Development of fibre-metal laminates, ARALL and GLARE, new fatigue resistant materials

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NEW FATIGUE RESISTANT MATERIALS

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Abstract: Research on adhesive-bonded sheet metal laminates is described as a predecessor of a new class of materials, the fiber-metal laminates. It is explained why fiber-metal laminates have such a high fatigue resistance, especially against fatigue crack growth. Additional advantages of the barrier function of the fiber layers are indicated. Applications are discussed. Comments are made on developing a material for practical applications.

INTRODUCTION

The development of new materials generally takes many years before structural applications are accepted by designers. This is especially true in the aircraft industry, where some well known attractive properties are: a low specific mass, high stiffness, improved durability (corrosion and fatigue resistance) in view of less maintenance. The aircraft designer is very much aware of the risk of introducing a new material, because much has to be learned about a large variety of properties, see Table 1. New production techniques have to be introduced. Moreover, the durability of the structure should still be satisfactory during its entire economical life time. It requires concerted efforts of various disciplines, and it anyhow starts with investing much time and money. In other words, it is economically risky.

The past decades have shown several examples of successful new materials. Aluminium has replaced steel and wood in many objects, e.g. in prefabricated building structures, sporting articles, medical systems, etc. Important arguments were lower weight, easy production techniques, resistance against environmental effects, etc. Also thermo-plastics (with and without fibers) have replaced several non-plastic materials in lots of simple parts and utensils. On the other hand, there are also examples, where new materials for engineering applications were unsuccessful, or where it has taken many years to arrive at a fully mature application.

For instance, the low static strength of rather cheap mild steel ($S_u = 400$ MPa) could easily be improved, which should allow a reduced weight of originally heavy structures (ships, bridges, etc.). Unfortunately, the fatigue strength of welded structures did not increase
Table 1: Survey of various material properties.

- Mechanical properties
  - $S_{ax}$, $S_{ay}$, $E$, $\delta$
  - blunt notch strength
  - $K_{ic}$, R-curves
  - fatigue resistance (o.a. $S-N$, $da/dN-\Delta K$)

- Durability
  - corrosion, stress corrosion, fretting
  - materials degradation by environment, high temperature
  - erosion
  - impact damage sensitivity

- Physical/chemical properties
  - specific mass
  - thermal expansion, heat conductivity
  - lightning strike
  - flammability, toxicity

- Technological properties
  - machinability
  - cold and hot working properties
  - jointing processes
  - repairability

- Cost-effectivity

Proportionally to the static strength, and the improvement is questionable. A classical example in aeronautics is the development of carbon fiber reinforced epoxies, the so-called advanced composites. They were announced already in the fifties as the fatigue insensitive material, giving 30 to 40% weight saving. In addition, they were claimed to lead to cheaper production (less components). Almost 40 years later, the percentage of the structure of civil transport aircraft made of composites is still rather low. Partly, it should be associated with specific issues, such as impact damage sensitivity, quality control, etc. In a broader scope, an entirely new material implies new design and production philosophies. A new "culture" must be introduced, and that requires time and money.

The situation is different if the new material is a simple replacement of the older material, with minor implications for design and production. This was supposed to be true for the new Al-Li alloys, developed for simple replacements of the traditional Al-alloys 2024-T3 (damage tolerant alloy) and 7075-T6 (high strength). The Al-Li alloys approximately have a 10% lower specific mass and a 10% higher elastic modulus. Other properties should be at least equivalent to those of the older alloys. It turned out that the fatigue crack growth resistance of the Al-Li alloys is not yet satisfactorily solved [1]. Application in civil aircraft is still limited.
The development of the fiber-metal laminates (ARALL and GLARE) started around 1980 on a small scale in the Structures and Materials Laboratory of the Faculty of Aerospace Engineering of the University in Delft [2,3]. At that time the Fokker Aircraft Industries already had a long lasting service experience of aircraft built up from adhesive-bonded metal sheets. They also carried out some exploratory tests with fibers in the adhesive layers in the seventies [4]. The developments in Delft were primarily guided by the aim to obtain a material with a high fatigue resistance, and more especially with a high resistance against fatigue crack growth. It was clear that it required very thin Al-alloy sheets (hardly available at that time) and high strength fibers. Our laboratory was considerably supported by Alcoa (thin sheets), Akzo (fibers) and the 3M-Company (fiber-adhesive prepregs). Test results indicated that ARALL and GLARE can be potential candidates for primary aircraft structures, especially to replace 2024-T3 and 7075-T6 in fatigue critical components. In 1991 Alcoa and Akzo came to a joint venture "Structural Laminates Company", which by now is producing and marketing the fiber-metal laminates. The present paper starts with basic elements of the development of fiber-metal laminates. The fatigue fracture mechanism of the hybrid fiber-metal laminates is emphasized. Structural applications are discussed afterwards.

BASIC ELEMENTS OF THE FIBER-METAL LAMINATES DEVELOPMENT

Fatigue crack growth in adhesive-bonded aluminium laminates

Laminated materials without fibers will be considered first. If a number of sheets are bonded together, a laminated material with a larger thickness is obtained. In the seventies experimental research was carried out on such sheet metal laminates [5,6]. Results are shown in Fig. 1 for through cracks and part-through cracks in central cracked specimens and in lugs (pin loaded hole). The crack growth life for the central through crack is about 60 % larger for the laminated material as compared to the solid material. The longer fatigue life results from a sheet thickness effect, see Fig. 2. In the laminated material with a through crack, fatigue crack growth occurs simultaneously in all thin sheets with a growth rate, corresponding to the thin sheet material. The adhesive layer has a very low stiffness, the modulus is more than 25 times lower than for the Al-alloy. Heiser and Hertzberg [7] refer
to the crack arrester and the crack divider concepts for materials with relatively weak interfaces. Here the adhesive layers behave as crack dividers. As a result, the cracks are growing independently in all layers of the laminate without mutual interference. Crack growth in thin sheet materials is slower than in plate material, because of the plane-stress effect.

In structural components cracks usually start as part through cracks. Figure 1 then shows a completely different picture. The crack growth life of the laminated material in comparison to the solid material is 10 times longer for the central cracked specimens and 5 times longer for the lugs. The large improvement for part-through cracks seems to be logical, because the adhesive layers are a barrier for crack growth in the thickness direction. This obvious argument is not fully correct, and it is also incomplete. A crack, which has penetrated one sheet thickness, is meeting a low stiffness adhesive, which reduces the stress concentration in the following sheet. Moreover, a little bit of delamination was observed in the adhesive layer, see Fig. 3. It implies crack branching for crack growth in the thickness direction, which further reduces the stress concentration in the subsequent sheet. Crack reinitiation in that sheet will be considerably delayed. The adhesive layer has the function of a crack arrester.

A still more important aspect is illustrated by Fig. 4, which shows crack growth in the 1st, 2nd and 4th sheet of the laminate, as started by a surface crack in the first sheet. Initially, very slow crack growth occurs in the first sheet, if compared to crack growth in a single sheet. It should be explained by a significant restraint on crack opening, because the other sheets are still uncracked. As a consequence, the stress intensity factor K will be significantly reduced as the crack becomes longer. The effect on the growth rate is evident in Fig. 5. Only after crack initiation in the 2nd sheet, and later on in the other sheets, crack growth acceleration occurs. The solid material exhibits the accelerated growth right away.

It is noteworthy that the natural fatigue cracks in the lug specimens (Fig. 1) also indicate a significant life improvement for the laminated material (4 times). Crack initiation in a lug is assisted by fretting corrosion. It does occur at different locations along the bore of the hole, at different moments. As a result, a fairly chaotic crack pattern is obtained as long as the cracks are small. In the laminated material it implies that the fatigue cracks have not the same length in all layers. Moreover, they occur at staggered positions. Interference between the cracks in the five layers will occur, which leads to a considerably delayed crack growth.
In summary, the investigations on the sheet-metal laminates have considerably stimulated the development of the fibre-metal laminates, i.e. laminated sheet material with strong fibers in the adhesive layers.

**Fiber-metal laminates**

In the seventies Fokker carried out CA tests on sheet materials with fibres [4]. Three 1-mm 2024-T3 sheets were bonded together with either carbon fibres or aramid fibres in the adhesive layers. The same material was tested in Delft under flight-simulation loading. The CA results were partly promising, especially for carbon fibres. However, in flight-simulation tests the improvement was marginal, due to the occurrence of fibre failure. Some essential steps to arrive at the new fiber-metal laminates were necessary:

 mucho thinner metal sheets to raise the fibre/metal ratio, UD fibers instead of weaves with a metal adhesive as the matrix material for the fibers, and an increased number of fibre-layer/metal-sheet interfaces.

An additional step for improvement is:

Post-stretching of the fibre-metal laminates to introduce a favourable residual stress system.

The fibre-metal laminates ARALL and GLARE are built up from very thin Al-alloy sheets (0.2-0.4 mm) and intermediate adhesive layers with unidirectional high-strength fibres (aramid fibres in ARALL and advanced glass fibres in GLARE). The constituents are indicated in Fig.6. Different compositions are possible, characterized by the type of Al-alloy, sheet metal thickness, number of layers, type of fibers and fiber direction, and an additional post-stretching if applicable. The laminates which are commercially available at the moment are listed in table 2. There are four grades with unidirectional fiber layers (UD) and two grades with cross ply (0/90°) fiber layers. The latter ones were developed for biaxial loading of pressurized fuselages. The picture in Fig.7 illustrates the cross ply of the prepregs. The fiber-metal laminates can be delivered in several thicknesses by increasing the number of metal sheet layers and selecting on optimal thickness of the individual layers. A survey of some properties is given in Table 3.
Table 2: Commercially available fiber-metal laminates [12].

<table>
<thead>
<tr>
<th></th>
<th>ARALL</th>
<th>GLARE</th>
</tr>
</thead>
<tbody>
<tr>
<td>2</td>
<td>3</td>
<td>1</td>
</tr>
<tr>
<td>Metal type</td>
<td>2024-T3</td>
<td>7475-T76</td>
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<tr>
<td>Metal thickness (mm)</td>
<td>0.3</td>
<td>0.3</td>
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<tr>
<td>Fiber layer (mm)</td>
<td>0.22</td>
<td>0.22</td>
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<tr>
<td>Fiber direction</td>
<td>UD</td>
<td>UD</td>
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Table 3: Properties of some fiber-metal laminates [8]

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<thead>
<tr>
<th>Property</th>
<th>2024-T3</th>
<th>ARALL 3 3/2</th>
<th>GLARE 3 3/2</th>
<th>GLARE 4 3/2</th>
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<tr>
<td>Tensile ultimate strength (MPa)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>L</td>
<td>455</td>
<td>765</td>
<td>717</td>
<td>1027</td>
</tr>
<tr>
<td>LT</td>
<td>448</td>
<td>352</td>
<td>716</td>
<td>607</td>
</tr>
<tr>
<td>Tensile yield strength (MPa)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>L</td>
<td>359</td>
<td>565</td>
<td>305</td>
<td>352</td>
</tr>
<tr>
<td>LT</td>
<td>324</td>
<td>290</td>
<td>283</td>
<td>255</td>
</tr>
<tr>
<td>Elastic tensile modulus (GPa)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>L</td>
<td>72</td>
<td>68</td>
<td>58</td>
<td>57</td>
</tr>
<tr>
<td>LT</td>
<td>72</td>
<td>49</td>
<td>58</td>
<td>50</td>
</tr>
<tr>
<td>Ultimate failure strain (%)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>L</td>
<td>19</td>
<td>1.8</td>
<td>4.7</td>
<td>4.7</td>
</tr>
<tr>
<td>LT</td>
<td>19</td>
<td>6.4</td>
<td>4.7</td>
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<tr>
<td>Bearing ultimate strength (MPa)</td>
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<td></td>
<td></td>
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<tr>
<td>L</td>
<td>890</td>
<td>669</td>
<td>819</td>
<td></td>
</tr>
<tr>
<td>LT</td>
<td></td>
<td>655</td>
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<tr>
<td>Gross blunt notch strength (MPa)</td>
<td></td>
<td></td>
<td></td>
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</tr>
<tr>
<td>(circular hole, W=100 mm, D=25 mm)</td>
<td>311</td>
<td>409</td>
<td>372</td>
<td>445</td>
</tr>
<tr>
<td>(circular hole, W=100 mm, D=25 mm)</td>
<td>311</td>
<td>264</td>
<td>372</td>
<td>311</td>
</tr>
<tr>
<td>Gross sharp notch strength (MPa)</td>
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<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>(center crack, 2a=25 mm)</td>
<td></td>
<td></td>
<td></td>
<td></td>
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<tr>
<td>(center crack, 2a=25 mm)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Density (g/cm³)</td>
<td>2.78</td>
<td>2.33</td>
<td>2.52</td>
<td>2.44</td>
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</table>

1) MIL HDBK 5, "S" basis allowables
2) Structural Laminates Company, "typical" values

The data in table 3 illustrate several remarkable characteristics of the fiber-metal laminates:
- high static strength in the fiber direction, associated with the high strength of the fibers,
- anisotropic behaviour, accept for GLARE 3 with a 50/50 cross ply,
- somewhat lower stiffness than for the Al-alloys,
- a limited elongation at failure, due to the limited failure strain of the fibers, which
is larger for the glass fibers (GLARE) than for the aramid fibers (ARALL),
a lower density than for the Al-alloys (more for ARALL than for GLARE).

The fatigue mechanism of ARALL and GLARE
The principal features of the fibre-metal laminates can be explained by referring to Fig.8. It shows that fatigue cracks in the metal layers are bridged by the fibres, which is advantageous for two reasons:
1. Crack bridging fibres imply a significant restraint on crack tip opening,
2. Unbroken fibres in the cracked area imply that part of the load in the Al-sheets is still transmitted through the crack.

As a result, there is a most significant reduction on the stress intensity factor $K$ in the Al-layers. The load in the crack bridging fibres will cause shear stresses on the interfaces between the fibre layers and the Al-sheets as analyzed by Marissen [9]. Due to the shear stresses some delamination will occur at the crack edges, see Fig.9, which shows the delaminated area as indicated by an ultrasonic C-scan. However, for a well balanced fibre to metal ratio and sufficient interfaces, the delamination is small and acceptable.

ARALL and GLARE sheets are produced with standard bonding technology. After the hot-curing cycle in the autoclave ($120^\circ$C), the fibre-metal laminates carry a residual stress system over the thickness of the material, with a tensile stress in the Al-sheets and compression in the fibres, see Fig.10. A post-stretching operation of the fiber-metal laminate sheet after the curing cycle (ca. 0.4% of plastic strain) reverses the residual stress system (Fig.10). Then a tension stress occurs in the fibres and compression in the Al-sheets [10]. In the as-cured condition the crack bridging mechanism gives already an impressive crack growth resistance, but it is still further improved by the post-stretching operation. Illustrative results are presented in Fig.11. It should be noted that the results for 2024-T3 apply to a lower stress level. Several tests were carried out in which crack arrest occurred at a short crack length (a few millimetres). Marissen [9] developed an analytical model, which predicted a decreasing crack growth rate in agreement with experimental observations.

Fractographic observations in the electron microscope (SEM) indicated that the adhesion between the aramid fibres and the adhesive matrix was rather weak. A very high fatigue
crack growth resistance was still obtained. Aramid fibre failure can occur in the as cured ARALL under certain loading conditions, especially under low $S_{\text{min}}$. The aramid fibres in the cracked area are pulled out of the matrix at maximum load. During unloading this process is not fully reversible, and then the fibres are buckled. If that occurs during many cycles, the fibres are damaged and fibre failure will occur at maximum load. Nevertheless, there are still unbroken aramid fibres behind the crack tip, which take care of a K-reduction and slow crack growth. Fibre failure does not occur in glass fibres. Various aspects of the fatigue phenomenon of ARALL and GLARE were extensively studied by Roebroeks [11,12]. It showed that fatigue in these hybrid materials is a rather complex phenomenon. However, it is thought that the fatigue mechanism is reasonably well understood, thanks to observations made by many undergraduate and post-graduate students. It should be pointed out that microscopic investigations on the fatigue mechanism of fiber-metal laminates have essentially contributed to understanding of the material behaviour and the improvement of the material quality.

Additional barrier functions of the prepreg layers
As pointed out before, the fiber-adhesive layers were primarily introduced to obtain a high fatigue crack growth resistance. At the same time the prepreg layers also behave as a barrier, which offers significant advantages with respect to:

1. fatigue crack growth of part-through cracks,
2. corrosion attack, and
3. fire prevention.

Fatigue crack growth of part-through cracks
Fatigue in a structure usually starts as a part-through crack, either from surface damage, or at notches. In a recent study, Fredell et al. [8] deliberately introduced surface damage in ARALL 3 and in GLARE 3 and 4. (3/2 lay up). Fatigue tests were carried out on specimens with 40 mm long machined scratches ($S_{\text{max}}=120$ and 100 MPa, $R=0$). Even for a scratch, that fully cut through a surface layer, fatigue crack growth was negligible, and penetration of the crack into the second layer was not observed. The prepreg behaved as an efficient barrier.
A more complex case of the prepreg barrier function occurs in fatigue of riveted lap joints. These joints are of prime importance for the fatigue life of a pressurized fuselage. A riveted lap joint is a complex joint due to the load transmission by the rivet, secondary bending induced by the eccentricity and fretting corrosion. Observations on fatigue cracks in riveted lap joints of GLARE 3 (0/90 cross ply) have recently been made by Roebroeks [12]. Fatigue tests were interrupted after different numbers of cycles. The specimens were then pulled to failure, and the amount of fatigue cracking could be observed. At the same time, the results indicate strength degradation due to fatigue. In other test series cracks were detected by an eddy current inspection. With this method it is possible to indicate separately cracks in the three Al-alloy layers of the GLARE material. In all tests cracks started in the inner Al-layer, i.e. at the mating surface of the lap joint. That should be expected in view of secondary bending, fretting corrosion and stress concentrations. When the first layer was fully cracked, the cracks in the second layer were still very small (≈ 1 mm), and the third layer was yet uncracked. The cracking behaviour of a comparable 2024-T3 riveted lap joint was fully different. Cracks initiated somewhat earlier as part-through cracks, which became a through crack at a crack length of a few millimetres. After some more crack growth specimen failure occurred. As a result a catastrophic strength reduction of the 2024-T3 lap joints was observed towards the end of the fatigue life, see Fig.12. Such a dramatic reduction was almost absent for the GLARE 3 riveted joints.

**Corrosion attack**

Surface corrosion can not always be prevented in an aircraft structure. Two problems should be mentioned here. (i) Corrosion may penetrate the thickness of the material, and thus locally degrade the integrity of a component. (ii) Secondly, corrosion pits can be the initiation point of fatigue cracks.

The corrosion resistance of the fiber-metal laminates has been tested under severe corrosion conditions (e.g. in a salt spray cabinet). The corrosion resistance appears to be quite similar to the resistance of the constituent Al-alloys with the same surface treatment. However, there is an important difference: corrosive attack is stopped at the first prepreg layer. The second Al-layer remains unaffected, thanks to the prepreg corrosion barrier.

Fatigue tests were carried out on fiber-metal laminate specimens with locally applied severe corrosion damage [13]. Crack growth started at the corrosion pits, but it slowed down outside the corroded area. The fatigue cracks did not penetrate into the second Al-layer.
Fire prevention

Some preliminary tests have shown that a fiber-metal laminate sheet, exposed to a burner at one side (flame temperature 1100°C), could survive the fire significantly longer than an Al-alloy sheet. The aluminium front layer of GLARE, exposed to the burner, disintegrated quite rapidly (melting point slightly over 500°C), but the first prepreg layer with glass fibers (melting point exceeding 1100°C) could survive for a considerable time. A relatively low temperature was maintained at the other side of the GLARE sheet for more than 15 minutes. The improved fire resistance should be considered for passenger aircraft in view of the short escape time during an accidental landing when the aircraft can catch fire [12].

APPLICATION OF FIBER-METAL LAMINATES IN AIRCRAFT STRUCTURES

Data as shown in Table 3 are instructive for the designer. However, information about various other properties (see Table 1) must also be considered. Because the fiber-metal laminates are produced as sheet material, it implies that the designer has to built up a structure from sheet material. Experience has shown that the fiber-metal laminates can be machined in much the same way as the traditional Al-alloys (e.g. cutting, drilling, contour milling). There are a few precautions to be kept in mind in order to prevent local edge delamination. The fiber-metal laminates can also be riveted and bonded. Bonding actually implies a second bonding cycle for the material, which is not objectionable. Sheet metal bending operations can be applied to the fiber-metal laminates, which is necessary for the production of stiffeners. The test panel in Fig.13 has four Z-stiffeners made from ARALL with three Al layers. Minimum bend radii have been determined.

The fiber-metal laminates were primarily developed to obtain a material with a high fatigue crack growth resistance. It implies that application to the aircraft structure can lead to:
- weight saving
- improved safety (by improved damage tolerance)
- longer inspection periods
- production simplifications

The last topic was recently proposed by Roebroeks [12], who pointed out that sophisticated design details in view of fatigue may become unnecessary.
Fiber-metal laminates should be considered in the first place for aircraft components of a large size:

1. The pressurized fuselage
2. The tension skin of the wing
3. Tailplanes

In small fatigue critical parts, the weight saving may be limited, but the high life time, and more in particular long inspection periods can be very attractive. An example is the single load path lug connections between larger parts of an aircraft structure.

Extensive test series and design studies were carried out by Vogelesang, Gunnink and others [10,14-16]. Designers want empirical evidence on the fatigue behaviour of joints and structures under load histories of practical relevance. Various tests on riveted joints, lugs and stiffened panels have been carried out. Outstanding results were obtained in two full-scale flight simulation fatigue tests on F-27 wing tension skin panels of ARALL, carried out by Fokker [17,18]. The ARALL wing structure was about 30 % lighter than the original F-27 structure, while the fatigue life was considerably longer. Here we will consider two examples of experimental evidence of joints. They are considered to be characteristic for the behaviour of ARALL and GLARE.

Skin panels with four stiffeners were entirely produced from ARALL. Finger plates were adopted as a design for a joint in the wing structure, see Fig.13. The fatigue life was excellent. Cracking occurred at the finger tips in the flange of the stiffener, but it stopped after some crack growth. Sectioning of the finger tip shows that the fatigue crack occurred in the upper layer of the stringer flange. After it reached the fibre layer some delamination occurred. The important consequence is that the delamination actually implies crack branching in a similar way as depicted in Fig.3. It is well known from fracture mechanics that crack branching has a favourable effect on crack growth in a homogeneous material. Here, in the hybrid material the branching effect is even more favourable because the stress concentrating effect of the finger tip is removed from the tip and spread over a much larger area. It might well be said that this is smart material behaviour. Such a favourable behaviour is possible as a result of the layered structure of the material and the possibility to accept some delamination.
The second example is the result of a flight-simulation test on a lug made of GLARE 1 (post-stretched) [19]. The lug shown in Fig.14 is identical to a lug of the CN-235 aircraft. It is part of the connection between the wing and the fuselage. Usually a low stress level is adopted for such an important joint. The load history consisted of blocks of 1000 flights with 10 different types of flights in a random sequence. After simulating 120,000 flights no indications of cracks were obtained. All load levels were then increased by a factor of 1.25 and again 120,000 flights were simulated without any cracking. This was repeated 3 times, bringing the load level to $1.25^3 = 1.95$ times the design load. Failure of the lug still did not occur, but fracture occurred in the aluminium alloy clamping of the lug after 452,000 flights (92,000 flights at the maximum load level). Three small cracks could be detected in the lug by an eddy current inspection. It is obvious that the test shows a large extra safety margin and favourable indications for long inspection periods.

ARALL-3 is currently in production as the skin material of the aft cargo door of the MacDonald-Douglas C-17 aircraft. Two lay-ups are used, viz. 3/2 and 4/3. The weight saving on the door is about 75 kg (27%)[20].

At the moment Airbus [21,22] is carrying out a full-scale barrel test to simulate a pressurized fuselage fatigue load history. Results until now indicate a significantly slower crack growth in GLARE 3 as compared to 2024-T3, although $t_p$ (thickness x specific mass) is 21% lower.

**DISCUSSION**

The present paper surveys various aspects of the development of new fiber-metal laminates (ARALL and GLARE), which are a class of new materials for fatigue critical components of the aircraft structure. The most outstanding property is the exceptionally slow growth of fatigue cracks, which can lead to significant weight savings, longer inspection periods and additional safety. The fiber-metal laminates, now commercially available, seem to be a serious candidate for several applications. Economical consequences for the aircraft industry and for the airlines are obviously present, but this aspect is beyond the scope of the present paper. However, as for any new material, application requires development studies to explore the various technological and design requirements involved. It is not simply a matter of some handbook data on mechanical and physical properties. Fortunately, a lot of work has already been done.
The above paragraph is hinting towards another perspective of this paper. The development of fiber-metal laminates started in a University laboratory and not in the materials industry. It occurred in a laboratory of the Faculty of Aerospace Engineering by aircraft engineers, who had some knowledge of aircraft structures, aircraft fatigue loads, production techniques and the behaviour of a structure in service. It is for that reason, that almost from the beginning tests were carried out also on representative structural elements under realistic service loads. The same applies to production problems. It was also realized that materials can be accepted only if its behaviour is well understood in terms of failure mechanisms and consequences of production techniques. Have a close look on the material, see what happens and try to understand. This is the more relevant in view of the hybrid character of the fiber-metal laminates. It is a complex world of research drivers, but it is fascinating.

CONCLUSIONS

1. Fiber-metal laminates (ARALL and GLARE) are a new class of materials with a high resistance against fatigue crack growth. As a result the material has excellent damage tolerance properties, including a slow degradation of residual strength after a very long fatigue life. As a result, the fiber-metal laminates are serious candidates for fatigue critical aircraft components. The fiber-metal laminates can contribute to weight saving, longer fatigue lives and longer inspection intervals.

2. From a design and production point of view, the fiber-metal laminates have much more affinity to metallic sheet materials than to full composites. They can relatively easily replace sheet materials.

3. The fatigue fracture phenomenon of the fiber-metal laminates is reasonably well understood in a qualitative sense. Both microscopical observations and fracture mechanics considerations have contributed to the understanding. The knowledge was essential in optimizing the composition of the fiber-metal laminates.

4. The fiber-metal laminates are attractive, not only in view of fatigue resistance. The barrier function of the prepreg gives additional advantages with respect to corrosion resistance and fire protection.

5. The development of a new material for engineering applications should include all kinds of design and component manufacturing problems at an early stage of the development. This has been done for ARALL and GLARE.
Acknowledgment: Special thanks are due to Professor L.B. Vogelesang and Dr. G. Roebroeks for helpful discussions and enthusiastic support.

REFERENCES

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<th>cross section</th>
<th>crack growth life (a = 2,\text{mm}) to failure (10^4)</th>
<th>crack growth life (a = 2,\text{mm}) to failure (10^5)</th>
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<tr>
<td>spark eroded notch</td>
<td>[Diagram]</td>
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<td>1.6</td>
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<td>central crack</td>
<td>[Diagram]</td>
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<td>natural cracks</td>
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Fig. 1 Crack growth life in solid material \(t = 5\,\text{mm}\) and laminated material \(5\times1\,\text{mm}\), 2024-T3 alloy, central cracked specimen \(W = 100\,\text{mm}\) and lug specimen \(W = 60\,\text{mm}, d = 25\,\text{mm}\) [6].

![Diagram](image)

Fig. 2 Crack growth in solid and laminated sheet material. Center cracked specimens [5]
Fig. 3  Growth of part through crack in the thickness direction. Some delamination before crack initiation in the second layer.

Fig. 4  Crack growth of part through crack in laminated material (five 1 mm sheets, 2024-T3). Retarded crack growth in 1st sheet due to crack opening restraint by the other sheets [5].
Fig. 5  Growth rates for part through cracks in laminated material and solid material [5].

Fig. 6  Fiber-metal laminate composition (3/2 lay-up).
Fig. 7  Cross section of "biaxial" GLARE 3 with unidirectional fibre layers (prepregs) in the L and T directions (Courtesy: Structural Laminates Company).

Fig. 8  Reduced K due to crack bridging and load transmission through the crack.
Fig. 9  Delaminated area around central fatigue crack, revealed after removing Al-surface layer by chemical etching. ARALL 2, $S_{max} = 180$ MPa, $R = 1/3$, no fiber failure [12].

![Image of delaminated area]

Fig. 10  Residual stresses in ARALL, effect of 0.4% post-stretching [10].

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![Diagram of residual stresses]
Fig. 11 Effect of post-stretching on crack growth in ARALL under flight-simulation loading. Comparison to 2024-T3. $S_{mf}$ = mean stress in flight, note the higher $S_{mf}$ in ARALL [10].

Fig. 12 Residual strength of riveted lap joints after fatigue [13].
Fig. 13  ARALL stiffened tension skin structure. Fatigue crack initiation at finger tip in top sheet only, followed by some delamination = crack branching.

Fig. 14  A lug for a wing-fuselage attachment made of GLARE 1 [19]. Lug thickness 19.0 mm, hole diameter 44.45 mm.