Chapter 21
Fatigue Resistance of Fiber-Metal Laminates

21.1 Introduction

The history of mankind has been characterized by an interesting development of materials originally used for tools, housing, weapons and other needs. Initially wood and clay were available materials, followed by stone (Stone Age) and much later, but still about 3000 years ago, by iron (Iron Age). Apart from the availability of building materials, the production and working processes were also decisive for the success of a material (which in fact is still true in the present time). In the past, material properties were related to strength, stiffness, and durability. It was recognized that stones could carry high-compression loads but not high-tension loads. Later, the engineering approach to new materials included the development of composite materials with the aim to combine favorable properties of different materials into a single composite material. Reinforced concrete is a well-known example and fiber-reinforced plastics another typical case. Several composite materials were designed for specific applications. Developing materials and designing composite materials for specific purposes is often essential for advanced applications. A noteworthy example is offered by modern ceramics for high-temperature applications with the space shuttle as an outstanding example.

The history of aircraft structures has seen a large variation of different materials. In the very beginning, the time of the Wright brothers, the materials of the aircraft structure were wood, steel and linen. In the first half of the 20th century, design criteria for materials used in aircraft structures were associated with low weight and sufficient strength. The introduction of aluminium alloys in the late 1920s was a kind of a revolution because it drastically changed structural design concepts. The efficiency of aircraft structures was significantly improved. Aluminium alloys also penetrated many other applications because of the low specific weight, e.g. in many household appliances.

In the second half of the 20th century, developments of aircraft materials had to face more criteria than just low weight and sufficient strength. It turned out that civil transport aircraft were going to be used for a service life well over 20 years which poses requirements for durability of the aircraft structure, in particular with respect to fatigue and corrosion. However, the aging aircraft problem also included safety aspects. Damage to aircraft structures can include fatigue cracks, corrosion damage, impact damage and other kinds of incidental damage. The danger is that cracked parts of the structure may no longer have sufficient strength. This has led to so-called damage tolerance requirements. Cracks should not grow too fast in order to detect the damage during periodic inspections of the aircraft, see the discussion on Figure 20.3 in the previous chapter. The designer can introduce structural elements to obtain crack growth retardation or even crack arrest. *But a different approach is to develop a material which has a high crack growth resistance as an inherent material property*. This was the basic idea for the development of the fiber-metal laminates. A number of thin sheets was bonded to a single laminate with long high-strength fibers embedded in the intermediate adhesive layers. The development of fiber-metal laminates started in the late seventies in the laboratory of Structures and Material of the Faculty of Aerospace Engineering of the Delft University of Technology. Originally aramid fibers were adopted and the commercial name of the laminate was Arall (Aramid reinforced aluminium laminates). Some ten years later Glare (Glass reinforced) was introduced in view of certain shortcomings of the aramid fibers. The aramid fibers were replaced by advanced glass fibers. Both Arall and Glare were built up with thin aluminium alloy sheets. The history is described in [1]. More recently a new version of the laminate family was developed for thick plates in aircraft wing structures with the name CentrAl [2].

A different evolution of aircraft materials occurred in the 2nd half of the 20th century, i.e. the development of the composite materials, an epoxy matrix with long fibers, usually carbon fibers with a high strength and stiffness, the so-called black composites, also referred to as advanced composites. The structural application of these composites has encountered entirely different problems because of essentially different design concepts, production techniques and material properties. For these reasons the black composites are not discussed in this textbook.

In the present chapter, fatigue crack growth in laminated sheet material without fibers is discussed in Section 21.2. The major topic is fatigue of fiber-metal laminates covered in Section 21.3. Other properties and production aspects of Glare are briefly addressed in Section 21.4. The chapter is completed with some general remarks.

21.2 Laminated sheet material without fibers

The development of the fiber-metal laminates (Arall and Glare) was preceded by research on fatigue of laminated sheet material without fibers. These investigations were stimulated by the successful application of adhesive bonded aircraft structures designed by the Fokker Aircraft Industries since the 1960s. The research on laminated sheets of aluminum alloys has been instructive for the application of bonded structures in general, but also for the development of the fiber-metal laminates starting around 1980. The experience is briefly summarized in this section.

Various fatigue crack growth tests were carried out on sheet laminate specimens [3]. Illustrative results of crack growth under CA loading are shown in Figure 21.1 by a comparison of the crack growth curves for through cracks in a thick specimen (thickness 5 mm, 0.2 inch), a specimen of laminated material obtained by adhesive bonding of 5 thin sheets (thickness 5×1 mm) and a single sheet specimen (thickness 1 mm). The graph shows that the crack growth life is about 60% larger for the laminated material as compared to the solid material. The longer fatigue life results from a sheet thickness effect. In the laminated material with a through crack, crack growth occurs simultaneously in all thin sheets in a similar way as in the single thin sheet specimen. The intermediate adhesive layer has a very low stiffness, the elastic modulus is more than 25 times lower than for the aluminium alloy. As

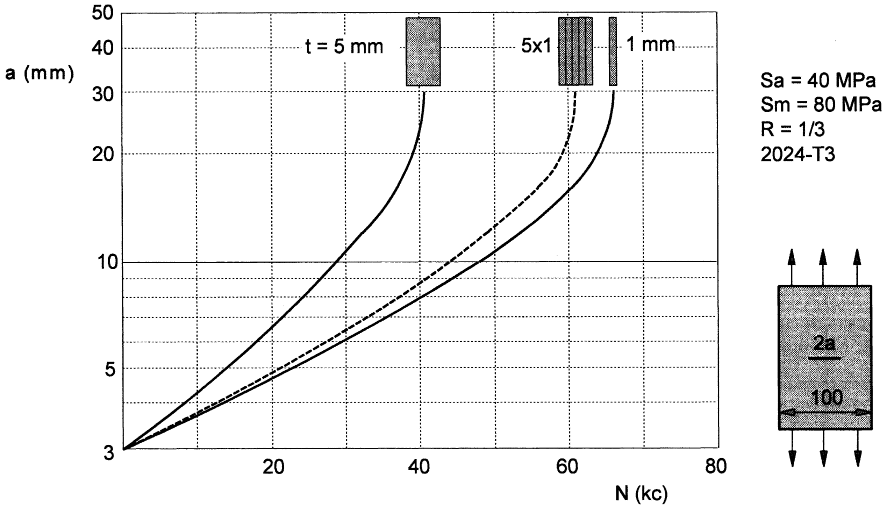


Fig. 21.1 Fatigue crack growth curves of solid and laminated material [3].

a result, fatigue cracks can grow independently in all layers of the laminate without mutual interference. The results confirm the tendency to faster crack growth under a more predominant plane strain situation at the crack tip in thicker material (see the discussion in Section 8.4.2). Thin sheets exhibit a better crack growth resistance.

Completely different results were obtained for part-through surface cracks as illustrated by Figure 21.2. A solid 5 mm thick specimen was tested with a semi-elliptical surface crack, crack depth 1 mm. The crack growth life is about 30% larger than for the through crack because the elliptical crack must first grow through the thickness. However, for the laminated specimen with a crack in the surface layer only, the fatigue life is approximately 10 times larger than the crack growth life obtained for the through crack. The same trend was observed in fatigue tests on lug specimens provided with a 1 mm deep saw cut as an initial through crack. The results in Figure 21.2 indicate a 1.5 times longer crack growth life for the laminated lug as compared to the life for the solid lug; whereas, the crack growth life for the corner cracks is about 5 times longer for the laminated material. Lug specimens were also tested without crack starter notches. The fatigue life then showed a good deal of scatter which is associated with crack initiation by fretting corrosion inside the hole. Cracks started at both sides of the hole, while several crack nuclei were observed at each side of the hole, not exactly in the same cracking plane. As a consequence, radial ridges occurred in the solid material where nuclei overlap. The tortuous crack front contributes to

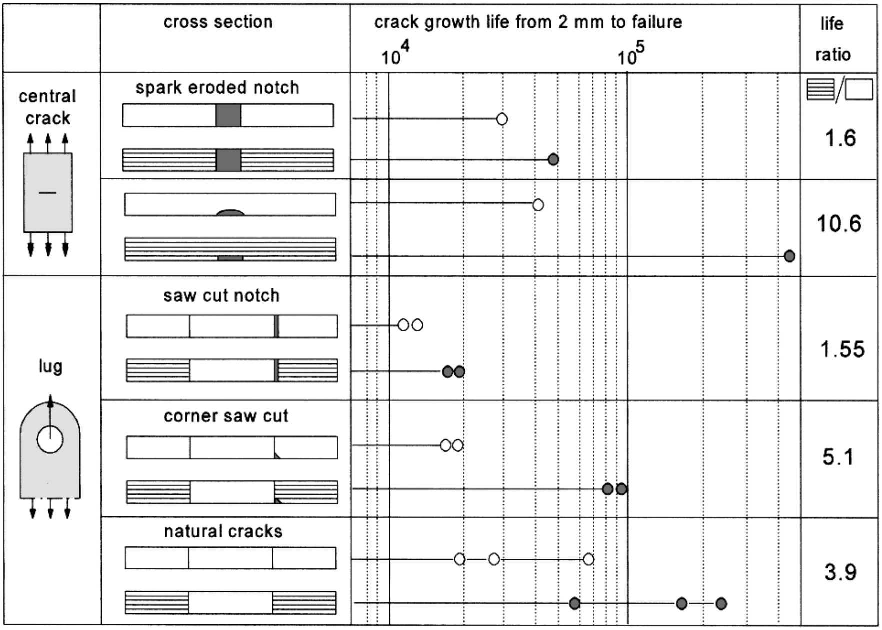


Fig. 21.2 Crack growth lives for solid material (t = 5 mm, 0.2 inch) and laminated material (5 × 1 mm, 5 × 0.04 inch). Central cracked specimens (W = 100 mm, 4 inch) and lug specimens (W = 60 mm, 2.4 inch, D = 25 mm, 1 inch), material: alluminium alloy 2024-T3 [3].

more scatter. In the laminated lug, cracks did not start simultaneously in all sheets, which again caused significant scatter. Anyway, the average fatigue life of the laminated lug is considerably larger (about four times) than for the lug of the solid material.

The large improvement of crack growth lives for part-through cracks seems to be logical because the adhesive layers might be a barrier for crack growth in the thickness direction. However, this argument does not correctly recognize the crack growth delaying effect. In addition to the visible crack growth in the surface sheet, crack growth in the second and the fourth sheet was measured by a special electrical potential drop method. The results are shown in Figure 21.3. Initially, very slow crack growth occurred in the first sheet only, much slower than compared to crack growth in a single sheet specimen. This should be explained by a significant restraint on crack opening in the first sheet because the other sheets are still uncracked. As a consequence, the stress intensity factor K will decrease as the crack becomes longer. Only after crack initiation in the second sheet, and afterwards in the other sheets, crack growth acceleration occurs. This finding

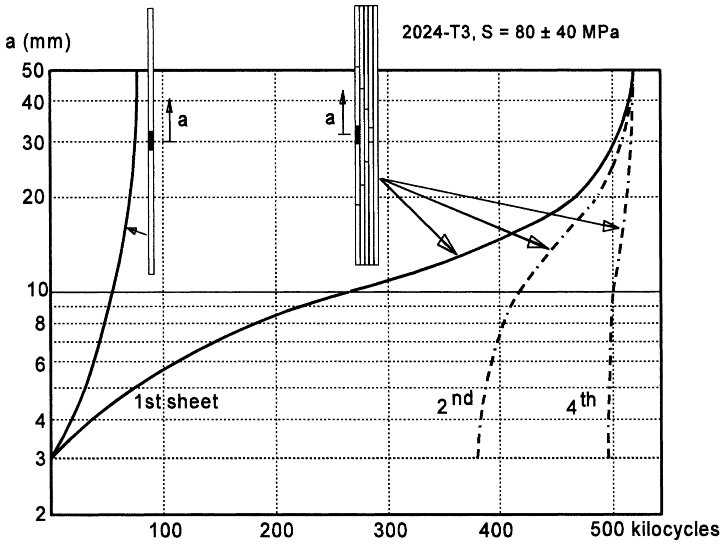


Fig. 21.3 Crack growth from a part through crack in laminated sheet material (5 × 1 mm). Slow crack growth in the first sheet due to crack opening restraint by the other sheets [3].

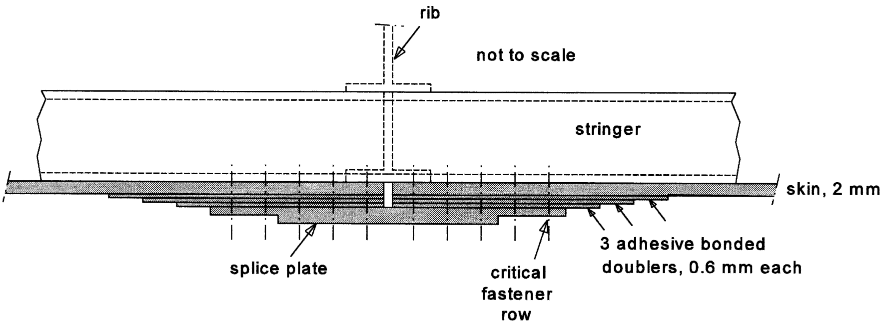


Fig. 21.4 A skin joint with adhesive bonded doublers in an aircraft wing tension skin [4].

on the sheet-metal laminates has been instructive for the development of the fiber-metal laminates.

A similar observation was made on crack growth developments in full-scale fatigue tests on wing tension skin panels (length 8.3 m, 27.3 ft) [4]. Two identical skin joints were present in the panels. As shown by Figure 21.4, the joint consisted of an outside splice plate and continuous inside stringers. The skin joint was locally reinforced with three adhesively bonded and staggered doublers. Invisible crack growth occurred at the rivet holes of the end row of rivets, a classical location for fatigue crack nucleation (Chapter 18). Cracks were observed after 40 to 50% of the total life of the

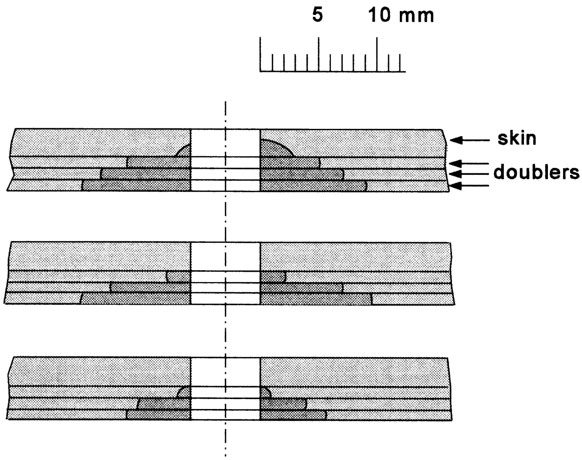


Fig. 21.5 Fatigue crack nuclei at three fastener holes of the joint in Figure 21.4. Small or no crack nuclei in skin.

panel defined by failure at another more critical location. Crack growth was followed by X-ray inspections. The cracks first occurred in the outer doubler, later in the inner doublers, and finally in the skin. This was confirmed by crack nuclei configurations revealed after opening the fracture at the end of the test, see Figure 21.5. Apparently, cracking developed slowly thanks to lamination of the material. Much faster crack growth would have occurred in an integrally machined wing structure.

21.3 Fiber-metal laminates Arall and Glare

The principle ideas of the fiber-metal laminates are discussed first, followed by a section on fatigue crack nucleation and crack growth, and another section on some experience of Glare in full-scale structural components.

21.3.1 The fiber-metal laminate concept

It has been shown in the previous section that crack growth in a single sheet of a laminated material was slowed down after some crack growth because the other sheets restrain crack opening and thus reduce the stress intensity at the crack tip. But this mechanism cannot be active for a through crack when cracks in all sheets of the laminate are growing simultaneously. One of the

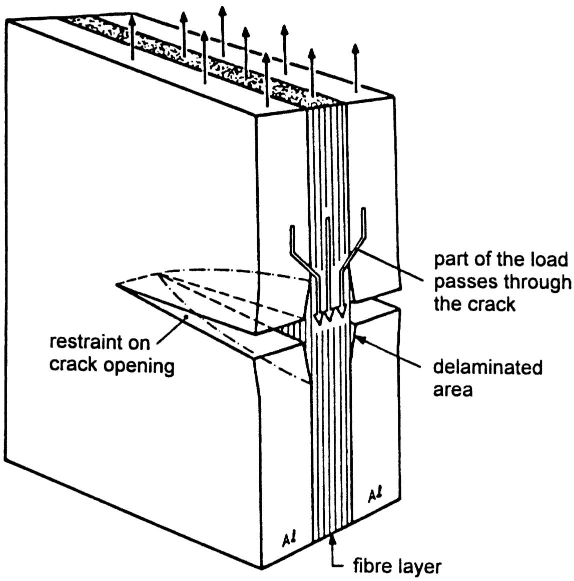


Fig. 21.6 Unbroken fibers in the wake of the crack are restraining crack opening. Part of the load is transmitted by the fibers through the cracked area.

basic ideas of the development of fiber-metal laminates is to restrain crack opening of through cracks by unbroken fibers in the wake of a fatigue crack. This situation is illustrated in Figure 21.6 for a laminate of two sheets with an intermediate adhesive layer as a matrix for uni-directional fibers in the loading direction. If the fibers would not be present, the remote load must be transmitted around the crack. However, as shown in Figure 21.6, the fatigue cracks in the metal layers are bridged by the fibers, which is advantageous: (i) the crack bridging fibers exert a significant restraint on crack tip opening, and (ii) the unbroken fibers in the cracked area imply that part of the load in the aluminium alloy sheets is still transmitted through the crack. As a result, a most significant reduction on the stress intensity factor K will occur.

The load in the crack bridging fibers are causing a shear stress on the interfaces between the fiber layers and the sheets, a problem first analyzed by Marissen [5]. Due to the shear stress some delamination will occur along the crack edges. A second basic idea of the fiber-metal laminates is that the shear stress can be significantly reduced by creating more interfaces. It has been accomplished by implementing a larger number of metal sheets with a low thickness. Typical values are 0.3 and 0.4 mm (0.012 and 0.016 inch). A certain laminate thickness is then obtained by more thin sheets and thus more intermediate fiber layers. A third aspect is that fiber failure in the wake of the crack must be prevented which is achieved by using fibers with a high tensile strength and more fibers in each layer. The latter aspect means that a sufficiently large fiber volume content should be obtained. Experiments have shown that the delamination is limited and acceptable for a well balanced fiber volume ratio and sufficient interfaces. First Arall was developed with aramid fibers. Later advanced glass fibers were adopted in Glare [6].

During fatigue tests on Arall specimens it was observed that failure of the crack bridging aramid fibers could occur under certain fatigue load conditions. In the wake of the crack in the metal sheets, the epoxy matrix is also cracked. The crack bridging fibers are loaded in tension during an increasing load. To some extent, the fibres will be pulled out of the cracked area of the matrix material. This is facilitated because the adhesion between the aramid fibers and the matrix is rather week. This pulling out process is not fully reversible during unloading. As a consequence, fiber buckling can occur during many cycles. The fibers will be damaged and fiber failure at $S_{\max}$ is possible. Even then, there are still unbroken fibers closely behind the crack tip which take care of a K reduction and slow crack growth. Fortunately this fiber failure mechanism does not occur in Glare aslo because of a good adherence between the glass fibers and the adhesive matrix. Aspects of the fatigue phenomenon in Arall and Glare were extensively studied by Roebroeks [7]. He showed that fatigue is a fairly complex phenomenon in these hybrid materials, but in a qualitative way the fatigue mechanism is reasonably well understood. By now, Arall is surpassed by Glare. Moreover, significant break-throughs for design and production of Glare structures have been achieved, e.g. splices and CentrAl to be addressed later. But the discussion in this chapter is primarily restricted to Glare.

21.3.2 Fiber-metal laminates as sheet material

Glare sheets are produced with standard bonding technology. The thin aluminium alloy sheets of 2024-T3 or 7475-T76 are pretreated for adhesive bonding. The fiber layers consist of a number of prepregs consisting of unidirectional fibers in a thin adhesive film. The strength of the fibers is 4000 MPa, the stiffness is 88 GPa, and the strain at failure 4.5%. A survey of different Glare grades now used for structural application is presented in Table 21.1 [8]. Prepreg layers can be arranged with fibers in different direction depending on the dominant load on a structural component. If all fibers are in the same direction, e.g. for Glare 1 and Glare 2, the anisotropy

Table 21.1 Standard Glare grades [8].

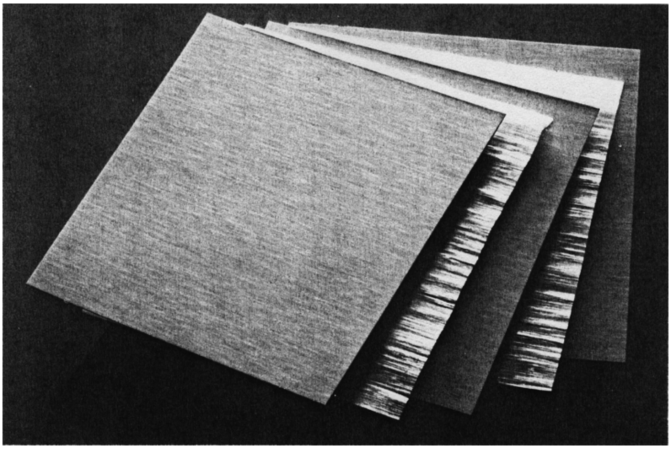
Glare grade	Sub-grade	Metal sheet thickness (mm) and alloy	Prepreg orientation* in each fibre layer**	Main beneficial characteristics
Glare 1	–	0.3–0.4 7475-T761	0/0	fatigue, strength, yield stress
Glare 2	Glare 2A	0.2–0.5 2024-T3	0/0	fatigue, strength
	Glare 2B	0.2–0.5 2024	90/90	fatigue, strength
Glare 3	–	0.2–0.5 2024	0/90	fatigue, impact
Glare 4	Glare 4A	0.2–0.5 2024	0/90/0	fatigue strength in 0° direction
	Glare 4B	0.2–0.5 2024	90/0/90	fatigue strength in 90° direction
Glare 5	–	0.2–0.5 2024	0/90/90/0	impact
Glare 6	Glare 6A	0.2–0.5 2024	+45/–45	shear, off-axis properties
	Glare 6B	0.2–0.5 2024	–45/+45	shear, off-axis properties

*The rolling direction is defined as 0°.
**The number of orientations in this column is equal to the number of prepregs in each fiber layer.

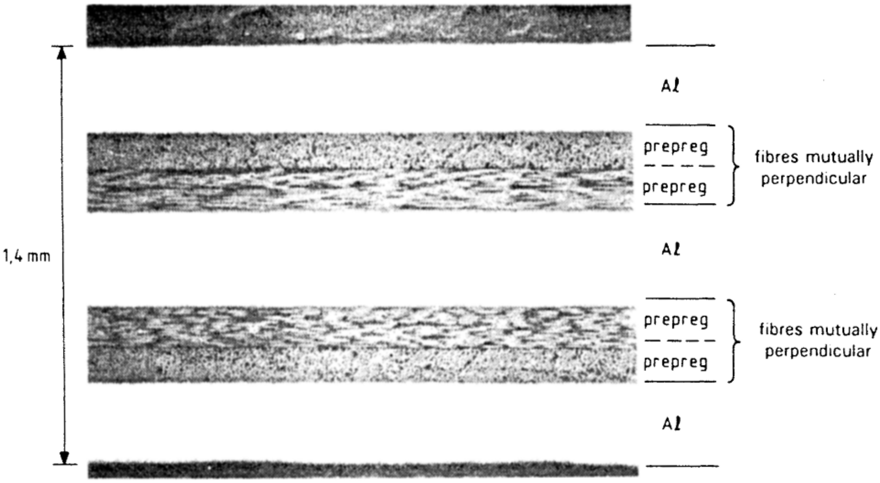
Table 21.2 Mechanical properties and density of Glare 1 and Glare 3. L = longitudinal direction, T = transverse direction.

	Aluminium alloys				Glare			
Sheet material	2024-T3		7475-T761		Glare 1		Glare 3	
Fiber orientation	–		–		uni-directional		cross ply	
Properties	L	T	L	T	L	T	L	T
S_U (MPa)	455	448	520	531	1282	352	717	716
$S_{0.2}$ (MPa)	359	324	476	466	545	333	305	283
Elongation (%)	19	19	11	11	4.2	7.7	4.7	4.7
E (GPa)	72	72	69	69	65	50	58	58
Density (g/cm^3)	2.78		2.78		2.52		2.52	

is significant. The anisotropie is illustrated by a survey of static properties of Glare 1 and Glare 3 in Table 21.2. Properties of the aluminium alloys of the two Glare grades are given in the same table. Cross ply introduced in Glare 3 and Glare 4 can be advantageous for biaxial loading, e.g. for the skin of an aircraft fuselage with the hoop stress as the larger stress component. For all Glare grades the number of layers and the thickness of the aluminium alloy sheets must be chosen in accordance with the loading conditions of a specific component. A coding system is used to refer to a specific composition of Glare. As an example, Glare 4B-4/3-0.4 applies to



(a)



(b)

Fig. 21.7 (a) Three thin metal sheets and two intermediate fiber prepregs before bonding occurs. (b) Cross section of "biaxial" Glare 3 with three 2024-T3 sheets and uni-directional fiber layers in the *L* and *T* directions. (Courtesy G. Roebroeks)

Glare 4B with 4 aluminium alloy sheets and 3 fiber layers (a so-called 4/3 lay up), and thickness of 0.4 mm of the metal sheets. A cross section of Glare 3 is given in Figure 21.7.

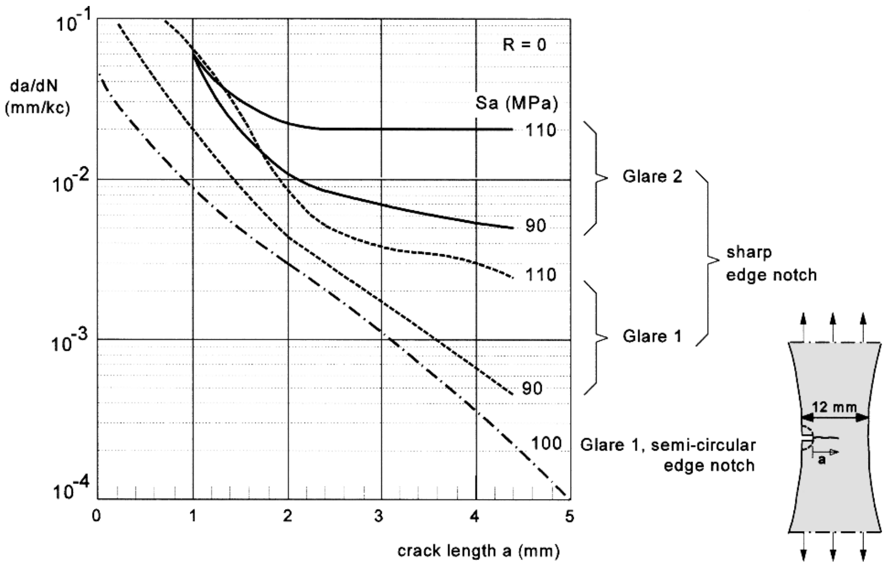


Fig. 21.8 The growth of small cracks in Glare under CA loading [9, 10].

21.3.3 Crack growth in Glare

About crack nucleation in Glare

The growth of fatigue cracks in Glare occurs very slowly as a consequence of the crack bridging effect. However, crack bridging can only occur if a fatigue crack is present. It implies that fatigue crack nucleation might occur earlier in Glare than in solid sheet material because of the slightly lower stiffness of Glare, see Table 21.2. The cyclic strain at the root of a notch will be somewhat larger for the same applied fatigue load. This question was investigated by Marissen for Arall specimens in 1988 [5]. He observed that microcrack growth initially occurred reasonably fast. But already at a crack length equal to the thickness of the aluminium sheets (0.5 mm) it was followed by a significant reduction of the crack growth rate. Later experiments were carried out on Glare specimens by Maria Papakyriacou [9, 10], see the results in Figure 21.8. Specimens with a sharp edge notch (depth 2 mm) and a semi-circular edge notch were tested in a resonance device at a very high frequency (21 kHz). Although very low crack rates occurred, the tests could still be done in a reasonably time. Results for Glare 1 and Glare 2 show a systematically decreasing crack growth rate already for cracks with a length of a few tenth of a mm. Apparently, some favorable

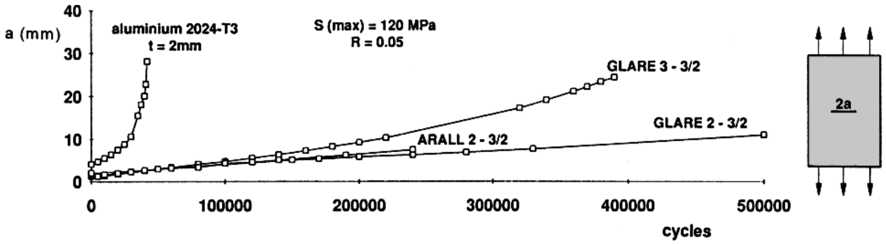


Fig. 21.9 Fatigue crack growth in fiber-metal laminates under CA loading [7].

effect of the fibers is observed as soon as some minute cracking has been initiated. How fast and how far $\mathrm{d}a/\mathrm{d}N$ will decrease will depend on the shape of the specimen and the notch geometry.

Crack growth of macrocracks in Glare

Crack growth results for constant-amplitude (CA) loading are shown in Figure 21.9 with a crack growth curve for 2024-T3 for comparison. The much slower crack growth in Glare as compared to crack growth in solid sheet material is evident. At a crack length $a = 10$ mm (0.4 inch) the crack growth rate in Glare as compared to the growth rate in 2024-T3 is seven times slower in Glare 3 and 20 times slower for Glare 2. Slow crack growth in Glare has been confirmed in several test programs, see e.g. [6, 7].

Fatigue crack growth in Glare has also been studied under VA loading. Simple CA tests with OLs have shown the crack growth retardation in a similar way as discussed in Section 11.2. The delay periods were smaller which should be associated with smaller plastic zones created by the OLs due to restrained crack opening because of crack bridging.

Crack growth in Arall and Glare specimens with 2024-T3 sheet material was also investigated under flight-simulation loading with the miniTwist load history [11]. Specimens with open holes were adopted in order to be informed about the crack initiation period as well. Illustrative results are shown in Figure 21.10. The initial crack growth period until a crack length of 1 to 2 mm (0.04 to 0.08 inch) is in the order of 10000 to 15000 flights. But the remaining crack growth life is very large. Actually, tests were stopped after 270000 flights for three of the four types of fiber-metal laminates because of marginal crack growth. The same tests were also carried out on similarly laminated specimens without fibers [12]. The crack initiation fatigue lives were of the same order of magnitude as in the tests on the fiber-metal

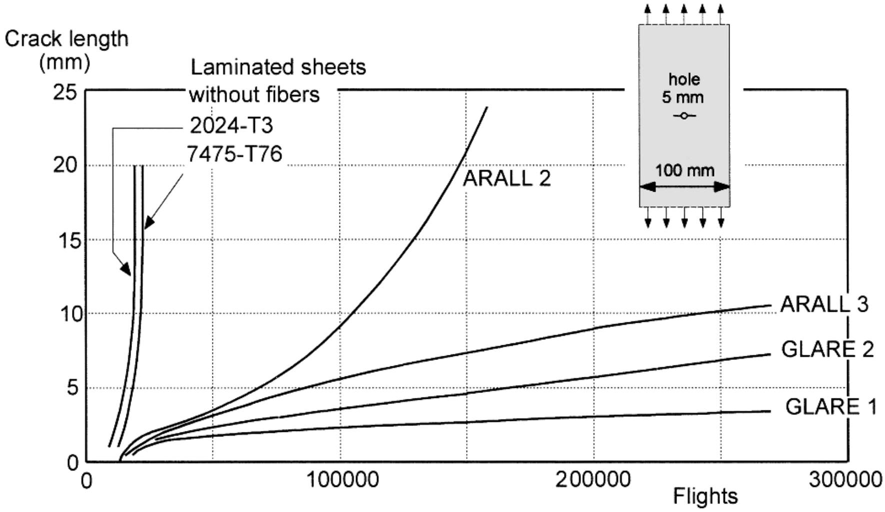


Fig. 21.10 Fatigue crack growth in open-hole specimens of fiber-metal laminates under flight-simulation loading (miniTWIST) [11].

laminated specimens, but the crack growth lives are highly different. The crack growth life until failure in the laminated specimens without fibers is in the order of 10000 kc. However, after 270000 flights the crack length was no more than approximately 3 and 7 mm in the Glare 1 and Glare 2 specimen respectively.

The effect of the truncation level of the gust spectrum was also considered in [11]. The gust spectrum of miniTwist is a steep spectrum, and the truncation level of the larger amplitude cycle has a large effect on fatigue crack growth in 2024-T3 specimen. as discussed in Chapter 11 (see the discussion on Figure 11.16). It turned out that a similar large effect was found for Glare 1 and Glare 2, i.e. slower crack growth for a higher truncation level. It appears that the crack growth behavior in the aluminium sheets of the fiber-metal laminates is similar to the behavior of solid sheet specimens. This observation is important for considering prediction models for fatigue crack growth in fiber-metal laminates. A major problem then is to predict the history of the stress intensity factor in the metal layers of fiber-metal laminates. A more elementary problem is to predict the *K* history in fiber-metal laminates under CA loading. This is the problem which was addressed by Marissen for Arall [5]. He made several simplifying assumptions to define the first prediction model for fatigue crack growth in Arall sheet specimens. A major problem was to predict the delamination zone around the growing fatigue crack in the aluminium sheets, and in

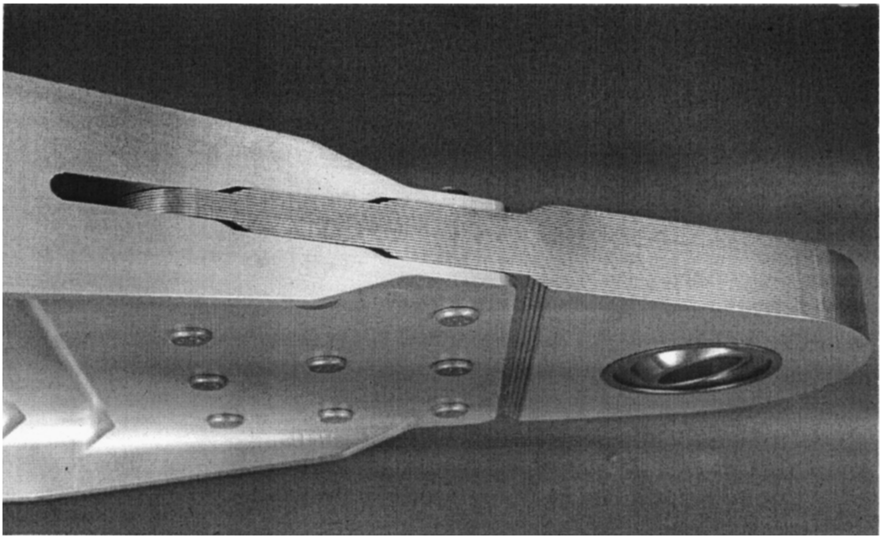


Fig. 21.11 A lug of an aircraft wing-fuselage attachment made of Glare 1 [16]. Lug thickness 19.0 mm (0.75 inch), hole diameter 44.45 mm (1.75 inch).

addition the crack bridging forces which are required for predictions of the K-value at the tip of the crack. Variants on the model of Marissen has been published in the literature. A more realistic prediction model was proposed by Alderliesten in 2005 [13, 14]. It includes the interaction of the increasing delamination zone and fatigue crack growth in the aluminium sheets. In addition to the crack growth resistance of the sheet material, the resistance to the simultaneous delamination is accounted for in order to predict the shape of the delamination zone. Prediction for VA loading are now investigated [15].

21.3.4 Fatigue properties of Glare components

Experience of three cases will be summarized:

- *Wing-fuselage connection by lugs*
  The connection between the wing and the fuselage of the CN-235 aircraft occurs with four lugs, two at the front spar and two at the rear spar. In view of damage tolerance it is unacceptable that a lug should fail by fatigue. A low stress level was adopted in view of the risk of a fatigue failure and the low fatigue limit of lugs due to fretting

corrosion (Chapter 15). An investigation was made to produce the lug as a Glare 1 component with all dimensions similar to a lug of the CN-235 aircraft, which implied a thickness of 19 mm (0.75 inch) (and a hole diameter of 44.45 mm (17.5 inch) [16]. The lug is shown in Figure 21.11. It was built up with 25 metal layers [16]. The lug was tested in a flight-simulation test with a load history consisting of blocks of 1000 flights with 10 different types of flights in a random sequence. No indications of cracks were obtained after simulating 120000 flights at the original design stress level for the lug connection. All stress levels were then increased by a factor of 1.25 and again 120000 flights were simulated without any cracking. This increasing of the load level was done three times, bringing the load level to 1.253 = 1.95 times the original design load level, almost doubling the stress level. Failure of the lug still did not occur, but fracture occurred in the aluminium alloy clamping of the lug after another 92000 flights at the last load level. Small cracks could be observed inside the hole in three of the 25 metal layers which should have an insignificant effect on the static strength of the joint. The test shows a large safety margin and favorable information for long inspection periods.

- *Riveted lap joint*
  In view of the application of fiber-metal laminates as a skin material for a pressurized aircraft fuselage, various fatigue tests were carried out on a riveted lap joints. Some results are shown in Figure 21.12 for lap joints with three rivet rows and seven rivets in each row [17]. Specimens were made from Glare 3 (cross ply) and monolithic 2024-T3. The Glare sheets consisted of three aluminium alloy layers with intermediate prepreg layers. The specimens were tested until multiples of 100000 cycles and then pulled to failure to determine the residual strength and to observe the amount of fatigue cracking on the fracture surface. The residual strength is plotted in Figure 21.12. Some small cracks were present in a 2024-T3 specimen after 100000 cycles. Larger cracks were found in other specimens after longer fatigue cycling periods. Finally, a specimen failed at a fatigue life of 470000 cycles and the residual strength is then equal to S_{max} of the fatigue load. Fatigue cracks in the monolithic 2024-T3 sheets can penetrate through the full thickness of the sheet. The increasing fatigue damage as a function of the number of load cycles is shown in Figure 21.12. It is typical "multiple site damage". However, cracking in the Glare 3 specimens occurred for a large number of cycles in one of the three metal layers only, which is the mating surface layer of the lap joint. This layer carries the maximum

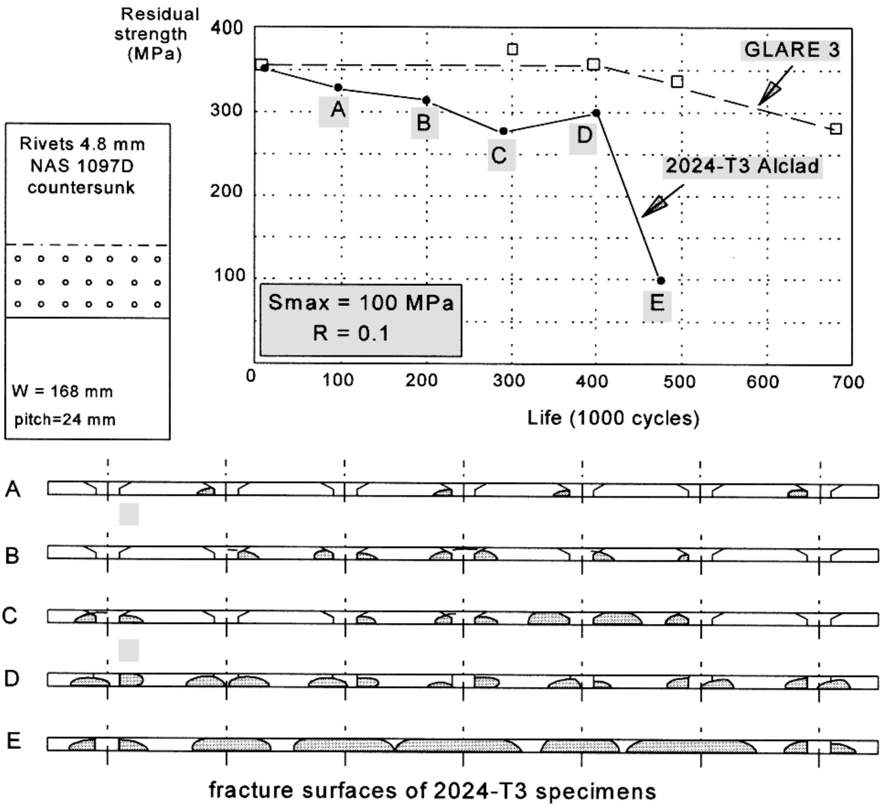


Fig. 21.12 The residual strength of riveted lap joints after previous fatigue cycling [17].

bending stress. However, after 500000 cycles fatigue cracks occurred in the mating surface layer only without any cracks in the other two layers. Only for the last Glare 3 result in Figure 21.12 after fatigue testing to 700000 cycles, cracks were found in the second sheet layer. Apparently, a superior damage tolerance behavior if compared to the full metal lap joint.

- *Crack growth in the Airbus barrel test*
  Results of fatigue crack growth in a full-scale test representative for the Airbus A-340 fuselage are shown in Figure 21.13 [18]. Different skin materials were used in the fuselage structure and two different cases were investigated: (i) crack growth under a broken stringer and (ii) crack growth under a broken frame. The cracks started from a saw cut of 75 mm (3 inch). The results show a considerably slower crack growth in Glare 3 (thickness 1.4 mm) than in 2024-T3 (thickness 1.6 mm). This illustrates the potential usefulness of Glare 3, also because of weight

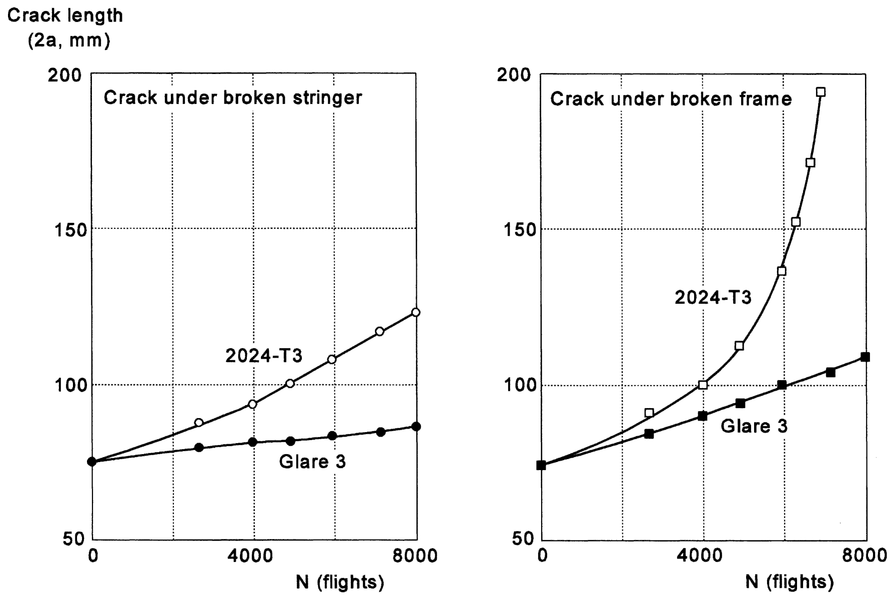


Fig. 21.13 Crack growth results obtained in the Airbus barrel test [18].

savings in view of the lower thickness and in addition the lower specific mass (2.52 g/cm^3 for Glare 3 and 2.78 g/cm^3 for 2024-T3).

21.4 More about Glare

In general, Glare should be considered as a potential material to be selected for structural elements for which fatigue is a critical design issue. In Section 21.3, the fatigue properties of Glare have been highlighted. However, the introduction and structural application of a new material requires more information than data about the static and fatigue properties only. Various aspects about durability, physical properties, and technological aspects must be explored. A few remarkable aspects of Glare associated with the laminated composition will be summarized in Section 21.4.1. Furthermore, the production techniques for sheet metal products are not completely similar to the standard techniques for aluminium alloy sheet material. Aspects of this topic are briefly covered in Section 21.4.2.

21.4.1 Some typical properties of Glare

The corrosion resistance of the fiber-metal laminates is good, partly because the thin sheets are anodized and coated with a corrosion-inhibiting primer prior to the bonding process. A specific feature is that the depth of corrosion damage, also in severe corrosive environments, is restricted to the outer metal layer. The first fiber-epoxy layer has been shown to be an efficient barrier for a deeper penetration of corrosion damage. The durability of Glare as applied in the fuselage of the Airbus A380 is extensively investigated as reported by Beumler [19]. Various specimens including joints have been exposed in extreme outdours environments, e.g. in Australia, which after several years are tested in the laboratory. Until now (2008) specimens exposed for six years have given satisfactory results.

A similar barrier function also occurs in case of a fire. The extraordinary flame resistance of Glare follows from the ability of the glass fiber epoxy layers to prevent fire penetration. Experiments have shown that Glare can be an attractive material for fire walls.

Another interesting quality of Glare laminates is associated with the impact damage resistance. Both low and high-velocity impact damage of Glare parts do not have significant consequences with respect to durability and aircraft inspection intervals if compared to damage of aircraft structures of aluminium sheet material. To some extent the impact resistance of Glare is comparable to the resistance of the aluminium alloys used in Glare. However, for an aircraft under a severe hailstorm Glare can even perform better. In sheet metal aircraft structures crack can be induced by a hailstorm. However, damage is less severe for Glare skins due to the fact that the fiber layers take care of maintaining the coherence of skin panels.

21.4.2 Production and design aspects of Glare structures

Workshop properties

The production techniques applied to aluminium alloy sheet material are largely adopted in a similar way for Glare. It includes techniques as cutting, contour milling, drilling, sheet metal bending, riveting and bonding. But there are differences which should be recognized and explored in trial experiments. Drilling holes and contour milling implies that new sheet edges

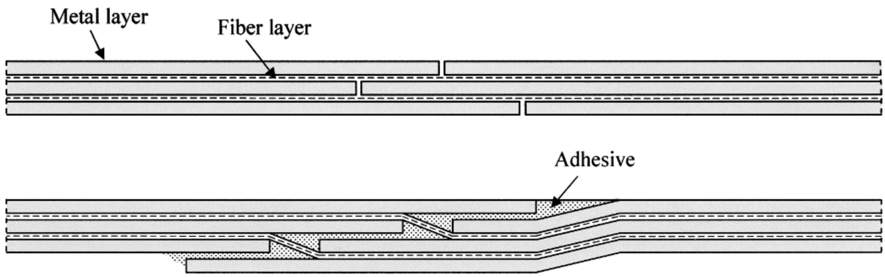


Fig. 21.14 Splicing concepts for obtaining large sheets [8].

are made which may cause delamination damage along the edges. Care must be taken.

Sheet metal bending of uni-directional fiber-metal laminates perpendicular to the fiber direction is possible and adopted for the production of stiffeners of various shapes. Again the occurrence of delamination should be explored.

Curved panels

In view of the application of Glare to fuselage structure, the question of producing single curved and double curved Glare sheet panels is also considered. Both types of panels have been made in the autoclave by using a single or double curved mould [8]. An attractive feature is that it occurs as part of the Glare production operation, in other words, in a single production cycle. The thorough stretching operation for double curved panels of solid metal sheets is thus avoided.

Large sheets

Initially the size of Glare sheets was limited to a width of 1.65 m (65 inch) because this was the maximum width to be obtained in the rolling production of the thin metal sheets. Roebroeks [8] developed a splicing concept to produce large Glare sheets by a bonding operation implemented in the production of Glare. The basic idea is illustrated in Figure 21.14a for a 3/2 lay-up. The interruptions of the three metal layers are located at a staggered position. The interruptions imply a local reduction of the static strength. If this is not acceptable, the overlap splice configuration shown in Figure 21.14b can be adopted. Large panels obtained with splices are

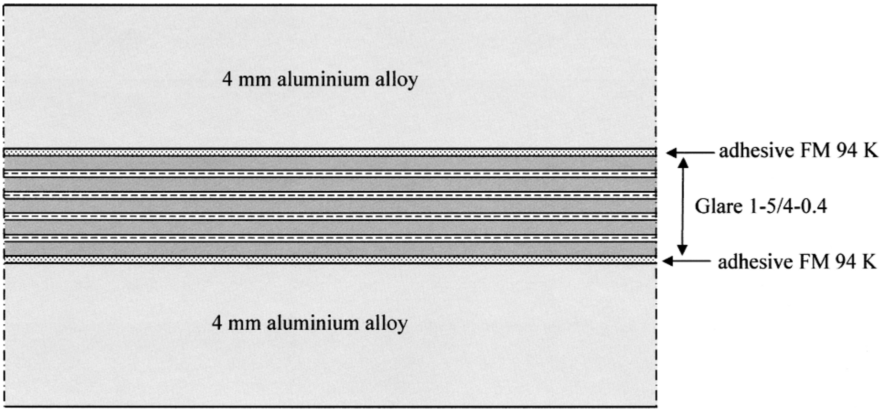


Fig. 21.15 Typical lay-up of CentrAl [2].

now applied in the fuselage of the Airbus A380. In addition to good fatigue properties another advantage is that many riveted joints can be eliminate.

Thick hybrid plates

Glare sheets were primarily developed as sheet material for pressurized transport aircraft. However, the tension skin of a wing structure may also be fatigue critical. The thickness of these skins can be fairly large. This has led to the development of CentrAl as described by Roebroeks [2]. CentrAl combines Glare and solid sheets in a single plate. An illustrative example is shown in Figure 21.15. Glare in the center of the thickness is bonded to the relatively thick outside metal sheets by a special adhesive without fibers in order to reduce possible delamination between Glare and the outside metal layers. CentrAl is a kind of a superposition of a crack growth retarding material and another solid material. Various mixed lay-ups of Glare sheets combined with solid metal sheets are possible. The designer should specify which combination is most useful for his structure. Actually, he should design the constitution of an optimal variant of Central. Roebroeks carried out fatigue tests on several types of CentrAl specimens which showed substantial improvements of the crack growth life. The crack growth retardation may be less than the impressive retardation shown for pure Glare in Figure 21.9, but increasing the crack growth life with a factor of 3 to 10 can already be most attractive for a damage-tolerant structure.

21.5 Some summarizing remarks

The development of new materials cannot be completed without research of a large variety of material properties which are significant for practical application. Much is learned by experiments and service experience of initial application in structures. With respect to the basic understanding of the fiber-metal laminates Arall and Glare, a vast amount of development research was carried out under the stimulating leadership of Professor Vogelesang in the Structures and Materials Laboratory of the Faculty of Aerospace Engineering of the Delft University of Technology. Many students were involved, including PhD students of several countries (USA, Germany, Belgium, China, and the Netherlands). The cooperation with Fokker Aircraft Industries, Airbus, and the National Aerospace Laboratory NLR should also be mentioned.

The original motivation for the development of the fiber-metal laminates was to obtain a material with a high fatigue resistance. Fatigue properties are discussed in the previous section. It is logical that fiber-metal laminates should be considered in particular for fatigue critical components. This can be a small component, e.g. the lug shown in Figure 21.11. For such a small part, the lower specific mass of Glare may not be significant, but the durability, limited inspection effort, and high safety are important improvements. For large parts, e.g. the aircraft fuselage skin, weight saving can certainly be of interest. Good fatigue properties of a fatigue critical component may imply that the design stress level can be increased. The weight saving is then coming from the smaller material volume, and in addition from the lower specific mass of the fiber-metal laminates.

It is remarkable that the fiber-metal laminates also have some favorable properties not related to fatigue but still associated with the laminated stacking of thin layers. Corrosion resistance, fire resistance and low-impact damage sensitivity are mentioned in Section 21.4.1. Several aircraft components made of fiber-metal laminates (Arall and Glare) have been installed in operational aircraft. The fiber-metal laminates were selected for reasons of fatigue, damage tolerance, weight saving and impact resistance. The behavior in service of these components is fully satisfactory. This is also true for components originally made of aluminium alloy material which showed early fatigue problems in service. Replacing the components by components of fiber-metal laminates eliminated the fatigue problems. The most appealing application of Glare is associated with the Airbus A380. Large panels on the fuselage are made from Glare. As said before, designing a structure from Glare requires various experiments to explore the behavior in a specific structural component for load histories applicable to the component. This is extensively reported in the doctor thesis of Beumler [19]. His work includes fatigue crack initiation and propagation in riveted joints under relevant load histories, variables of riveting, effects of temperature and load frequency, and environmental effects. The outdoor exposure program was already mentioned in Section 21.4.1. It may well be emphasized that extensive and diverse investigations are necessary for the development of a new material until it becomes a mature and potentially useful material for structural application. Physical and mechanical understanding of the material is essential. Production techniques are another important topic to be explored. The fiber-metal laminates offer another dimension to designing against fatigue, especially for fatigue critical components in aircraft, but possibly also for other structures if damage tolerance and durability are important issues.

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