn} - a_j = W_d$. 
The results of the parameter study are given in running loads (LPUW = Load per Unit Width). These are maximum loads that the cracked panels can withstand per unit width of the panels. To obtain the maximum gross stress for a panel, these values should be divided by the thickness of the panels where the cross-section of the doublers is distributed over the unit panel width:

\[ t_{eq} = t_{skin} \frac{A_{doubler}}{p} \]  

\[ \text{eq. 9-4} \]

where \( p \) = doubler pitch.

Elastic calculations

The reference panel that is considered is a 1.6 mm thick 2024-T3 skin with 1.6 mm thick 2024-T3 doublers that were riveted to the skin with a 4.0 mm rivet and a rivet pitch of 20 mm. The doubler width is 25 mm and the doubler pitch is 130 mm. During the study, all parameters are kept constant while only one parameter is varied. The variable parameters and their ranges are indicated in Table 9-2.

For all cases studied, the panel remained doubler critical. Figure 9-7 gives an overview of the influence of a certain geometry parameter on the LPUW per unit mass. A value of 100 along the x-axis represents the value of the parameter in the reference panel. It was found that an increase of the doubler cross-section, either by increasing the thickness or the width, is favourable. This is remarkable because increasing the doubler cross-section results in an increase of stiffness and consequently in an attraction of load to the already critical doublers. However, fracture of the stiffener is based on the blunt notch properties, which allow a much higher load than if fracture would be dominated by the fracture toughness like for the skin. For the same reason, increasing the skin thickness results in a less efficient panel geometry. The optimum situation would be a zero skin thickness. In that situation, the load will be carried by the doublers that do not contain cracks but blunt notches. Decreasing the doubler pitch was the most effective solution. This does not only result in a larger doubler cross section but also in more doublers that will decrease the stress intensity at the crack tip and raise the skin fracture curve.
Residual strength of Glare with "crack stoppers"

Table 9-2, Variation in- and basic parameters applied in P-Cracks study

<table>
<thead>
<tr>
<th></th>
<th>Thickness [mm]</th>
<th>E [GPa]</th>
<th>σ_{bn} [MPa]</th>
<th>Specific weight [kg/m³]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Skin</td>
<td>variation</td>
<td>1.0 - 3.0</td>
<td>-</td>
<td></td>
</tr>
<tr>
<td>2024-T3</td>
<td></td>
<td>1.6</td>
<td>72.0</td>
<td>2770</td>
</tr>
<tr>
<td>Glare 3</td>
<td>3/2 0.3</td>
<td>1.4</td>
<td>58.5</td>
<td>2420</td>
</tr>
<tr>
<td>Glare 4A</td>
<td>4/3 0.3</td>
<td>2.32</td>
<td>56.6</td>
<td>2390</td>
</tr>
<tr>
<td>Doubler</td>
<td>variation</td>
<td>1.0 - 3.0</td>
<td>6.8 - 79.2</td>
<td>294 - 1050</td>
</tr>
<tr>
<td>2024-T3</td>
<td></td>
<td>1.6</td>
<td>72.0</td>
<td>420 2770</td>
</tr>
<tr>
<td>Glare 1</td>
<td>3/2 0.3</td>
<td>1.4</td>
<td>63.6</td>
<td>669 2420</td>
</tr>
<tr>
<td>Glare 2</td>
<td>4/3 0.3</td>
<td>1.95</td>
<td>65.0</td>
<td>588 2470</td>
</tr>
<tr>
<td>Other parameters</td>
<td>Minimum [mm]</td>
<td>Maximum [mm]</td>
<td>Basic [mm]</td>
<td></td>
</tr>
<tr>
<td>Rivet pitch</td>
<td>5</td>
<td>50</td>
<td>20</td>
<td></td>
</tr>
<tr>
<td>Doubler width</td>
<td>10</td>
<td>50</td>
<td>25</td>
<td></td>
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<tr>
<td>Doubler pitch</td>
<td>90</td>
<td>190</td>
<td>130</td>
<td></td>
</tr>
</tbody>
</table>

Figure 9-7, Results elastic parameter study 2024-T3 skin and doublers

From a material characteristic point of view, the use of a doubler material with a higher blunt notch strength is most efficient. A doubler material with a lesser Young’s modulus gives also an improvement but is relatively less efficient.
Figure 9-8 shows once again the parameter results of the aluminium 2024-T3 skin and 2024-T3 doublers presented in Figure 9-7. These are the lowest three lines in Figure 9-8. The two lines directly above represent the situation when the 2024-T3 doublers are replaced by Glare 1 3/2 0.3 doublers with a smaller thickness of 1.4 mm and a reduced specific weight (2770 kg/m³ for 2024-T3 versus 2424 kg/m³ for Glare 1 3/2 0.3). Because the stiffness of these doublers is lower while the blunt notch strength is considerably higher, the reduced cross section is more than compensated, resulting in a higher LPUW per mass. For all the variations in geometry parameters, the panel remained doubler critical. The trends are comparable with the trends for the panel with 2024-T3 doublers but more severe. Only when the blunt notch strength is increased or when the doublers are bonded, the panel becomes skin critical.

Changing also the skin to a Glare 3 3/2 0.3 material with a smaller thickness, Young’s modulus, specific weight and slightly smaller fracture toughness results in the three upper lines of Figure 9-8. The panel was doubler critical for all the parameter variations. The smaller thickness and specific weight of the skin material resulted in a more efficient design compared to the previous panels. The trends are again comparable as for the previous panel configurations.

The last configuration analysed contained a Glare 4 4/3 0.3 skin and Glare 2 4/3 0.3 doublers. This results in a skin thickness of 2.35 mm and a doubler thickness of 1.95 mm. 2/3 of the fibres in the Glare 4 skin were directed in...
the loading direction, resulting in a considerable higher fracture toughness of the skin in this direction. The results, however, show only a slightly higher LPUW per mass compared to the Glare 3 skin with Glare 1 doublers. This is expected because the blunt notch strength of the Glare 2 doubler is decreased compared to the Glare 1 doubler in the previous study, while the stiffness is increased.

**Elastic-Plastic calculations**

When the elastic-plastic behaviour of the doubler is taken into account, the peak of the skin fracture curve due to the presence of the doubler clearly reduces while the doubler fracture curve is considerably higher. This higher curve is due to the higher blunt notch strain in the case of elastic-plastic doubler behaviour. However, by taking plasticity into account, the crack stopping effect is not influenced since the blunt notch strength remains the same. Consequently, the LPUW per mass remain the same.

The P-cracks results illustrated that a doubler is more effective as crack stopper when the cross section and its stiffness increases. However, the effectiveness of the doubler decreases significantly if the doublers become plastic. Therefore, materials with a high yield stress like titanium or elastic materials like fibre prepregs should be used as crack stopper material.

### 9.4 Fibre prepreg or titanium crack stoppers

As described in the introduction, an experimental study has been carried out to improve the residual strength of a Glare skin material that had to be tested with a two-bay longitudinal crack by DCAA in an A380 full-scale fuselage test. In case of a two-bay crack in longitudinal direction a crack is considered that starts in one frame bay, runs over a broken frame and ends in the next frame bay. Where the two-bay crack should start and end within the frame bays is unclear. Since it is no requirement yet, it is also not set by the aviation authorities. DCAA has decided to test a crack of two full bays with a broken central frame where the crack tips end between the rivets of the clip-skin connection, illustrated in Figure 9-9. This choice prevents crack arresting in a rivet hole but can be criticised for not taking into account the history of the crack development, the specific characteristics of the skin material like in the case of Glare and the resistance of the crack stopper to crack initiation. However, this is not further discussed here but considered as a boundary condition for this research. With an A380 like frame pitch of 25 inch, the total initial crack length $2a_o$ considered becomes 1270 mm.
The base line fuselage material investigated in this study is Glare 3 3/2 0.4 with a thickness $t$ of 1.7 mm, a configuration considered for that particular part of the A380 fuselage located in front of the wing. Possible crack stopping solutions that are investigated in this report are:

1. Adding extra fibre layers in the main loading direction, as illustrated in Figure 9-10. This choice of locating the extra fibre-layers changes the material locally from a Glare 3 3/2 0.4 into a Glare 4B 3/2 0.4 material.

2. Adding titanium doublers, either internal or external to the laminate.

The difference between these two solutions is the large difference in stiffness; titanium is almost twice as stiff as a UD glass prepreg in fibre direction. Based on the results presented in § 9.2 and § 9.3 the adding of local thin aluminium doublers was not considered to be effective.

When the local thickness increases are limited to 0.4 mm the necessity of stringer joggling can be prevented. The reinforcement width was set to be 200 mm or less.
Residual strength of Glare with "crack stoppers"

Figure 9-10, Concept idea of adding crack stopping fibre layers.

9.4.1 Predictions for base line material
First the residual strength of the baseline material Glare 3 3/2 0.4 is considered in case the material has to sustain a longitudinal initial crack of 1270 mm. Since the rolling direction of the aluminium sheets corresponds to the longitudinal or flight direction of the aircraft, the residual strength of the Glare material in the T-L direction is investigated. The residual strength predictions are based on the $K_R$-curve approach for flat non-stiffened panels for Glare 3 4/3 0.4 T-L derived in chapter 6. Unfortunately, no $K_{IR}$-curve data are available for Glare 3 3/2 0.4 T-L but the curves used are representative for this Glare variant and lay-up as discussed in chapter 6. Initially, the panel width $W$ is set to 2540 mm assuming that the external load will be carried by the two surrounding frame bays. The effects of stiffening elements, curvature and bulging are not taken in account.

Figure 9-11 contains the $K_R$-curves obtained with Irwin and compliance plastic zone correction together with a curve fit that is extrapolated to an effective crack extension of 300 mm. Based on this fit, the predicted residual strength is 91 MPa. DCAA test results for a Glare 3 full-scale test gave an allowable ultimate stress of 90 MPa and are thus comparable with the predictions. However, it must be noted that the prediction is for a flat, unstiffened panel and is not based upon test-data but on an extrapolation.

Actually, the largest width for which a residual strength prediction can be based upon test data, considering an initial crack $2a_0 = 1270$ mm, is 2000 mm. In this case, the crack driving force $K_0$ touches the $K_{IR}$ curve based on experimental data, see Figure 9-11. The residual strength prediction for this crack configuration is 70 MPa. For a comparable relative crack length ($2a_0/W = 1270/2000 = 0.635$) in an 800 mm wide specimen, the predicted residual strength is 86 MPa. This is of interest because 800 mm is the maximum width of the test machine that was available for this initial study.
Figure 9-11. Residual strength prediction for base line material Glare 3 3/2 0.4 T-L, for $W = 2540$ mm and $2a_0 = 1270$ mm.

9.4.2 Predictions for additional prepreg layers

The addition of two 0.125 mm thick UD prepreg layers in the L direction as illustrated in Figure 9-10 results in a local GLARE 4B 3/2 0.4 material with $t = 1.95$ mm. These reinforcements will be placed at the location of the frames. Initially, DCAA limited the width of the added fibre layers to 200 mm.

For this reinforced Glare 3 skin, a two-bay crack means that the material within two bays between the stiffened locations is completely broken while also the central stiffened part and half of the stiffened material at the adjacent intact frames is broken. This is illustrated in Figure 9-12.
Residual strength of Glare with "crack stoppers"

**Figure 9-12**, Two-bay crack configuration in Glare 3 sheet with reinforcements.

**Figure 9-13**, Residual strength prediction for the stiffening material GLARE 4B 3/2 0.4 T-L for 800 and 2000 mm wide panels and $2a_0/W = 0.635$.

To carry out residual strength predictions for this reinforced panel, the panel is assumed to behave like a 2000 mm wide Glare 4B 3/2 0.4 T-L sheet with $2a_0 = 1270$ mm. This width is based on the maximum width that yields to residual strength predictions based on experimental $K_R$-curve data. In this
case, the $K_{tt}$-curve of a comparable laminate Glare 4B 4/3 0.4 predicts a residual strength of 86 MPa and an effective crack extension ($\Delta a_{\text{eff}}$) of 142 mm. This is illustrated in Figure 9-13. A first estimation of the physical crack extension ($\Delta a_{\text{phys}}$) using the Irwin plastic zone correction $r_y$ gives:

$$\Delta a_{\text{phys}} = \Delta a_{\text{eff}} - r_y = \Delta a_{\text{eff}} - \frac{1}{2\pi} \left( \frac{K_{\text{eff}}}{\sigma_y} \right)^2 = 142 \text{mm} - 106 \text{mm} = 36 \text{mm} \quad \text{eq. 9-5}$$

with: $K_{\text{eff}} = 228$ MPa/m and the yield stress $\sigma_y = 279$ MPa

For a crack stopper width $W_{cs}$ of 200 mm, only 64 mm ($\frac{1}{2}W_{cs} - \Delta a_{\text{phys}}$) of the crack stopper will be intact at the onset of unstable fracture. However, it is questionable whether this is reasonable to predict the residual strength of a locally reinforced Glare 3 panel as if the panel is completely reinforced. If this is possible, the allowable stress in the Glare 3 skin can be calculated with the stiffness ratios given in Table 9-3 and assuming a constant displacement along the loaded edges far away from the crack.

$$\sigma_{\text{Glare3}} = \frac{\sigma_{\text{Glare4B}} \cdot E_{\text{Glare3}}}{E_{\text{Glare4B}}} = \frac{86 \cdot 60000}{59000} = 87\text{MPa} \quad \text{eq. 9-6}$$

This is an increase of 23% compared to the initially predicted 70 MPa for the basic Glare 3 skin without extra fibre layers.

### Table 9-3, Material and structural properties.

<table>
<thead>
<tr>
<th>Material</th>
<th>t [mm]</th>
<th>Width [mm]</th>
<th>E [MPa]</th>
<th>E·t (MPa·mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3 3/2 0.4</td>
<td>1.7</td>
<td>435 (in between frames)</td>
<td>60000</td>
<td>102000</td>
</tr>
<tr>
<td>Glare 4B 3/2 0.4</td>
<td>1.95</td>
<td>200 (under frames)</td>
<td>59000</td>
<td>115050</td>
</tr>
</tbody>
</table>

#### 9.4.3 Base line material with titanium crack stoppers

Another possible crack stopper material is titanium as used in the DC-10$^8$. From the discussion and calculations in § 9.4.2 and eq. 9-6 follows that an improvement of the allowable stress in the Glare 3 skin is reached by:

- Adding crack stoppers with a significantly better fracture toughness than the Glare 3 skin, preferably with
- a lower stiffness
Residual strength of Glare with "crack stoppers"

The reason of a less stiff crack stopper material compared to the basic skin material follows from the main goal of this research; to increase the allowable stress of the baseline material. A higher stiffness of the crack stopper will attract stresses from the skin material and subsequently reduce the allowable skin stress. This results in a less efficient panel design.

Both the stiffness and the fracture toughness of titanium are high. The titanium alloy Ti-6Al-4V has a 25% higher fracture toughness than Al 2024 T3\(^9\) while the stiffness is almost twice as high compared to a UD fibre prepreg layer. Earlier investigations for fibre metal laminates using titanium in combination with glass fibres showed high fracture toughness values\(^{10}\). The titanium crack stoppers will be bonded to or integrated in the Glare skin. Because of the large stiffness difference and in order to decrease the possibility of delamination, it was chosen to integrate the titanium crack stoppers in the laminate. In this way, the titanium can be loaded along two bond lines instead of only one. The schematised configuration is given in Figure 9-14.

![Figure 9-14, Location of titanium crack stoppers in Glare 3 laminate.](image)

Since there are no crack growth resistance curves available for Glare in combination with titanium, no predictions are made. The increase of fracture toughness by adding titanium can improve the residual strength but the stiff crack stopper bands can attract extra load, what can have a negative influence on the residual strength and the allowable skin stress.
9.5 Initial crack stopper experiments

9.5.1 Definition specimen geometry

To verify the approach and the expected residual strength improvements, an initial test program was carried out. Only flat CCT specimens with \( W = 800 \) mm were tested. The properties are:

- base line material = Glare 3 3/2 0.4 T-L
- "local crack stopper material" = Glare 4B 3/2 0.4 or Ti-6Al-4V
- and maximum sheet width = 800mm

The questions to be solved before testing of these panels are:

1. What should be the initial crack length?
2. What should be the width of the crack stopper bands?
3. What is the predicted residual strength of this configuration?

The initial crack length in the 2000 mm wide specimens is 1270 mm. Thus \( 2a_0/W = 0.64 \). Maintaining this ratio results in \( 2a_0 = 508 \) mm for an 800 mm wide CCT specimen. For the test it was decided to take \( 2a_0 = 500 \) mm.

The width of the glass fibre crack stopper bands in the 800 mm wide panels is based on predictions for the amount of stable crack extension. It was chosen to have a constant ratio of the

\[
\frac{\text{Intact crack stopper width}}{\text{total crack stopper width}}
\]

at the onset of unstable crack growth for both configurations. In § 9.4.2 it was predicted that the amount of stable crack for \( W = 2000 \) mm, \( 2a_0=1270 \) mm and \( W_{cs} = 200 \) mm is equal to 64 mm. As a result this ratio becomes 64 mm / 200 mm = 0.32.

The amount of stable crack growth for a comparable Glare 4B 4/3 0.4 panel with \( W = 800 \) mm and \( 2a_0 = 500 \) mm can be predicted with Figure 9-13. From this figure can be predicted found that \( a_{\text{eff}} = 63 \) mm and \( K_{\text{eff}} = 158 \) MPa at the onset of fracture. The yield stress of the glare 4B 4/3 0.4 laminate in T direction is 279 MPa. Filling these values in eq. 9-5 results in a prediction of the stable crack extension of 12.5 mm. Thus the formula to be solved for the crack stopper width is:

\[
\frac{W_{cs} - (0.5W_{cs} + 12.5\text{mm})}{W_{cs}} = 0.32 \quad \Rightarrow \quad W_{cs} = \frac{12.5}{0.5 - 0.32} = 70\text{mm}
\]
Residual strength of Glare with "crack stoppers"

where \( W_{cs} \) is the crack stopper width in the 800 mm wide panel.

\[
\sigma_{Glare3} = \frac{\sigma_{Glare4B} E_{Glare3}}{E_{Glare4B}} = \frac{96.60000}{59000} = 97.6\text{MPa}
\]

This is an increase of 13.5% compared with the predicted 86 MPa (see § 9.4.1 for Glare 3 3/2 0.4, \( W = 800 \text{ mm} \) and \( 2a/W = 0.635 \)) for the baseline material.

In case of the titanium crack stoppers the thickness of the titanium strips is 0.55 mm, and is determined by the amount of available sheet material. To determine the width of the titanium crack stopper, the stiffness \( EA \) of the titanium crack stopper is chosen to be the same as the stiffness \( EA \) of the
Chapter 9

Glare 4B crack stopper to allow direct comparison of the maximum loads between the different panels. This results in a crack stopper width of 55 mm.

Table 9-4, Stiffness and geometry of Glare 3 material with crack stoppers.

<table>
<thead>
<tr>
<th></th>
<th>Glare 4B</th>
<th>Ti-6Al-4V + Glare 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>E [MPa]</td>
<td>59000</td>
<td>66500</td>
</tr>
<tr>
<td>t [mm]</td>
<td>2.0</td>
<td>2.2</td>
</tr>
<tr>
<td>W [mm]</td>
<td>70</td>
<td>55</td>
</tr>
</tbody>
</table>

9.5.2 Test matrix and procedure

Based on this specimen geometry, Table 9-5 contains the proposed test matrix on CCT specimens. The matrix consists of three different residual strength tests:

1. Test 1, a Glare 3-3/2-0.4 CCT panel with an initial crack of 500 mm. This test functions as a reference to investigate the effectiveness of the applied crack stoppers in the next tests.

2. Test 2 is a Glare 3-3/2-0.4 CCT panel equipped with glass-fibre crack stopper bands in the loading direction. The initial crack length is 500 mm. The crack stopper bands consist of two UD glass fibre prepreg bands, integrated in the skin laminate as illustrated in Figure 9-10.

3. Test 3 is a Glare 3-3/2-0.4 CCT panel equipped with internal titanium crack stopper bands. This specimen is tested in the same way as test 2. The rolling direction of the titanium crack stoppers is in loading direction.

All Glare 3-3/2-0.4 skins are tested in the T-L direction.

The residual strength tests are carried out according to ASTM standard E 5611. During the tests the clamp-displacement of the tension machine is controlled at a rate of 0.5 mm/min. The test included a number of stops to measure the crack extension; first at determined load levels, later when visible crack growth occurred. At these stops, the crack extension was measured at a minimum of two locations; at the front and at the back side of the panel. These measurements were carried out with a microscope and the imaging system. To avoid buckling, A.B. guides were mounted to the specimens.
Residual strength of Glare with "crack stoppers"

Table 9-5, Experimental program on CCT specimens.

<table>
<thead>
<tr>
<th>Skin material</th>
<th>Specimen lay-out</th>
<th>N</th>
<th>Geometry (baseline material)</th>
<th>Crack stopper</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3 3/2 0.4 Test direction: T-L</td>
<td><img src="image" alt="Specimen Lay-out" /></td>
<td>1</td>
<td>2a₀ = 500 mm t = 1.76 mm W = 802 mm L = 1042 mm</td>
<td>None</td>
</tr>
<tr>
<td></td>
<td></td>
<td>2</td>
<td>2a₀ = 500 mm t = 1.70 mm W = 801 mm L = 1054 mm</td>
<td>2 UD S2 glass fibre prepregs tcs = 2.0 mm Wcs = 70 mm pcs = 250 mm</td>
</tr>
<tr>
<td></td>
<td></td>
<td>3</td>
<td>2a₀ = 500 mm t = 1.70 mm W = 801 mm L = 1054 mm</td>
<td>Ti-6Al-4V tcs = 2.52 mm Wcs = 55 mm pcs = 250 mm</td>
</tr>
</tbody>
</table>

Strain gages measured the strain at 7 points along the net section; their positions are depicted in Figure 9-16. Strain gages 1 and 4 to 7 are located at the same location for all specimens. The location of gages 2 and 3 differs between test 1 and test 2 and 3. All gages measure strains only in loading direction of the specimens. Gage 1 and 7 were used to check the occurrence of in plane bending and net section yielding.

Additionally to the residual strength program, uni-axial tensile tests were carried out on several materials that are included in the CCT-specimens. These tests were carried out to obtain the material characteristics necessary to carry out residual strength analyses. The Glare tensile specimens were
cut from the CCT specimens after testing. They were taken out of the centre of the specimens above the crack where no plasticity occurred.

![Diagram](image)

Figure 9-16, Position strain gages.

### 9.5.3 Tensile test results

The results of the tensile tests on the Glare specimens with and without additional crack stopping materials are given in Table 9-6. The stress-strain curves are plotted in Figure 9-17. The curves with dark coloured labels are used for the residual strength calculations. All curves reach the ultimate strength at a strain of 4.5%; the ultimate strain of the fibres.
Residual strength of Glare with "crack stoppers"

Table 9-6, Tensile test results.

<table>
<thead>
<tr>
<th>Material</th>
<th>Dir.</th>
<th>E</th>
<th>$\varepsilon_{y,0.0}$</th>
<th>$\sigma_{y,0.0}$</th>
<th>$\varepsilon_{y,0.2}$</th>
<th>$\sigma_{y,0.2}$</th>
<th>$\varepsilon_{ult}$</th>
<th>$\sigma_{ult}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3 - 3/2-0.4</td>
<td>L</td>
<td>59.6</td>
<td>0.3</td>
<td>228</td>
<td>0.65</td>
<td>264</td>
<td>4.59</td>
<td>617</td>
</tr>
<tr>
<td></td>
<td>T</td>
<td>58.5</td>
<td>0.2</td>
<td>149</td>
<td>0.59</td>
<td>236</td>
<td>4.58</td>
<td>603</td>
</tr>
<tr>
<td>Glare 4B- 3/2-0.4</td>
<td>T</td>
<td>57.5</td>
<td>0.2</td>
<td>145</td>
<td>0.65</td>
<td>256</td>
<td>4.56</td>
<td>800</td>
</tr>
<tr>
<td>Glare 3- 3/2-0.4 + Ti doubler</td>
<td>L</td>
<td>63.7</td>
<td>0.9</td>
<td>485</td>
<td>1.20</td>
<td>501</td>
<td>4.6</td>
<td>707</td>
</tr>
<tr>
<td></td>
<td>T</td>
<td>63.8</td>
<td>0.9</td>
<td>467</td>
<td>1.17</td>
<td>490</td>
<td>4.4</td>
<td>672</td>
</tr>
<tr>
<td>Ti-6Al-4V t=0.53 mm</td>
<td>L</td>
<td>118</td>
<td>0.9</td>
<td>942</td>
<td>1.03</td>
<td>965</td>
<td>7.0</td>
<td>982</td>
</tr>
<tr>
<td>S2 prep. 0°</td>
<td></td>
<td>54</td>
<td>-</td>
<td>-</td>
<td>-</td>
<td>-</td>
<td>4.6</td>
<td>2480</td>
</tr>
</tbody>
</table>

Figure 9-17, Stress strain curves for Glare 3- and Glare 4B-3/2-0.4.

The Glare 3 3/2 0.4 curves in the L- and T directions are comparable; only at the initiation of yielding they differ. The Glare 4 B 3/2 0.4 material in the T
direction starts to yield approximately at the same strain and in the same way as the Glare 3 3/2 0.4 material in T direction. However, the strain hardening of the Glare 4 material is significantly larger due to the larger amount of fibres in loading direction.

The curves for the Glare 3 materials with titanium showed a two-phase yielding behaviour. At a strain of about 0.5 %, the aluminium starts to yield, causing a slight decline in the tensile-curve (indicated with Al). At 1% strain, the titanium starts to yield too (indicated with Ti). The 0.2% yield is determined from this point. The Young's modulus is defined as a straight line between the yield point of the titanium and the origin. From this point, the curve continues almost linear again - typical for a fibre metal laminate. When a strain of 4.6% is reached, the glass fibres in tensile direction failed.

9.5.4 Residual strength results
The results of the residual strength tests together with the predictions derived in this chapter are given in Table 9-7. The predictions are given bold and between brackets. The percentages in the table give the increase or decrease of a parameter compared to the baseline material.

Table 9-7, Residual strength results.

<table>
<thead>
<tr>
<th></th>
<th>Test 1</th>
<th>Test 2</th>
<th>Test 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>W = 800 mm</td>
<td>Baseline</td>
<td>Glare 3-3/2-0.4</td>
<td>Glare 3-3/2-0.4</td>
</tr>
<tr>
<td>2a₀ = 500 mm</td>
<td>material:</td>
<td>+ glass fibre crack</td>
<td>+titanium crack</td>
</tr>
<tr>
<td>Glare 3-3/2-0.4</td>
<td></td>
<td>stoppers</td>
<td>stoppers</td>
</tr>
<tr>
<td>Fₘₐₓ [kN]</td>
<td>131.1</td>
<td>142.2</td>
<td>132.9</td>
</tr>
<tr>
<td></td>
<td>+8.5%</td>
<td>+1.4%</td>
<td></td>
</tr>
<tr>
<td>σ₉₀gross [MPa]</td>
<td>92.9</td>
<td>97.7</td>
<td>89.1</td>
</tr>
<tr>
<td></td>
<td>+5.2</td>
<td></td>
<td>-4.1</td>
</tr>
<tr>
<td>Mass [kg]</td>
<td>3.46</td>
<td>3.56</td>
<td>3.87</td>
</tr>
<tr>
<td></td>
<td>+2.9%</td>
<td>+11.8%</td>
<td></td>
</tr>
<tr>
<td>A₉₀gross [mm²]</td>
<td>1412</td>
<td>1460</td>
<td>1492</td>
</tr>
<tr>
<td></td>
<td>+3.4%</td>
<td>+5.7%</td>
<td></td>
</tr>
<tr>
<td>σ₉₀gross-skin [MPa]</td>
<td>92.9 (86)ǂ</td>
<td>97.8 (97.6)ǂ</td>
<td>86.5</td>
</tr>
<tr>
<td></td>
<td>+5.3%</td>
<td>-6.9%</td>
<td></td>
</tr>
</tbody>
</table>

ǂ Predictions made in § 9.4

The gross stress is determined by dividing the applied load F, by the gross cross section area of the panel:
Residual strength of Glare with "crack stoppers"

\[
\sigma_{\text{gross}} = \frac{F}{A_{\text{gross}}} = \frac{F}{W \cdot t}
\]

eq. 9-7

The cross section of the panel equipped with crack stoppers includes the extra material of the crack stoppers. The reason why the mass increases less than the cross section for the Glare 3 material with glass fibre crack stoppers is due to the lower specific weight of the glass epoxy compared to the Glare 3 laminate. In the same way, the mass of the Glare laminate with titanium crack stoppers increases more than the cross section.

The stress in the Glare 3 skin material can be derived as:

\[
\sigma_{\text{gross-skin}} = \left( \frac{A_{\text{tot}} \cdot E_{\text{skin}}}{A_{\text{skin}} \cdot E_{\text{skin}} + A_{\text{cs}} \cdot E_{\text{cs}}} \right) \sigma_{\text{gross}}
\]

eq. 9-8

The addition of fibre crack stopping bands offers an improvement in maximum load of 8.5%. Because of the added material, the improvement in maximum gross stress is lower. Considering a weight increase of 2.9%, an increase of the allowable skin stress with 5.3 % is a clear improvement.

The titanium crack stopper performed worse: \( F_{\text{max}} \) increased slightly but considering the extra added cross section and mass the allowable stress in the skin material decreased with almost 7%. There are two reasons for this decrease:

- The titanium strip is significantly stiffer than the basic skin material and attracts load.
- With the definition of a two-bay crack, used in this report, the titanium crack stopper contains already a large crack. Therefore, the crack does no longer have to initiate in the titanium, which decreases the crack stopping effect considerably.

The predictions based on the \( K_\text{R} \)-curves are rather good. The residual strength of the base line material is underestimated with 7.5% while the residual strength of the crack stopper specimen is underestimated with only 0.2%!

Figure 9-18 illustrates the gross stress versus the physical crack extension (\( \Delta a_{\text{phys}} \)) for all tested CCT- specimens. The crack in the crack stopper specimens initiate at higher stresses compared to reference panel 1. In test 2 as well as in test 3, all stable crack extensions take place in the crack stopper area. The figure also shows the edges of the crack stoppers at the
right side, followed by the base line material. At the onset of unstable crack
growth, the critical crack extension ($\Delta a_c$) in test 2 with fibre crack stopper
was 17 mm. The amount of predicted stable crack growth for this test was
12.5 mm, see § 9.5.1.. $\Delta a_c$ in test 3 with titanium crack stoppers was
significantly less; only 6 mm. The remaining widths of those parts of the
fibre - and titanium crack stoppers in front of the crack that were still intact
at the onset of unstable fracture were respectively 18 and 21 mm.

![Diagram](image_url)

Figure 9-18, Gross stress versus physical crack extension for all tested
CCT-specimens.

For test 1 with $2a_0 = 500$ mm, $K_{IC}$-curves have been generated with the *Irwin*
and *compliance correction*. These curves are illustrated in Figure 9-19
together with the $K_{IC}$-curves for Glare 3 4/3 0.4 T-L for both correction
methods and $2a_0 = 200$ mm. The curves for Glare 3 4/3 0.4 T-L tend to
become identical for both correction methods after some effective crack
extension. For test 1, the $K_{IC}$-curves determined with the different methods
are clearly different.

The $K_{IC}$-curve of test 1 based on *Irwin* is slightly below the curves for Glare 3
4/3 0.4 T-L. The reason for this is the lower yield stress of the Glare material
in test 1. If the yield stress is set equal to the yield stress of Glare 3 4/3 0.4
material in the T- direction, the curves lie on top of each other. The curve for
test 1 is interrupted after $\Delta a_{phys} = 4$ mm because of the occurrence of net
Residual strength of Glare with "crack stoppers"

This is far before the onset of unstable fracture at $\Delta a_c = 18$ mm.

The $K_R$ - curve based on the compliance correction for test 1 and the reference laminate Glare 3 4/3 0.4 T-L are initially approximately equal. However, after $\Delta a_{phys} = 5$ mm, the curve for test 2 continuous above the Glare 3 4/3 0.4 T-L curve. This is attributed to the width effect or cracks that approach the edge of the panel so closely; the specimen dimensions are no longer valid for $K_R$-curve determination. The curve based on the compliance correction seems to have passed its validity boundary. This explains the residual strength prediction for test 1 in § 9.5, which is too low.

Figure 9-19, $K_R$ -curves for base line materials Glare 3 3/2 0.4 T-L and Glare 3 4/3 0.4 T-L, W = 800 mm.

Figure 9-20 contains the compliance $K_R$ - curves for the base line specimen test 1, the specimen with fibre crack stoppers test 2 and a Glare 4B 4/3 0.4 T-L reference material. No $K_R$ - curves based on the Irwin correction have been generated for tests 2 and 3 and it was not possible to generate a representative $K_R$ - curve for test 4 with titanium crack stoppers since the COD fluctuated too much.

The curve for test 2 follows the Glare 4B 4/3 0.4 curve for a large part. This is the reason why the residual strength prediction for the crack stopper panel was so accurate. Also this curve moves upward in relation to the
reference curve for large effective crack extensions. This is again attributed to the width effect.

![Diagram of K\(_R\)-curves](image)

Figure 9-20. \(K_R\)-curves based on compliance correction for base line and crack stopper material, \(W = 800\) mm.

It can be concluded that the \(K_R\) - curve approach based on the compliance correction can be used to do residual strength predictions for the Glare laminate with glass fibre crack stoppers by using the \(K_R\)-curve of the locally created laminate.

All specimens were instrumented with strain gages in the net-section to record the strain during loading. Figure 9-21 shows the strain gage results for test 1, the reference panel. Strain gage 4, close to the crack tip shows plasticity at an early stage, the typical very large strain hardening for Glare and fails as the crack grows through the gage. Before final fracture, net section yielding occurs as illustrated by strain gage 1 where the strain exceeds the yield strain of Glare 3-3/2-0.4.
Residual strength of Glare with "crack stoppers"

Figure 9-21, Strain gage readings test 1; Glare 3 3/2 0.4 T-L reference panel.

Figure 9-22, Strain gage readings test 2; the Glare 3 panel with glass fibre crack stoppers.

Figure 9-22 illustrates the measured strains for test 2; i.e., the panel with glass fibre crack stoppers. By comparing the strain gage readings of gages 1 and 7 and 2 and 4 it can be concluded that no in plane or out of plane
bending took place in the specimen during the test. Like in test 1, net section yielding took place just before final failure. The strain behaviour of test 2 is almost the same as for the reference panel test 1. At the same locations, yielding took place at comparable gross stress levels. Also the slopes of the curves agree very well, indicating that there is no significant change in load attraction. This was expected because of the minor difference in Young's modulus between the skin and crack stopper material.

The strain gage readings for the titanium-reinforced panel test 3, illustrated in Figure 9-23, indicate that this panel behaves differently from test 1 and 2. Due to the asymmetric lay-up and differences between thermal expansion coefficients of the constituents the panel was bent after production. During testing the curved panel is straightened what caused extra tension at the crack stopper side (gage 1 and 4 to 7) and compression at the other side of the panel (gage 2 and 3). All gross stress versus strain curves are steeper compared to test 2 and 3. This is due to the significant higher stiffness of the crack stopper (gages 2, 4 & 5) and the subsequent load attraction away from the Glare 3 skin material (gages 1, 3, 6 & 7). The titanium crack stopper became only plastic at the crack tip (gage 4). The plastic region in the aluminium layers was much larger but at the onset of failure the aluminium at the edges was still elastic (gage 1 and 7).

![Figure 9-23](image-url)  
Figure 9-23, Strain gage readings test 3; the Glare 3 panel with titanium crack stoppers.
9.6 Full scale experiments

DCAA has carried out two full-scale residual strength experiments with improved Glare 3 skin material. These panels are curved and contain stringers, frames and lap joints and are loaded both in the longitudinal- and circumferential direction and with internal pressure. The first full-scale test article design (XB-6) was based on the investigations presented in § 9.5 and contained extra fibre bands in circumferential direction under every frame. This paragraph starts by giving a summary of the test procedure and the test results of the XB-6 test. Based on these results, a new panel (XB-6.1) was designed and the philosophy behind this design is described. This paragraph concludes with a description of the XB-6.1 test results.

9.6.1 Test procedure

The full-scale component tests have been carried out at the IMA GmbH test laboratories in Dresden in a fuselage panel test machine. The panels were 6 m long in longitudinal direction, 3.5 m wide in circumferential direction and had a radius of 3.4 m. The panels contained 6 full frame bays. Within these panels, several initial damages were created both in circumferential and in longitudinal direction:

- cracks within one frame bay and
- two-bay cracks with the centre stiffening element broken

The testing of these panels is divided in two parts. During the first part, the panels are fatigue loaded with 1-Δp flight pressurisation cycles what results in a maximum circumferential stress in the skin of 90 MPa. The crack growth is monitored as a function of the amount of flight cycles. Since the fatigue crack growth in the Glare material tends to be arrested during fatigue loading due to the fibre bridging, the cracks were extended several times with a saw. This included cutting of the fibres. When the two-bay crack in longitudinal direction has reached a length of two full frame bays, all the other damages are repaired and the residual strength for this particular crack is tested. This is the second part of the test.

In panel XB-6, 175 mm wide fibre bands have replaced the 200 mm wide crack-stoppers that were proposed. The distance between two frame bays is 25\". Consequently, the mass of the baseline material Glare 3/2 0.4 per frame-bay and per meter in circumferential direction is equal to 2.65 kg/m. Including the crack stopper-bands, the mass per frame bay per meter in circumferential direction is 2.74 kg/m, an increase of 3.4 %.
9.6.2 Test results XB-6 panel

Fatigue testing
During the fatigue loading process, the crack in the Glare material was extended by sawing after 15, 25, 33 and 37 kilocycles. After 45 kilocycles, the crack was extended from 400 mm to 500 mm and subsequently fatigue loaded at 90 MPa circumferential stress. However, during the next inspection, 177 flight cycles later, the crack was grown at both sides to the next frames, resulting in a total crack length of more than 1270 mm. This is illustrated in Figure 9-24. The crack extended further than the rivet row that connects the skin to the clips. Based on the illustrated unstable crack extension, the used two-bay crack length definition of DCAA seems to be realistic in case of fibre crack stoppers. The fibres are broken over the total crack length. Although the crack stopper bands failed for more than 50%, they showed their crack stopping effect.

Residual strength testing
During the residual strength test, the specimen failed in a different way at both crack tips:

- At one side skin failure occurred; the crack extended in an unstable manner through the skin along the centre of the stiffener bay.
- At the other side, rivet pull through took place of the rivets that connected the skin with the clip. The crack extended along this rivet row in circumferential direction and continued growing in longitudinal direction along the rivet row that connects the next stiffener to the skin.

The residual strength in circumferential direction increased from 90 MPa for the Glare 3 3/2 0.4 skin to 107 MPa for the Glare 3 3/2 0.4 skin reinforced with S2-glass fibre crack stoppers. This is an increase with 18%. Only based on the skin material an improvement of 23% was predicted in § 9.4.2. This is a very good prediction.
Figure 9-24, Photographs of the XB-6 two-bay crack after fatigue loading.
9.6.3 Extra improvement

The improvement due to the 2 extra S2-glass prepreg layers in circumferential direction was not satisfying enough compared to the full-scale residual strength results obtained for the new aluminium alloy 2524-T3. During these tests, rivet head failure took place for the rivet rows that connect the skin to the clips while the aluminium skin remained intact. For that reason, a small investigation was carried out to determine the necessary amount of extra fibre layers in the Glare 3 3/2 0.4 laminate, to increase the residual strength beyond 2524-T3.

This is done with the residual strength calculation method described in chapter 6. It assumes that the relation presented in Figure 6-70:

- also holds in the T-L direction and
- that the residual strength of a crack stopper panel can be predicted as if the whole sheet is made of this crack stopper material

In this way, the amount of extra fibre layers necessary to obtain a certain residual strength can be estimated. This is illustrated in Figure 9-25. The residual strength for a 2524-T3 T-L panel with \( W = 800 \text{ mm} \) and \( 2a_0/W = 0.25 \) is 232 MPa. The figure shows how large the combination of \( FVF_{LD} \) and MVF should be for a Glare material in T-L direction to outperform 2524-T3 in T-L direction. It was found that \( T = \alpha FVF_{LD} + MVF \) must be in the order of 1.2. Table 9-8 contains the T-values for a Glare 3 3/2 0.4 laminate with a different amount of addition fibre layers. An addition of 4 extra fibre layers is sufficient. This means that the thickness increases from 1.7 mm between the stiffening elements to 2.2 mm below the frames and the stiffeners need to be joggled.

<table>
<thead>
<tr>
<th>Material</th>
<th>FVF(_{LD}) [%]</th>
<th>MVF [%]</th>
<th>( T = \alpha FVF_{LD} + MVF ) 1)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glare 3 3/2 0.4 T-L</td>
<td>14.7</td>
<td>70.6</td>
<td>0.99</td>
</tr>
<tr>
<td>Glare 3 + 2 extra fibre layers</td>
<td>25.6</td>
<td>61.5</td>
<td>1.11</td>
</tr>
<tr>
<td>Glare 3 + 3 extra fibre layers</td>
<td>30.1</td>
<td>57.8</td>
<td>1.16</td>
</tr>
<tr>
<td>Glare 3 + 4 extra fibre layers</td>
<td>34.1</td>
<td>54.5</td>
<td>1.21</td>
</tr>
</tbody>
</table>

1)\( \alpha = 1.94 \) in T-L direction, see chapter 6
9.6.4 Test results XB-6.1 panel

Based on the predictions of the previous section, panel XB-6.1 contained 175 mm wide crack stopper bands consisting of 4 extra S2-glass fibre prepregs. To improve the rivet pull through strength, extra 0.4 mm thick and 40 mm wide 2024-T3 strips were included in the crack stoppers\(^\text{12}\) what results in a small decrease of the residual strength based on gross stresses but in an small increase of the maximum load. The crack stopper configuration is illustrated in Figure 9-26.

Fatigue testing

The fatigue testing was performed in an identical manner as the XB-6 panel. Also in this panel, the crack became unstable during the fatigue loading process. However, an operator was able to stop the test before rapid crack growth occurred. At that time, the crack was more or less 100 mm in front of the crack stopper band and the total crack length was 940 mm.

Residual strength testing

It was decided not to extend the crack by sawing directly, but to start the residual strength test and to see how the crack would behave. During loading, the crack started to grow clearly below the 1 $\Delta p$ situation (1 $\Delta p =$
605 mBar). When $1 \Delta p$ was reached, the crack had extended up to the crack stoppers and was stopped.

Additionally, the panel with a two-bay crack of 1100 mm was fatigue loaded with 500 additional pressurisation cycles. During this fatigue loading, stable crack growth took place in the aluminium layers in the order of 30 mm. Both cracks showed the tendency to flap to the centre of the stiffener bay, see Figure 9-27.
Chapter 9

Figure 9-28, Test procedure and results XB-6.1 panel

CA fatigue loading
S_{\text{max}} = 90 \text{ MPa}
(= 1.0 \Delta p = 605 \text{ mBar})
R = 0

Unstable crack extension
(interrupted)

Static loading
(crack extension to edge crack stopper)

XB 6 = Glare 3 3/2 0.3 + 2 UD
(770 mbar) t = 1.7 - 1.95
skin failure (1)

XB 5 = 2524 - T3
(830 mbar) t = 1.8 - 2.4
rivet head failure (2)

XB 6.1 = Glare 3 3/2 0.3 + 4 UD + Al
(980 mbar) t = 1.7 - 2.6 (3)

Glare 3 3/2 0.4

+ 4 UD (0.5mm)
0.4 Al

100 mm

100 cycles

87.5 mm

cycles

clip

2 bay crack

cycles

Crack extension

Internal pressure [mbar]

389
Figure 9-29, Illustration of final failure XB-6.1 panel
Residual strength of Glare with “crack stoppers”

Subsequently the panel was tested for residual strength. At $1.6 \Delta p$, the crack flipped at both sides up to next stiffener, remained there and allowed stable decompression. This is illustrated in Figure 9-29. A summary of the test procedure and results is given in Figure 9-28.

9.7 Conclusions

*Glare with splices and thin covering doublers*

- The combination of splices and doublers is able to increase the residual strength, however only if they are located close to the initial crack tips and if all the aluminium layers are spliced below a doubler.
- The computer code P-Cracks was not able to predict the residual strength of Glare panels with Glare doublers. It predicted failure of the doubler while in reality the skin failed. This is probably due to the occurrence of static delamination between the doubler and the skin that is not taken into account in P-Cracks.
- The P-cracks results illustrated that a doubler is more effective as crack stopper when the cross section and its stiffness increases. However, the effectiveness of the doubler decreases significantly if the doublers become plastic. Therefore, materials with a high yield stress like titanium or elastic materials like fibre prepregs should be used as crack stopper material.

*Glare with and without crack stoppers*

- The local addition of two additional UD fibre prepreg layers in the T-direction of the material is an effective way to improve the residual strength of a Glare 3 3/2 0.4 material in the L-T direction. Tests on 800 mm wide panels gave an improvement of the allowable stress in the Glare 3 3/2 0.4 skin material of 5.3% while the mass increased with 2.9%.
- The application of Ti-6Al-4V crack stoppers in Glare 3 material was not effective; the allowable skin stress decreased with 7%. This is attributed to:
  1. the higher stiffness of the crack stopper compared to the skin material which attracts load and,
  2. the definition of the two-bay crack which resulted in an unfavourable initial crack in the titanium crack stopper.
- Predicting the residual strength of the panel with glass fibre crack stoppers with the $K_R$-curve of the local crack stopper material resulted in an underestimation of the residual strength of the panel of only 0.2%. The $K_R$ - curve approach based on the compliance correction can be
used to do residual strength predictions for the Glare laminate with glass fibre crack stoppers by using the $K\text{R}$-curve of the local created laminate.

- Residual strength tests on full-scale fuselage panels performed by DCAA illustrated the crack stopping effect of the fibre prepreg crack stoppers. By adding two extra fibre layers in circumferential direction in a Glare 3 3/2 0.4 laminate, the residual strength of the structure improved by 18%. This panel failed in two ways; at one side skin failure occurred while at the other side rivet pull through took place.

- By adding two more fibre layers in circumferential direction (total addition of 4 layers), the residual strength of the structure increased tremendously. The crack flapped in the crack stopper region in circumferential direction and arrested at the next stringer. In this way controlled decompression could take place and the situation remained stable.

- The introduced residual strength calculation method in chapter 6 to calculate the residual strength of an arbitrary Glare laminate based on a combination of MVF and $FVF_{LD}$ was accurate.
Residual strength of Glare with “crack stoppers”

References


Summary and conclusions
SUMMARY AND CONCLUSIONS

The present thesis describes the behaviour of Glare laminates in the presence of holes or blunt notches and through-the-thickness cracks or sharp notches. These laminates, which are bonded laminates of thin aluminium layers that are connected by glass fibre reinforced epoxy (prepreg) layers, are now ready for industrial applications. It is most likely that they will be applied in the Airbus A380.

Aeroplane structures are full of blunt notches, which are hard to avoid but decrease the static and dynamic properties of the material. The presence of blunt notches in Glare material has a large impact on the remaining strength of the material, much larger than for example for the current 2xxx and 7xxx aluminium alloys. Therefore, Glare laminates are relatively notch sensitive. For that reason, both the blunt and sharp notch behaviour of Glare are described in this thesis.

Sharp notches in aeroplanes are damage (cracks) created by accidents. For Glare, a distinction must be made between damage caused by fatigue where only the metal layers are cracked, and through-cracks where both the metal and fibre layers are cracked. The residual strength of Glare laminates with fatigue cracks and fibres in loading direction is superior compared to aluminium alloys. Therefore, his thesis focuses in particular on the behaviour of through-cracks.

For the sizing of an aeroplane, the strength of a structural detail with a blunt notch is considered a static, ultimate load case. This means that the structure must prove to be capable of supporting the ultimate load for at least 3 seconds. The residual strength of a structure with a sharp notch on the other hand, is a limit load case, which means that the aeroplane with such a damage must be capable to complete its flight safely while carrying \( \frac{2}{3} \) of the ultimate load.

The thesis is largely based on experimental results and analysis to understand the failure or fracture mechanisms and to investigate the influence of several parameters. It is tried to develop calculation methods that are as much as possible related to, or based on the current methods for metals. This approach will result in the easiest and fastest way for stress departments to apply this new material.

For the investigation of these failure mechanisms, an imaging system has been developed within this study, called "Impress". This is described in
chapter 4. This system is able to visualise the fracture area during testing and to carry out in-situ measurements and image filtering functions. These measurements include the recognition of markers in a specially designed grid that can be bonded on the fracture region and allows the determination of the local strain field. Other measurements are for example the measurement of the crack extension and the Crack Tip Opening Angle (CTOA). The system is able to scan the area around the crack in a highly automated way, to keep it in focus, to store images and to carry out measurements.

The blunt and sharp notch related items, discussed in chapter 3, 5 and 6, will be summarized here separately. After the development of engineering calculation methods in chapters 5 and 6, three studies have been carried out that focus on the technical application of Glare in an aeroplane: Chapter 7 discusses the blunt and sharp notch behaviour of spliced Glare laminates, chapter 8 focuses on the occurrence of Multiple Site Damage (MSD) in Glare and the possibility to predict link up and failure and chapter 9 presents a study towards the development of Glare with integrated crack stoppers.

**Chapter 3 and 5: Blunt notch behaviour**

Previous experiments showed that the blunt notch strength of Glare laminates is related to the Metal Volume Fraction (MVF). There is no relation between the stress concentration factor $K_t$ of a notch and the blunt notch strength of Glare laminates (Chapter 3).

Chapter 5 describes an extensive experimental study carried out in this thesis, to develop a general blunt notch prediction tool. First, the fracture mechanism has been investigated by loading open hole specimens to different levels of their blunt notch strength and subsequently investigate the fracture service and the delamination behaviour. In contrast with the expectations found in literature (Chapter 3), the results gave no indication that fibre breakage occurred at the notch root prior to failure. During loading, fibre splitting occurs within the matrix at the hole edge at approximately 75% of the failure load. Further load increase causes an increase of matrix cracking. Close to failure, static delaminations are created between the resin and fibre rich layer of the prepreg. From the load-displacements curves can be derived that small-scale plasticity occurred between 40 and 45% of the failure load. At ± 70% of the failure load, the net section of the specimens starts to yield.
Because of the better blunt notch behaviour of metals, no standard uni-axial blunt notch test method exists for metals. Therefore, a study has been carried out to obtain a specimen geometry that suffices. This study revealed that the uni-axial blunt notch strength related to the net section (net blunt notch strength) is independent of the width and that the net blunt notch strength of a single hole is representative for the net blunt notch strength of a row of holes with constant hole pitch. Filling the hole with an unloaded rivet had no influence on the blunt notch strength. Based on these results, a specimen with a width of 50 mm and a single central hole with a diameter of 4.8 mm was defined.

Many different Glare variants but also aluminium and prepreg specimens were tested. Some of the specimens were instrumented with strain gauges and a grid in the net section next to the notch. The results confirmed that filling of the holes with non-load carrying rivets has no influence on the blunt notch strength. Also the applied squeeze force had no influence. Uni-axial loading of the laminates under an angle with their fibre direction decreased the blunt notch strength significantly. Measurements of the strain at failure directly next to the notch root indicated strains that were significantly larger than the ultimate strain of the fibres. This supports the observations within the fracture mechanism research that delamination occurs at the notch root prior to failure.

The scatter in the results was small; more than 97% of the results had an absolute difference with the average results that is less than 3.5%. The scatter showed a normal distribution.

For the prediction of the uni-axial net blunt notch of a specific Glare laminate in one of the principle material directions, a calculation method is derived based on the MVF and the Fibre Volume Fraction (FVF) of the fibres in loading direction. These parameters describe accurately the blunt notch strength of a certain Glare lay-up.

Besides the uni-axial blunt notch tests, also shear and bi-axial tests were performed. All these specimens were instrumented with a grid to measure the strain fields. These results were used to verify a large FE calculation for different Glare laminates and loading combinations. 3D fracture surfaces were proposed that predict blunt notch failure based on the ultimate strain of the fibres and the aluminium. The results were rather conservative since delamination has not been taken into account.
Chapter 3 and 6: Sharp notch behaviour of through-cracks

Glare with through-cracks that were loaded under increasing load exhibited a capability for slow stable crack growth prior to rapid failure. This behaviour is comparable with aluminium alloys.

Chapter 3 summarised and described the available residual strength of Glare laminate as a function of different parameters. It was observed that the residual strength:

- was higher for a laminate loaded in rolling direction of the aluminium alloy than loaded perpendicular to the rolling direction,
- increased as a function of the pre-stretching after curing,
- increased for lower temperatures,
- decreased for larger angles between the loading direction and the principle material directions,
- decreased significantly like aluminium alloys due to the occurrence of buckling,
- decreased significantly when the bridging fibres are cut and the ratio between the crack in the fibre layers and the crack in the aluminium layers decreases to a factor 0.5 or less,
- was not influenced by the origin of the crack (saw-cut versus impact by a sharp object) nor by the shape of the crack tip (fatigue starter crack versus sharpened saw-cut),
- of a Glare 3 laminate is comparable with 2024-T3.

During the search for a material parameter to describe the residual strength of a Glare laminate, it was found that the use of a critical stress intensity factor is not applicable since it depends on the specimen geometry. Another material parameter that was investigated is the $K_{IC}$-curve that incorporates stable crack growth and plasticity. This is done by implementing these effect in a so-called effective crack length. This effective crack is either determined using the Irwin plastic zone correction or with a compliance plastic zone correction.

The $K_{IC}$-curves obtained with an Irwin plastic zone correction that are found in the literature for Glare are independent of the specimen geometry. However, the amount of stable crack extension in these curves is limited because of Net Section Yielding (NSY) due to the small specimen widths that were applied. As a consequence, the presented curves are mainly a function of the gross stress only. Implementation of the compliance plastic zone correction results in more or less identical $K_{IC}$-curves for a specific
Summary and conclusions

Glare variant and different geometries and can therefore be identified as a material parameter.

The $K_{IR}$-curves obtained with the Irwin plastic zone correction assume plasticity effects from the very beginning of loading and are therefore initially below the $K_{IR}$-curves obtained with the compliance plastic zone correction. After some crack extension, the shape of the curves according to the different methods become identical but a comparable physical crack extension gives a different effective crack extension for both methods. The effect of buckling was only reflected in the $K_{IR}$-curves based on the compliance plastic zone correction.

For the development of a general calculation method for through-cracks, the fracture mechanism has been studied in more detail. In chapter 6, a new phenomenological model for fracture of Glare laminates with through-the-thickness cracks is presented. It was known that the interfaces between the aluminium layers and the fibre rich layers in front of a uni-axially loaded crack are loaded with shear stresses. The new model assumes that these shear stresses are not only the result of plasticity but also of crack blunting in the aluminium layers. Although the shear stresses due to plasticity move with the crack extension and increased loading, the shear stresses due to blunting reduce with extending crack length. Superposition of these effects results in a decrease of the amount of static delamination for larger crack extensions. Static delaminations release the fibre layers from following the large strain concentrations of the aluminium layers at the crack tip. Because the ultimate strain of the fibres is significantly less than for aluminium, failure of the fibres is delayed by this kind of delamination. As a result, the aluminium layers control crack initiation and initial crack growth but the fibre layers will dominate larger crack extensions and fracture. Semi-destructive- and non-destructive inspections confirmed the creation of static delamination before dynamic delamination and illustrated that static delamination advanced before the crack tip. Fracture of the fibre layers occurred in bundles, randomly in the close front of the crack tip.

Besides focussing on the fracture mechanism, the influence of several important parameters like the constituent properties, the amount of layers and the thickness of the aluminium layers on the residual strength have been investigated. The following important relations were observed:

- the residual strength of a laminate is directly related to the strain hardening capacity of the laminate and thus to the aluminium layers in relation to the prepreg; a larger strain hardening increases the residual
Blunt and sharp notch behaviour of Glare laminates

A reduction of the failure strain and strength of fibres decreases residual strength of a Glare laminate,

- the residual strength of a laminate increases with lower bond strength, since this results in larger static delaminations,
- impact tests carried out with a blunt impactor on specimens that were pre-loaded showed no influence of the pre-loading on the residual strength,
- the laminate thickness has no influence on the residual strength and a small influence of the aluminium layer thickness between 0.3 and 0.5 mm is seen as all the laminates show a relation with the MVF.

A large test program illustrated the applicability of the $K_{eq}$-curve approach as a material parameter. Both methods, the Irwin and the compliance method, gave comparable results. A calculation method is derived that is based on the MVF and the FVF in loading direction of a specific laminate. Based on the amount of load carrying layers and a derived ratio between the different layers, a multiplication factor is obtained which is to be multiplied with a standard $K_{eq}$ curve for 2024-T3 to obtain the laminate curve. This works well for curves obtained with the compliance method. To investigate this for the Irwin method, not enough data are available. The derived prediction model is only validated for Glare variants that are based on 2024-T3 alloys and are loaded under tension in either L-T or T-L direction. Additional research is necessary to extend this prediction model for different loading conditions.

Chapter 7: Blunt and sharp notch behaviour of spliced laminates

The splicing concept for Glare allows the production of large sheets, flat, single or even doubly curved. This has several advantages like a decrease of weight, lower assembly costs and reduced inspection and maintenance costs. The splicing concept has been developed since 1993. The driving parameters are

- No significant reduction of material properties like ultimate strength, yield strength, blunt notch strength and fatigue behaviour.
- Maintaining the durability of the material.
- Minimum weight increase,
- Minimum impact on the production process, and
- A flat outer skin and no interference of the inner skin with the locations of stiffeners, frames, doublers and joints.
Chapter 7 presents and discusses the fatigue and static behaviour of several different splice configurations that resulted in the current splice design based on the Self Forming Technique with “overlap” splices. This latter splice configuration is now a mature solution with no stiffness, yield strength, blunt-notch and residual strength reduction, a flat outer skin and the ingress of moisture is almost eliminated.

The rather complicated overlap splice contains several different details. However, these details can be divided in three groups:

- **the external doubler**, where an aluminium layer ends at the outside of the laminate,
- **the internal doubler**, where an aluminium layer ends within the laminate and
- **the simple splice with external doubler**

The **external doubler** does not reduce the static properties but at higher loads the doubler delaminates from the laminate. Only after a large amount of fatigue cycles and at a relatively high load (240 kcycles, $\sigma_{\text{max}}$ between 120 and 150 MPa) the outer aluminium layer below the end of the external doubler cracks.

The static strength of the **internal doubler** detail reduced from 4% for a 0.3 mm thick doubler up to 28% for a 0.5 mm doubler. Fracture is initiated in the covering aluminium layer at the location of the internal doubler ending that shows local necking at high loads. The fatigue properties of this detail are not influenced within a normal lifetime of an airplane. Also the static blunt notch properties are not influenced by the presence of an internal doubler ending.

The **simple splice with external doubler** results in a decrease of static properties, which are less severe than for the **internal doubler** detail. It is concluded that none of the splice details will become critical in an aeroplane fuselage when designed according the current design principles.

The research on the total **splice based on SFT technique including internal doublers** showed that the properties of this splice design can well be described and predicted based on the static and dynamic behaviour of the different splice details. The stiffness, yield stress, blunt notch strength, fatigue properties and remaining strength after fatigue of a splice with and without filled holes are not decreased compared to non-spliced reference laminates. Also the residual strength of a spliced panel with a through crack
at the most critical locations is not decreased. The only property that decreased over the splice is the static strength, which is no critical design parameter for the ultimate load.

Based on these results, it can be concluded that a spliced panel based on the Self-Forming Technique with internal doublers does not influence the design allowables and adds extra advantages to the design of a large capacity aeroplane in Glare.

**Chapter 8: FOD in row of rivet holes**

This chapter described the residual strength behaviour of a large through the thickness crack with the presence of holes with or without fatigue damage in front of the crack. From a geometrical point of view, this situation seems comparable with the presence of MSD in an aluminium sheet, where a large lead crack is created by link up of several small MSD cracks. However, the occurrence of MSD due to fatigue loading is different for Glare and aluminium. MSD due to fatigue loading in aluminium exists of through the thickness cracks while MSD in Glare only occurs in the aluminium layers since the fibres in the wake of the crack remain intact.

Several fatigue tests on open hole specimens illustrated that the crack initiation period of Glare was significantly shorter but the crack growth period was much longer compared to aluminium. Since rows of open holes, are not common in aircraft structures, the test program also contained specimens with a row of holes filled with rivets. Fatigue loading under the same circumstances as for the open holes showed no fatigue crack initiation after 350 kcycles. Also when these holes contained initial flaws 0.5 mm long initial flaws, no crack initiation occurred. Müller carried out several test programs on Glare lap joints where the rivets transfer load. He found crack growth within the Design Service Goal (DSG) of an aircraft but only as a part through crack. After 600 kcycles and complete link up of the cracks in the first aluminium layer, crack initiation took place in the second layer. From these results it can be concluded that MSD will occur in a Glare lap joint, but it will not turn into WFD within the DSG of an aircraft.

It has been investigated whether this changes if "lead cracks" occur in the area with MSD caused by FOD instead of MSD. FOD can cause a large, through the thickness crack. However, the probability that it occurs in a row of holes is so small that it is no requirement. Three different configurations have been tested, (1) FOD in a row of open holes, (2) FOD in a row of filled
holes and (3) FOD in a lap joint. Every test configuration also contained specimens with initial manufacturing flaws that were additionally fatigued.

To predict the residual strength for the open hole and rivet row specimens, several prediction tools that are developed for aluminium have been evaluated. These are the net ligament loss criterion, the ligament yield criterion and the $K_{npp}$ criterion. This latter criterion has been modified to a $K_G$ criterion.

As expected, the residual strength of a panel with a lead crack and open holes without fatigue damage reduces compared to a reference panel with only a lead crack. Filling of the holes reduced the holes resulted in a smaller residual strength reduction. When the holes contained fatigue cracks, the residual strength values were further reduced. However, when the residual strength was based on the intact net section (after fatigue loading), the strength of the open holes specimens was almost comparable with the reference and the strength of the filled hole specimens even increased. This effect is due to the crack stopping effect of the holes and the intact fibres below the fatigue cracks emanating from the holes.

Predictions of link-up and fracture based on the $K_R$-curve of the material and a corrected $K_G$-curve for the geometry are rather accurate for the open hole specimens and the filed hole specimens. When the filled holes contained fatigue cracks, the fracture prediction became too conservative.

The lap joint specimens with and without fatigue cracks gave no difference in residual strength. It is expected that the fatigue cracks in the single mating aluminium layer do not influence the fracture behaviour of the other, un-cracked aluminium and fibre layers significantly especially since it is a relatively small damage which is also surrounded by a fatigue delamination that will prevent high local strain concentrations in the neighbouring fibre layer. For these specimens, no predictions were carried out with the $K_R$-curve approach since it is unclear how to correct for intact aluminium layers.

**Chapter 9: Development of crack stoppers**

To increase the residual strength of Glare laminates like Glare 3, a study has been carried out to investigate the crack stopping effect of doublers, eventually in combination with splices.
Blunt and sharp notch behaviour of Glare laminates

It is found that the combination of splices and doublers is able to increase the residual strength, however only if they are located close to the initial crack tips and if all the aluminium layers are spliced below a doubler. Therefore, the splicing concept was not further regarded as a crack stopping solution and the attention was focussed on the application of internal crack stoppers, either by adding extra fibre bands or a titanium doubler.

The local addition of two additional UD fibre prepreg layers in the T-direction of the material is an effective way to improve the residual strength of a Glare 3 3/2 0.4 material in the L-T direction (cracks in longitudinal direction). These extra fibre layers create a Glare 4B 3/2 0.4 material locally. The residual strength could be predicted with the $K_R$ – curve of the material Glare 4B 3/2 0.4 material.

The application of Ti-6Al-4V crack stoppers in the Glare 3 material was not effective; the allowable skin stress even decreased. This is attributed to:

- the higher stiffness of the crack stopper compared to the skin material which attracts load and,
- the applied definition of the two-bay crack which resulted in an unfavourable initial crack in the titanium crack stopper.

The residual strength of Glare 3 material with local Glare 4B has been tested in a full-scale fuselage test performed by DCAA. The experiment illustrated the crack stopping effect of the fibre prepreg crack stoppers. By adding two extra fibre layers in circumferential direction in a Glare 3 3/2 0.4 laminate, the residual strength of the structure improved by 18%.

Based on the calculation method presented in chapter 6 and a preferred crack stopping capability, a new fibre crack stopper configuration with four instead of two extra fibre layers has been defined. This new configuration with crack stoppers in circumferential direction and a two bay crack has been tested in a full-scale test as well and resulted in a tremendous residual strength increase. The allowable circumferential skin stress increased with 75%.! The crack flapped in the crack stopper region in circumferential direction and arrested at the next stringer. In this way controlled decompression could take place and the situation remained stable.
SAMENVATTING

HET GEDRAG VAN GLARE LAMINATEN IN DE AANWEZIGHEID VAN GATEN EN SCHEUREN

door Tjerk-Johan de Vries

Deze studie beschrijft het gedrag van Glare laminaten in de aanwezigheid van gaten en scheuren waarbij alle lagen zijn gebroken (vergelijkbaar met zaagsnedes). Glare laminaten zijn opgebouwd uit afwisselende dunne laagjes aluminium en glasvezel epoxy. Na geruime tijd van onderzoek is dit materiaal nu gereed voor toepassingen op grote schaal. Het is zeer waarschijnlijk dat het materiaal wordt toegepast in de Airbus A380.

Gaten met afgeronde randen zoals klinknagelgaten zijn moeilijk te voorkomen in een constructie maar verlagen de statische - en vermoeiingseigenschappen, vooral in laminaten zoals Glare. In deze studie is het bezwijkgedrag en de sterkte van Glare in de aanwezigheid van gaten onderzocht. De test variabelen waren verschillende Glare varianten, gat configuraties (inclusief rijen van gaten en gaten gevuld met klinknagels) en belastingscombinaties. Voor éénzijdig belaste Glare laminaten met gaten is een voorspellingsmethode voor de sterkte afgeleid gebaseerd op de Metaal Volume Fractie (MVF) en de Vezel Volume Fractie (FVF) van de vezels liggend in belastingsrichting. Voor meer ingewikkelde combinaties van belastingen is een voorspellingsmethode afgeleid gebaseerd op een 3D bezwijkoppervlak dat wordt bepaald door de maximale rek van de glasvezels en het aluminium. Deze voorspellingsmethodiek leidt tot conservatieve voorspellingen aangezien het optreden van lokale delaminatie rond de gatrand niet is verdisconteerd in de methodiek.

Scheuren zijn een vorm van schade. Voor Glare moet een onderscheid worden gemaakt in de oorsprong van de schade. In het geval van vermoeiingsscheuren stroomt de aluminium lagen terwijl de glasvezel lagen intact blijven. Alleen in het geval van zware impact schade treden scheuren op in alle lagen van het laminaat. De reststerkte van een constructie met vermoeiingsscheuren en vezels in belastingsrichting is superieur ten opzicht van aluminium. Deze studie richt zich op scheuren door alle lagen. In het onderzoek is veel aandacht besteed aan het begrijpen van het breukmechanisme. Daarnaast is de invloed onderzocht van verschillende parameters zoals de materiaal eigenschappen van de het aluminium en de glasvezel epoxy, de kwaliteit van de hechting tussen de verschillende lagen, de dikte van de aluminium lagen, enz. Het test
Samenvatting

programma toonde nogmaals aan dat de $K_R$ curve benadering die is gebaseerd op de scheurweerstand van het materiaal kan worden beschouwd als een materiaal eigenschap. Zoals bij de voorspellings-methodiek voor gaten is ook voor het optreden van scheuren een voorspellingsmethode afgeleid gebaseerd op de MVF en de FVF van de vezels die liggen in belastingsrichting. Gebaseerd op de hoeveelheid belastingdragend materiaal voor een willekeurig Glare laminaat kan een vermenigvuldigingsfactor worden afgeleid om de $K_R$-curve van het laminaat te verkrijgen uit de standaard $K_R$-curve van het gebruikte aluminium.

Aanvullend is het gat- en scheurgedrag onderzocht van Glare laminaten met interne delingen in de aluminium lagen. Het is onderzocht dat de huidige delingsconfiguraties gebaseerd op overlappende aluminium lagen, geen verdere vermindering van de sterkte tot gevolg heeft van de laminaat sterkte bij aanwezigheid van gaten. In het geval van scheuren kan de aanwezigheid van deze delingsgebieden zelfs een positieve invloed hebben op de reststerkte afhankelijk van de locatie. De conclusie van dit onderzoek is dat de sterkte in geval van gaten en scheuren in Glare met aluminium delingen kunnen worden beschouwd als de sterkte van het Glare zonder de aanwezigheid van deze delingen.

Een studie naar het optreden van wijdverspreide vermoeiingsschade in Glare die aanleiding geeft tot interacties (WFD) vanuit een gat met initiële beschadigingen gaf aan dat de kans hierop uitermate gering is tijdens de geplande levensduur van een vliegtuig, helemaal wanneer deze gaten ook nog zijn gevuld met klinknagels. Aangezien het linken van deze aparte vermoeiingsscheuren niet optreedt is het uiterst zeldzame optreden onderzocht van een impact schade in een rij met klinknagelgaten. Het bleek dat het mogelijk is om het linken van deze impact schade met de gaten (met en zonder vermoeiingsscheuren) alsmede de reststerkte van de constructie te bepalen met behulp van de $K_R$-curve benadering.

Tot slot is een studie uitgevoerd naar de scheurstopping werking van het intern mee lamineren van extra vezel banden of titanium lagen in het Glare laminaat. Het bleek dat het gebruik van extra glasvezel epoxy lagen betere resultaten opleverde. De reststerkte van deze Glare laminaten met interne scheurtoppers van glasvezel banden kan worden voorspeld met de eerder voorgestelde voorspellingsmethodiek voor scheuren. Dit is geverifieerd in ware grootte reststerkte testen van romp panelen waarin scheuren over twee span baaien waren aangebracht.
ABOUT THE AUTHOR

The author was born on July 28, 1966 in Zevenaar, the Netherlands. After graduating from high school in 1985, he started to study Aerospace Engineering at the Delft University of Technology. During this study he worked as a mechanical instructor. He graduated in 1994 by Prof. Dr. ir. J. Schijve and Prof. ir. L.B. Vogelesang on the topic of “The residual strength of various Aircraft materials”. This graduation work was partially carried out at the University of Pisa in Italy. Directly after his graduation, he started his doctoral studies at the Delft University under supervision of Prof. ir. L.B. Vogelesang and Prof. dr. ir. R. de Borst. From October 1997 to January 2000, he worked as project leader within the Dutch Glare Technology Program to support the development of Glare for the large-scale application in aeroplanes. The author was responsible for the development of calculation methods. In August 1999 he started to work as project manager for ADSE and was responsible for the selection and installation of a large C-scan facility at Fokker Aerostuctures b.v., that was necessary to check the quality of large curved Glare panels to be tested in a full-scale A380-like barrel test at EADS in Hamburg. In January 2000, he returned on a part time base to his previous PhD position to finish his thesis work. Since October 2000, he is working as Glare lead engineer at EADS in Hamburg for the A380. He is a part time associate professor in the field of aircraft materials at the Delft University of Technology in the chair of Dr. ir. A. Vlot.
APPENDIX A, $K_G$ DETERMINATION FOR MSD SCENARIOS

Table A-1. Determination of correction factor $\beta_1$ for presence of second crack as a function of physical crack extension.

$\Delta a_{phy} = 0$

$\Delta a_{phy} = a_0 + \Delta a_{phy}$

$2b = \text{constant}$

$t = t - \Delta a_{phy}$

$k = 2 \frac{a_{phy} b}{[2a_{phy} + t](2b + t)}$

$E(k) = \frac{\pi}{2} \sqrt{\left[1 - k^2 \sin^2 \phi \right]} d\phi$

$K(k) = \frac{d\phi}{2 \sqrt{1 - k^2 \sin^2 \phi}}$

$\Delta a_{phy} = \Delta a_{phy} + dx$

with $0 \leq dx \leq t$

$\Delta a_{phy} > t ?$

Curve fit $\beta_1$ vs. $\Delta a_{phy}$

Output: $2^{nd}$ degree polynomial relation for $\beta_1$ and $a_{phy}$
Table A-2, Determination of $K_G$-curve for presence of second crack.

\[
\beta_w = \frac{1}{\cos \frac{\pi a_{\text{eff}}}{W}}
\]

\[
K = \beta_w \sigma_y \sqrt{\pi a_{\text{eff}}}
\]

\[
r_y = \frac{1}{2\pi} \left( \frac{K}{\sigma_y} \right)^2
\]

\[
\Delta a_{\text{phy}} = \Delta a_{\text{eff}} - r_y
\]

\[
\Delta a_{\text{phy}} \neq 0 \quad \text{n} \quad \beta_1 = 0
\]

\[
\beta_1 = f(\Delta a_{\text{phy}}) = f(\Delta a_{\text{eff}} - r_y)
\]

\[
K_{\text{corr}} = \beta_1 K(\Delta a_{\text{eff}})
\]

\[
\Delta a_{\text{eff}} = \Delta a_{\text{eff}} + dx \quad \text{with} \ 0 \ ? \ dx \ ? \ t
\]

Store $K_{\text{corr}}$ vs. $\Delta a_{\text{eff}}$

Output: $K_G$ curve
Appendix A, $K_G$ determination for MSD scenarios

Table A-3, Determination of link-up and fracture.

\[ \sigma, a_{\text{eff}}, W \]

\[ K_{G,\text{corr}} = \beta \sigma \sqrt{a_{\text{eff}} \frac{1}{\cos \left( \frac{\pi a_{\text{eff}}}{W} \right)}} \]

and

\[ \frac{dK_{G,\text{corr}}}{da_{\text{eff}}} = \frac{dK_R}{da_{\text{eff}}} \]

\[ \sigma = n^\text{th} \text{ link-up stress} \]

\[ \sigma = \sigma - d\sigma \]

\[ \sigma = \sigma + d\sigma \]

\[ K_{G,\text{corr}} > K_R? \]

\[ K_R = \text{material property} \]

\[ \text{Restart calculation to calculate } n^{n+1} \text{ link-up stress with new } a_0 \text{ due to link up.} \]