FIBRE METAL LAMINATES

The Development of a New Family of Hybrid Materials

19th Plantema Memorial Lecture

L.B. Vogelesang

Delft University of Technology, Faculty of Aerospace Engineering

A brief history about the development of the fibre-metal laminates is presented, starting with laminates without fibres, followed by laminates with aramid fibres (ARALL) and finally laminates with glass fibres (GLARE). The crack bridging effect and the need for thin metal sheets is emphasised to arrive at optimal fatigue properties. A full coverage of various properties is required to introduce a new material for designing aircraft structures. This is illustrated by results of several investigations which have indicated possibilities for structural weight saving. In addition, GLARE has excellent impact behaviour and a high resistance against corrosion and flammability. Applications in aircraft structures are summarised. With respect to technological properties, a new splicing concept for very large skin panels for fuselages has been developed which is also economically attractive and will be applied in the Airbus A380. It appears that the GLARE family of fibre-metal laminates is ready for various applications.

INTRODUCTION

The origins of the development of fibre-metal laminates (FML's) can be traced back to the bonded plywood wing structure introduced by Anthony Fokker in 1916. By bonding various layers of plywood, he could position the fibre orientation of the wood in the optimal directions for which strength was required. This avoided the problem that a sheet made from one single piece of wood would have its fibres running in only one direction (Fig. 1).

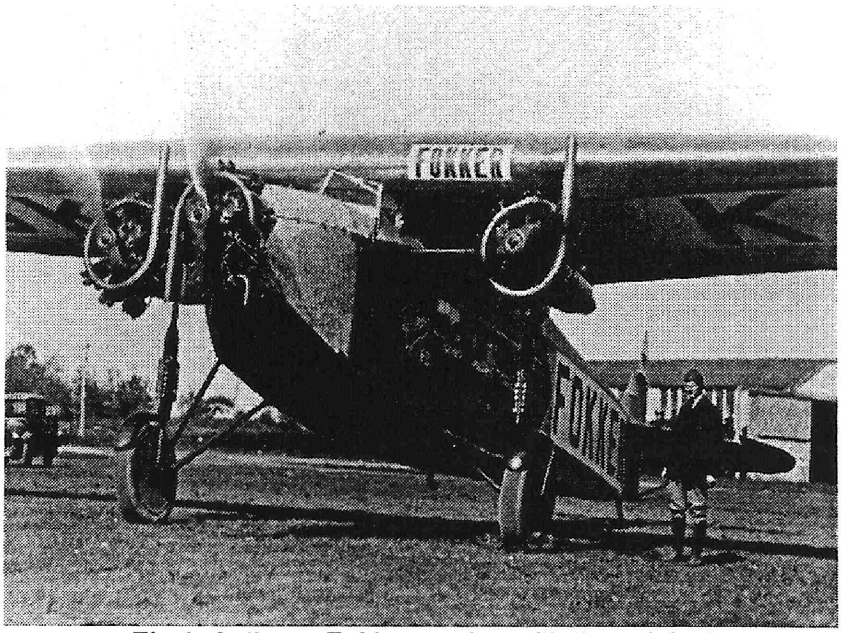


Fig.1: Anthony Fokker posing with the original Fokker F.VII-3m in 1925

Also the British company De Havilland had extensive experience with bonding. In the years around the Second World War, the De Havilland aircraft company was something of an exception in aircraft construction, being one of the last manufacturers to build aircraft predominantly made from wood. During the war, De Havilland built the famous Mosquito (Fig.2).

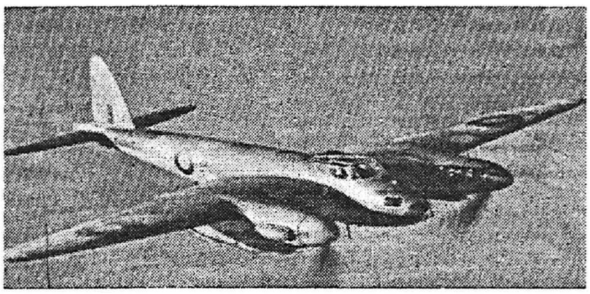


Fig. 2: The Havilland Mosquito

The wing and fuselage were built by laying up thin layers of wood with adhesive in a curved mould, in a manner similar as used by Fokker for the wing of the successful aircraft of the 1920s. As will be discussed later, this method also resembles the way in which curved GLARE fuselage panels are produced nowadays for the Airbus A380. To achieve a thin wing for the Mosquito aircraft De Havilland had chosen a unique hybrid structure for the wing spar. The spar was built up from a plywood shear web with redux bonded to aluminium alloy flanges. Because of its alternative construction practices De Havilland became the first company that also bonded metal parts together.

During the early fifties, the Fokker Aircraft Industries was studying the possibilities of metal bonding for the new Fokker Friendship aircraft, the F-27. At that time, the fatigue problem was a nightmare because fatigue failures of transport aircraft had occurred in service (Martin 202, De Havilland Comet). Considerable research on flight-by-flight experiments has been carried out since then. These tests demonstrated superior fatigue resistance for built-up bonded structures compared to integral-machined parts (Fig.3). Another important finding was the favourable resistance to fatigue crack growth of laminated sheet metal reinforcements around a large wing joint (Fig.4). The growth was relatively slow because crack nucleation started in a single layer. The other intact layers effectively bridged the crack and as a result considerably slowed down further crack growth. This was an important finding for later research on laminated sheet material and FML's [1], [2].

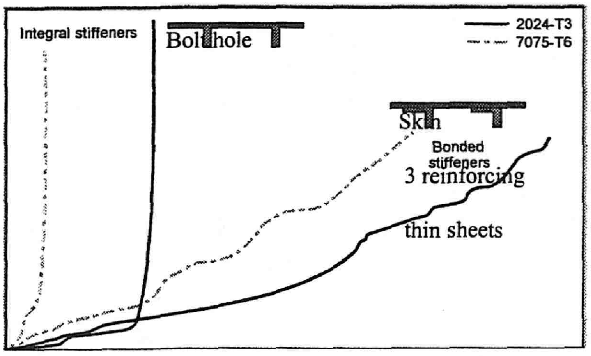


Fig. 3: Comparison between crack growth of integral axis: Crack length along horizontal axis: Cycles

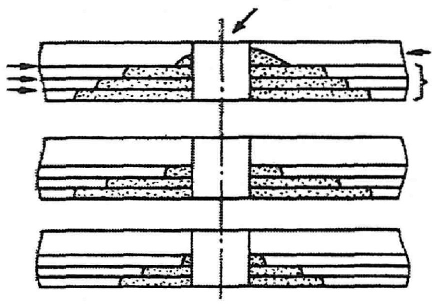


Fig. 4: Fatigue crack extension in reinforced machined and bonded wing structure. Along vertical wing joint

The aim of this Plantema Memorial Lecture is to survey the basic key points of developing FML's to become efficient and attractive materials for application in aircraft structures. The superior fatigue and damage tolerance properties were evident right from the beginning, but a large range of topics had to be explored before the FML's could be considered for new aircraft structures, see Table 1. Experience varying from laboratory investigations to practical applications of FML's in aircraft structures is summarised in this lecture. Illustrative examples will be discussed. A full account cannot be given in a single lecture but detailed information can be obtained from the Fibre-Metal Laminates Centre of Competence (FMLC) in Delft.

Mechanical properties
* $S_{0.2}$, S_U, E, elongation
* Blunt notch strength, residual strength
* Fatigue properties of notched elements, joints
* Fatigue crack growth
Durability
* Different types of corrosion, erosion
* Material degradation by environmental aspects
* Impact damage sensitivity
Physical aspects
* Specific mass
* Thermal expansion, heat conductivity
* Lightning strike
* Flammability, toxicity
Technological properties
* Machinability
* Cold-forming processes
* Jointing processes
* Repairability
Cost-effectivity

Table 1: Characteristic aspects of materials for application in aircraft structures

THE FIRST STEPS TO FML'S

During the life of an aircraft, fatigue, corrosion and incidental (impact) damage can damage the structure. These types of damage have to be considered during the design process for reasons of safety. It is also important for economical reasons, because the damage has to be detected and repaired during maintenance. Vlot [3] classified the repairs of 71 fuselage of the Boeing-747 aircraft operated by 17 airlines. The repairs were associated with three types of damage: fatigue, corrosion and impact. Only the primary structure was considered. The average life of the aircraft was 29,500 flying hours. The aim was to compare the importance of the three types of damage. The distribution of the 688 repairs was:

Fatigue cracks	369 repairs (58 %)
Corrosion	202 repairs (29 %)
Impact damage	90 repairs (13 %)

For an airline maintenance covers almost 20% of the direct operating costs (D.O.C). Therefore in the early eighties there was a strong call for the so-called "no-repair" structure. In view of this problem FML's are promising materials.

In 1974 Fokker started work on fracture toughness of bonded sheet laminated material. The fracture toughness of the laminated material was some 25% larger than for the monolithic material. Fatigue crack growth in laminated sheet material was studied by Schijve et al.[4] in the 1970's. The crack growth rate of through the thickness cracks was systematically lower than for monolithic material of the same thickness, also for through cracks in lugs. A most significant reduction of the crack growth rate was obtained for part through cracks. Crack growth starting from a single crack in the outer layer of the laminated sheet material was considerably delayed. Because this crack does not immediately penetrate into the sub-surface layers, opening of the fatigue crack in the surface layer is effectively restraint. This reduces the stress intensity at the crack tip in the outer layer. Actually, the inner layers are bridging the fatigue crack of the outer layer. This phenomenon of crack bridging also plays a key role in the static and dynamic behaviour of fibre-metal laminates.

Fokker started experiments on reinforcing the adhesive layers in laminated sheet material by uni-directional fibres. Improvements of the crack growth results were obtained in some cases but analysis of the results indicated that this approach was questionable. In Delft the group of Schijve and Vogelesang continued research on laminated sheet material, and they came to the conclusion that reinforcing adhesive with fibres was a wrong approach, in spite of some improvements being found. The real break-through came with the Delft philosophy: not simply reinforce the adhesive layers, but develop an effective crack bridging function of the fibres. It then became clear that intensifying crack bridging requires sheet metal layers with a significantly lower thickness than the 1 mm thickness used until that time. Sheets with a thickness of 0.3 mm were introduced. A new hybrid material was obtained by an optimal combination of thin metal sheets with high strength fibre composite layers: a marriage between two different materials with different properties resulting in a FML combining the best properties of both constituents to combat fatigue, corrosion and impact. Originally, aramide fibres were used, and so ARALL was born, see Figs.5 and 6.

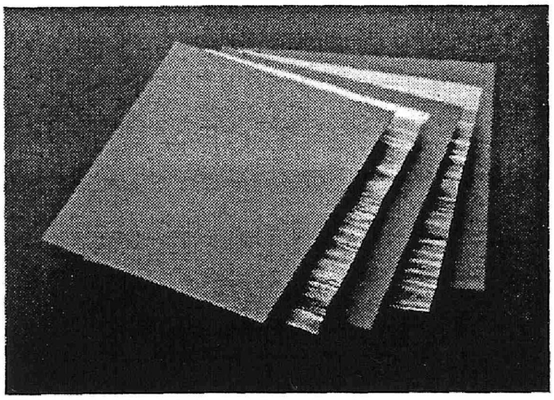


Fig. 5: Typical fibre-metal laminate with 3/2 lay-up structure Al-alloy sheet

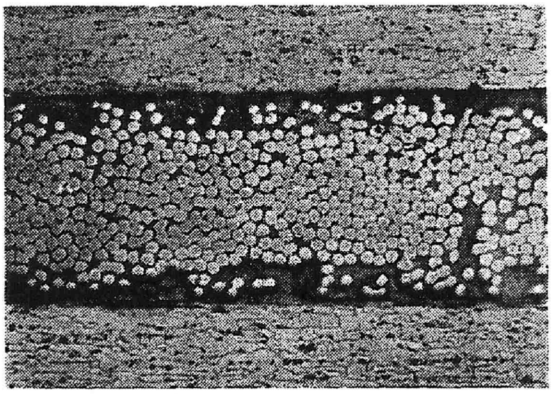


Fig. 6: Cross section of ARALL showing unidirectional Fibres and grain

THE FIBRE-METAL LAMINATES DEVELOPMENT

The first generation of fiber-metal laminates based on aramid fibres, ARALL, was produced by ALCOA. This aluminium industry was the first one which could deliver sheets of AL 2024-T3 with a thickness as low as 0.3 mm in close tolerances. Unidirectional fibre layers were used for the first ARALL sheets because they were developed primarily for application in the tension skin of the aircraft wing, a fatigue critical component. In view of the bonding cycle to produce ARALL, a full ultrasonic scanning was applied to all sheets, which has remained a standard procedure for all fibre-metal laminates developed later. After numerous tests to cover a large variety of properties (see Table 1) ARALL became a mature material which was commercially available. Its promising characteristics were demonstrated by testing a full scale Fokker 50 wing panel in the mid 1980's (see fig. 7). A weight saving of 20 % was possible. The weight saving argument also applied to the application of ARALL in the C17 cargo door.

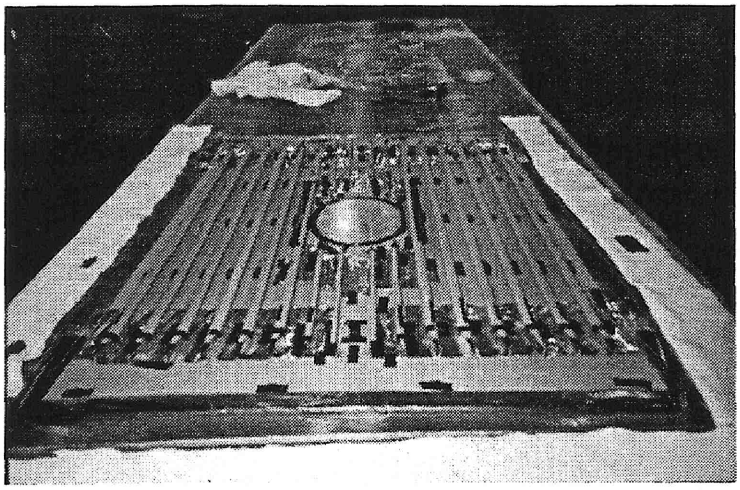


Fig.7: ARALL F-27wing panel on the autoclave table

In 1987 the second generation of fiber-metal laminates was introduced with the name GLARE. This FML was based on high strength glass fibres [5]. Where ARALL had unidirectional fibre layers because it was optimised for wing structures, GLARE was developed in both unidirectional variants (GLARE 1 based on Al 7475 and GLARE 2 on Al 2024) and biaxial variants (GLARE 3 based on 2024 with an equal percentage of fibres in 0 and 90 direction, and GLARE 4 with twice the percentage in 0 direction). The biaxial cross-ply variants were required for the application as a fuselage skin material in view of biaxial stress fields in the pressurised structure. Moreover, glass fibres were introduced because fibre failure was observed in ARALL under cyclic loading at a zero stress ratio (R=0) which is relevant to the fuselage skin. Compressive stresses are present in the fibres after curing of the laminate. Cyclic compression of the crack bridging fibres will occur even if the minimum applied stress on the laminate is zero. For aramid fibres this leads to failure of the crack bridging fibres. Deutsche Airbus extensively demonstrated the technology of GLARE in 1988/89 in Hamburg by testing an A330/340 fuselage barrel (Fig.8). In the fuselage section crack growth was measured in panels of different aluminium alloys and GLARE. The crack growth rate was significantly slower in GLARE in comparison to crack growth in the aluminium alloy panels.

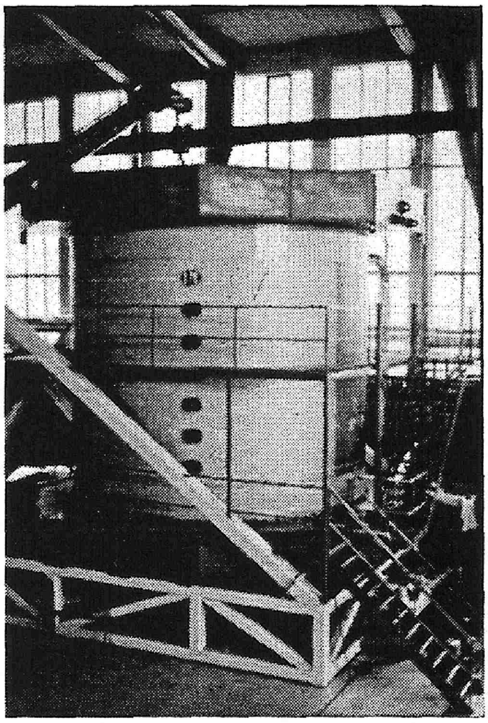


Fig. 8: Full-scale barrel test in Hamburg

During investigations on the various properties of GLARE listed in Table 1 the superior fatigues properties of GLARE components was confirmed. However, it turned out that some other characteristic properties were also significantly better than these properties for monolithic aluminium alloys. This applies to impact damage, residual and blunt notch strength, flame resistance and corrosion properties as will be discussed later.

The first civil applications of GLARE were mainly associated with a better impact damage resistance, which applies to GLARE in the bulk cargo floor of the Airbus A330 and the front bulkhead of the Bombardier Learjet 125.

The so-called splicing concept to be discussed later was developed in 1993. It implies that the production of very large panels without joints is possible. The combination of favourable properties makes GLARE an attractive material to be selected for transport aircraft. Recently the Airbus Consortium has selected GLARE as the material for large parts of the skin of the fuselage of the Airbus 380 (Fig. 9).

Various properties of GLARE will now be illustrated and discussed in more detail. As said earlier, a complete picture of the state of the art cannot be given. In this lecture the discussion is more focussed on the application of GLARE as the skin material for pressurised fuselages. Problems of this application are associated with damage tolerance, safety, production, economy and aircraft utilisation in service.

STATIC PROPERTIES

FML's are produced with standard bonding technology. After the hot curing cycle in the autoclave (120°C), the FML's carry a residual stress system over the thickness of the material, with a small tensile stress in the aluminium sheet and compression in the fibres. Static properties are presented in Table 2.

	2024-T3	GLARE 2	GLARE 3	GLARE 4
Tensile ult. L	440	910	625	795
Tensile ult. LT	435	300	610	525
Tensile yield L	325	325	285	290
Tensile yield LT	290	215	260	230
Compr. yield L	270	305	260	
Compr. yield LT	310	230	270	
Bearing ult.	890	680	760	630
Blunt notch L*	410	600	452	510
Blunt notch LT*	400	235	415	360
E-modulus L	72400	66500	59500	59000
G-modulus L	27600	20500	20500	18700

**Static strength on net section of sheet specimen with a central hole, diameter/specimen width 25/100 mm.*

Table 2: Properties of some GLARE 4/3 – 0.4 lay-up (MPa)

The data in Table 2 illustrate several remarkable characteristics of the FML's:

- High static strength in the fibre direction, associated with the high strength of the fibres,
- Anisotropic behaviour, exept for GLARE 3 with a 50/50 cross ply,
- Somewhat lower stiffness than for the Al-alloys,
- Specific weight approximately 10% lower than for aluminium.

Fatigue behaviour

Fig. 9: Artist's impression of Airbus A380

The fatigue behaviour of a structure is characterised by crack nucleation followed by crack growth. Comparative observations on crack nucleation period for 2024-T3 specimens and similar GLARE specimens suggest small differences. Apparently, the fibres do not have a substantial influence on the initiation period. However, as soon as the crack is growing as a macro-crack, the growth rate is much smaller in GLARE specimens than in 2024-T3 specimens. The fatigue properties of FML's have been evaluated in numerous test programs at the Delft University and throughout the aircraft industry, including tests on notched specimens, joints and components. Figure 10 compares the fatigue performance of two GLARE variants (GLARE 3 and GLARE 4) and monolithic 2024-T3 under simulated fuselage loading from a central saw cut. While the crack growth rate of the monolithic material increases rapidly with increasing crack length, the laminate materials exhibit their characteristic, almost constant slow crack growth behaviour. Under realistic loading conditions, FML's exhibit crack

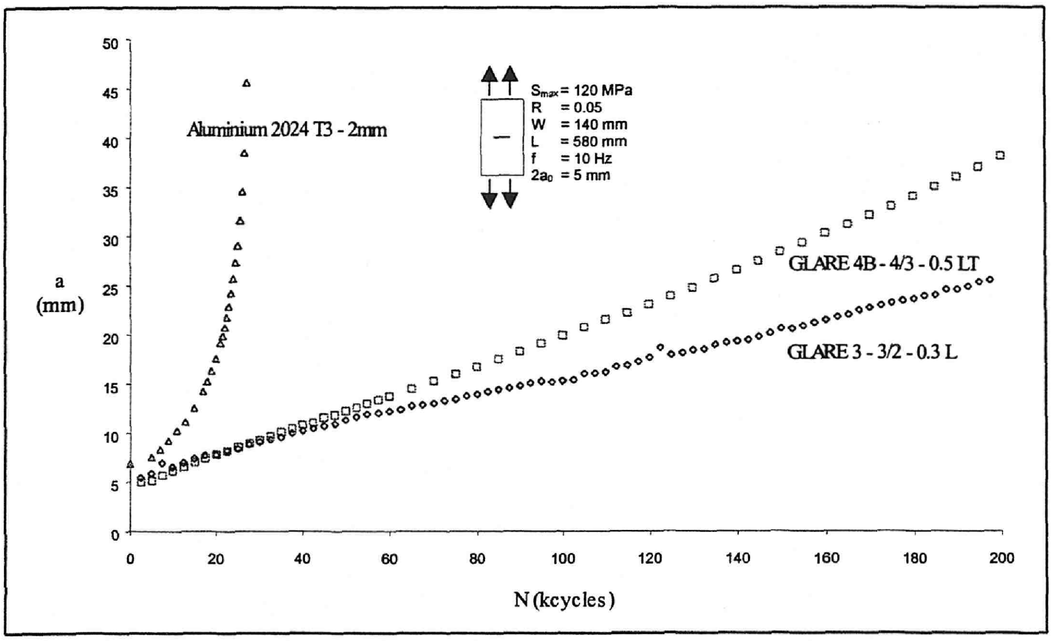


Figure 10: Crack growth curve of aluminium 2024-T3 and Glare 3-3/2-0.3L and Glare 4B 4/3-0.5LT for constant amplitude fatigue loading.

growth rates 10 to 100 times slower than in aircraft aluminium alloys. As a consequence, inspection of the structure for fatigue is not really necessary during the operational life of the aircraft (no repair structure).

The key to this favourable behaviour is the so-called fibre crack bridging mechanism (see Figure 11). The intact fibres impose a significant restraint on crack opening. Furthermore, the fibres in the cracked area transmit part of the load through the cracked area. As a result, there is a large reduction in the stress intensity factor K at the crack tip.

The load transfer from the metal sheets to the crack bridging fibres causes shear stresses on the interfaces between the fibre layers and the aluminium alloy sheets. These local shear stresses cause delamination between the metal layers and the fibre layers along the crack edges [6]. The shear stress is small because of the low thickness of the metal layers. This is another essential feature of the new fibre-metal laminate concept. The delamination will avoid fibre failure and for a well-balanced fibre-metal ratio, the delaminations remain acceptably small during the life of the

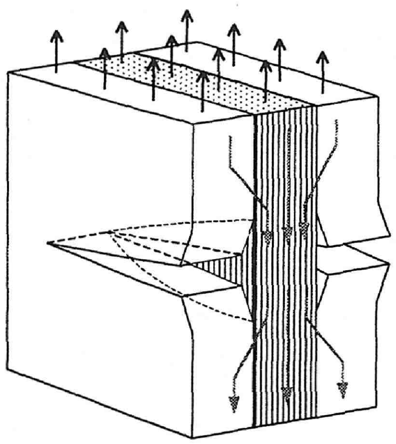


Fig.11: Crack bridging by unbroken fibres in the wake of the crack restraining crack opening. Part of the load is transmitted by the fibers through the cracked area. Some delamination occurs between the fibre layers and the aluminium-alloy layers along the crack edges.

aircraft. The result is a very slow crack growth at a rate almost independent of the crack length.

RESIDUAL STRENGTH

The longitudinal and circumferential joints in a fuselage structure are the relevant locations where fatigue damage will most probably occur. The residual strength of Al 2024 and GLARE lap joints is shown in Figure 12. The graph shows the reduction of the residual static strength after large numbers of fatigue cycles at a stress level representative for the hoop stress in a pressurised fuselage. Multiple site damage (MSD) occurred in the lap joints of both materials. However, the fatigue cracks in the 2024-T3 lap joint penetrated through the full sheet thickness whereas in the GLARE lap joint the cracks were confined to a single metal layer. As a result,

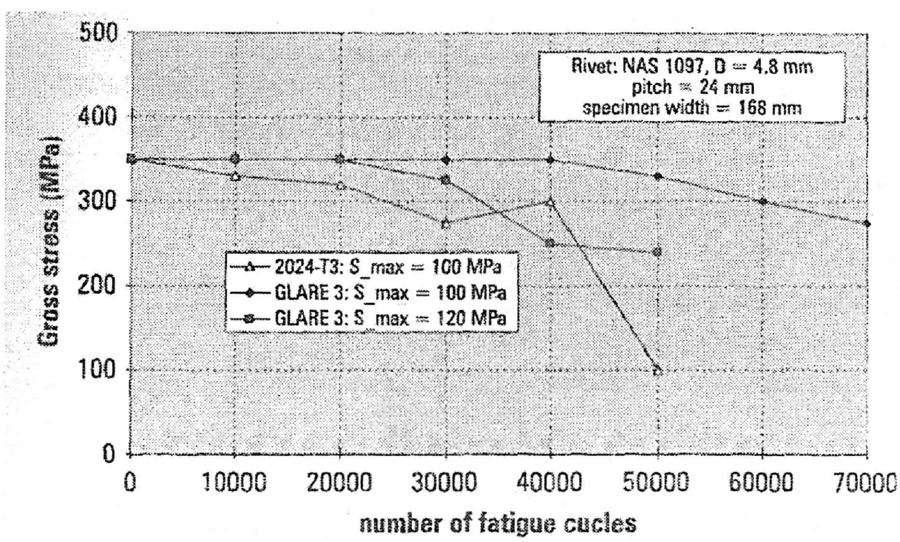


Fig.12: Residual strength of 2024-T3 and GLARE 3-3/2-03 riveted lap joints after fatigue.

the GLARE lap joints show a superior behaviour over the aluminium alloy lap joints; a high initial residual strength and a slow strength reduction are indicated. The 2024-T3 riveted lap joint shows a sudden decrease of the residual strength after through-the-thickness cracks (MSD) are present. Relatively short inspection intervals are then required to prevent unstable crack extension as it occurred in the Aloha B737 accident (MSD failure). A similar disaster would not have occurred in GLARE joints.

Residual strength investigations also covered situations of damage somewhere in the skin of a fuselage, which can be fatigue damage with the fibres still intact or impact damage which has cut the fibres, e.g. by a rotor burst. Illustrative results are shown in Fig.13 for GLARE 3-3/2-0.3. The residual strength of a GLARE sheet with a fatigue crack is substantially superior to the residual strength of a GLARE sheet with through-the-thickness damage (saw cut) due to the intact fibres in the wake of the fatigue crack. Furthermore, the residual strength is hardly affected by the relative crack length.

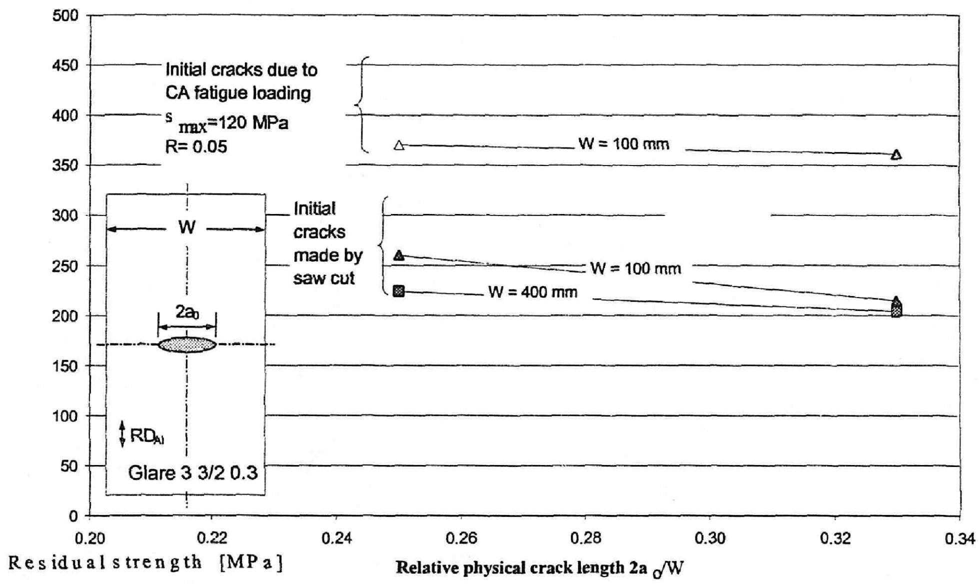


Fig. 13: Residual strength as a function of relative crack length and initial crack creation for GLARE 3-3/2-0.3

Comparative experiments on residual strength were also made by adopting the classical R-curve concept, which is the stable crack growth resistance curve under quasi-static loading. The tests are started with a initial through the thickness damage, usually a saw cut. The energy available for crack growth (energy release rate *G*) must be larger than the crack resistance *R* of the material. Results shown in Fig.14 have been corrected for different specific weight of the materials, and instead of presenting the R-values, the corresponding stress intensity factor K is adopted. The K_R-curve is much better for GLARE 2 than for Alclad 2024-T3. Differences between the GLARE variants are associated with the amount of fibres being cut. It turns out that the K_R-curve depends on the fibre volume content, the properties of the metal and fibre layers, the interfaces between these layers, the rolling direction in comparison with load direction and the quantity of fibres in load direction [7].

To fulfil the so-called two-bay crack criterion after FOD (foreign object damage), the residual strength of a GLARE skin can be locally improved by adding extra "crack stopping" fibre layers in the laminate. A test on a skin panel showed a positive result because the crack tended to flap to the centre of the stiffener bay rather than giving an unstable final failure.

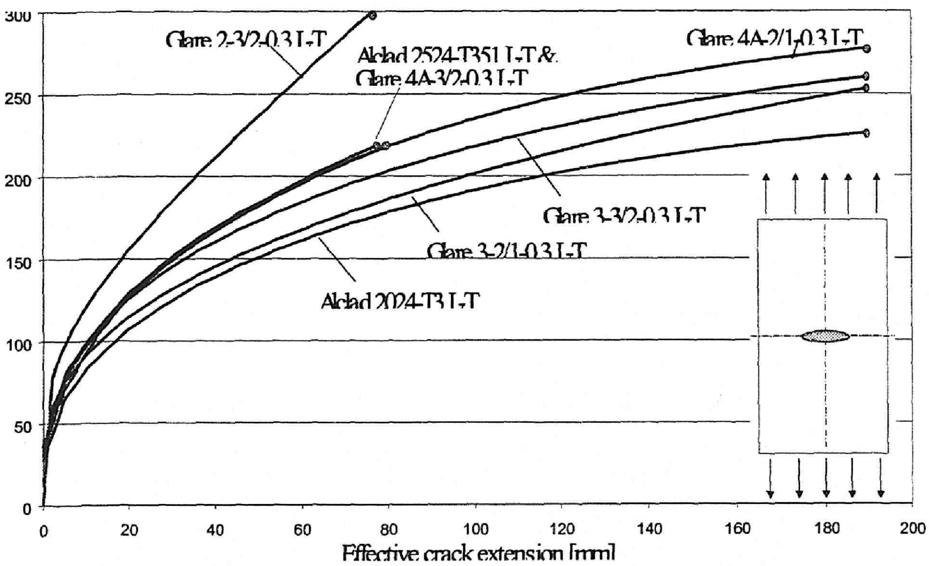
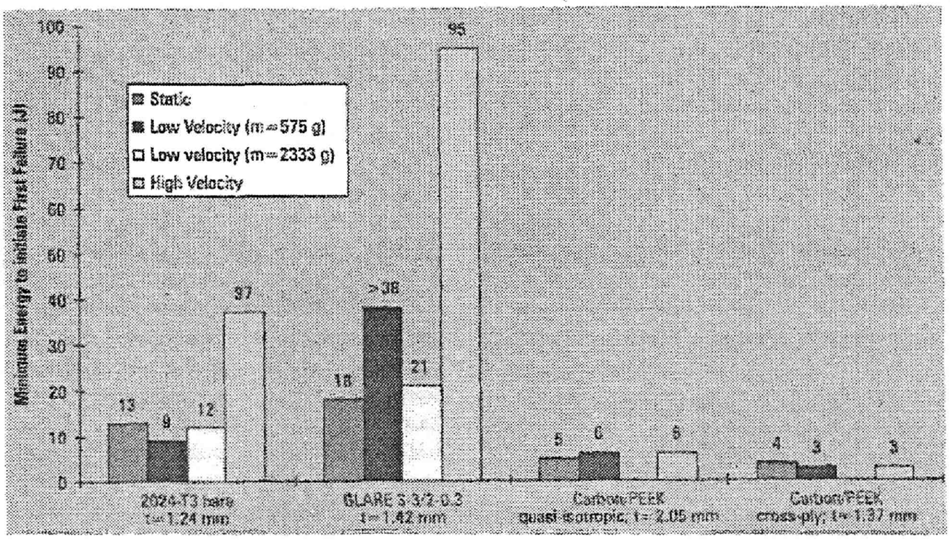


Fig. 14: Indexed K_R-curves obtained with compliance correction for different GLARE laminates and aluminium alloys in L-T direction

IMPACT RESISTANCE

Impact damage is a very relevant type of damage for aircraft structures. Impact damage can be caused by low and high velocity sources: runway debris, hail, maintenance damage (i.e. dropped tools), collision between service cars or cargo and the structure, bird strike, ice from propellers striking the fuselage, engine debris, tire shrapnel from tread separation and tire rupture, and lightning strike. A comparison of the minimum energies to cause first failure is presented in Fig.15 for different materials [8]. For the full range of thickness, GLARE shows a higher resistance to cracking than non-clad 2024-T3 in a standard drop-weight set-up or a gas gun experiment. The favourable impact performance of GLARE is attributed to a high strain rate strengthening phenomenon occurring in the glass fibres and the relatively high failure strain of the fibres. The size of the inside damage is smaller than the visible outside dent. Impact damage of GLARE is always visible due to the plastically deformed dent in the outside metal layer. Visually inspection will reveal the existence of impact damage.



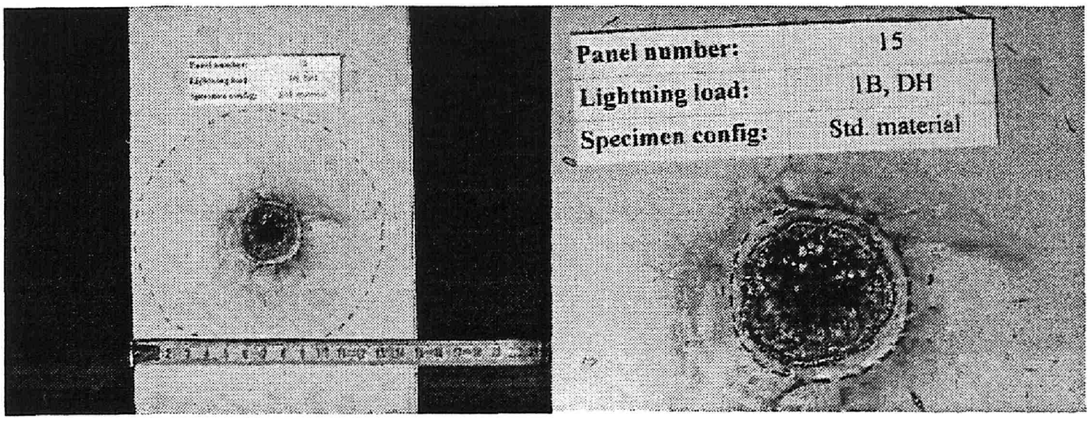


Fig. 16: Details of damage due to lightning. The outer aluminium layer is melted and part of the first fibre layer is damaged after a 1B Direct Hit.

Under cyclic loading applied after impact damage is present, fatigue cracks will not grow at the inside of the structure because of the out of plane bending of the dented area. A fatigue crack will initiate at the inspectable outside of the structure, but again the crack growth rate will be low.

Compression tests after impact damage was present revealed a buckling strength for GLARE similar to the value for 2024-T3. No delamination buckling or growth was found.

Lightning strike tests performed on GLARE showed surface damage only. The outer aluminium layer was locally melt and a small debonded area around the point of impact was visible. The second aluminium layer was still intact. The damage is restricted to the outer aluminium layer and the first underlying fibre layer (see Figure 16).

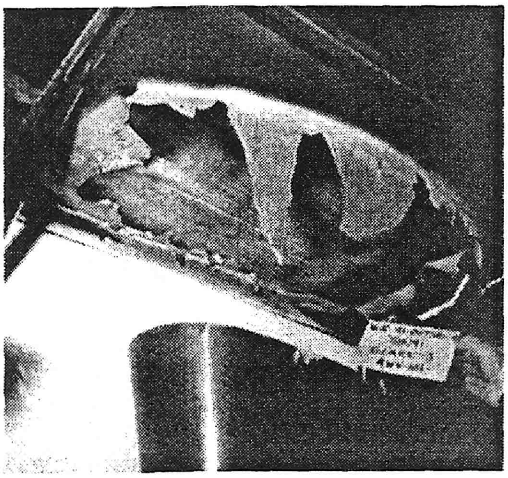


Fig. 17: Typical damage on a GLARE panel after 10 minutes of exposure to 1100 °C flames (courtesy Boeing).

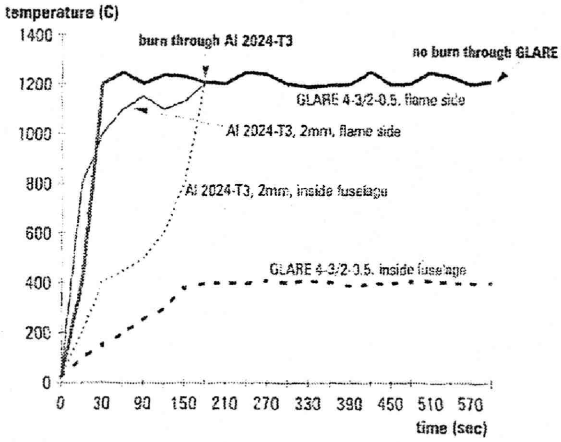


Fig. 18: Measured temperature during flame Penetration test of GLARE 4-3/2-0.5 and 2024-T3 (t=2mm)

FLAME RESISTANCE

The flame resistance of GLARE is extremely good. Current aircraft fuselages with aluminium alloy skins will melt away in 20-30 seconds in case of an outside kerosene fire. As a consequence, a number of passengers can be exposed to those flames within the 90 seconds escape time required by the airworthiness authorities. GLARE has shown the capability to resist fire conditions for much longer periods. Although the top layer will rapidly melt away, the fire then meets with s-glass fibres with a high melting point. In combination with the insulating delamination zone filled with air, the second aluminium layer is protected from melting for a significant period of several minutes. It will therefore protect the passengers for a substantially longer period, see Figs. 17 and 18. GLARE as a fuselage skin material will also protect the fuselage structure (skin, stringers and frames) for a long period against an outside fire, which will ensure the structural integrity of the fuselage during that period.

CORROSION AND DURABILITY

The fibre/epoxy layers are only exposed to moisture at the edges of the laminates. Moisture penetration is therefore limited and edges can be sealed if necessary. All aluminium sheets used in the production of FML's are anodised and coated with a corrosion inhibiting primer prior to the bonding process in an autoclave Furthermore, the outer aluminium surfaces can be supplied with a thin clad layer to improve surface corrosion resistance if desirable. Through-the-thickness corrosion is prevented due to the barrier role played by the fibre-epoxy layers. This limits the extent of corrosion damage in severe environments. The barrier role is illustrated by Fig.19, which shows a cross section of a GLARE sheet exposed at both sides to a very aggressive corrosion agent. The laminate is pitted in the outer metal layers only.

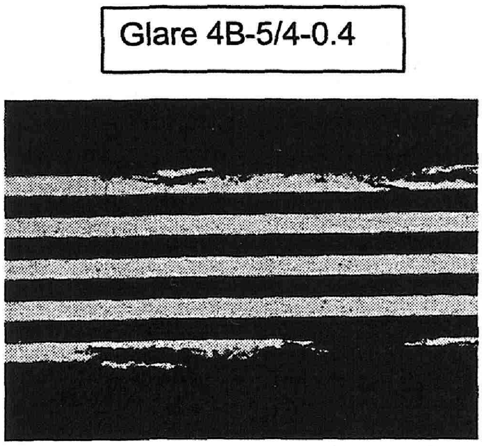


Fig. 19: Corrosion of GLARE 4B-5/4-04 after175 hours of EXCO testing.

INSPECTION AND REPAIR

After production of GLARE sheets and components a through-transmission ultrasonic C-scan inspection is used to assure a high quality laminate. A C-scan quality assurance method especially developed for and dedicated to complex GLARE panels was established in Delft and is now used for production. [9].

When fatigue cracks occur in a FML component, crack initiation can occur below the material surface. For example, with riveted lap joints, the inner metal layer at the countersunk side is the first one, and probably the only one, in which cracks will be present. Subsurface cracking has been observed by using eddy current techniques. The eddy current method is capable of finding small cracks (3 mm in length) in the third aluminium layer of a 3/2 lay up. However, for a well designed GLARE structure, it may be shown that inspections for fatigue cracks need not be required because the crack will be limited to one layer and the residual strength will be higher than the ultimate design load (see Figure 20).

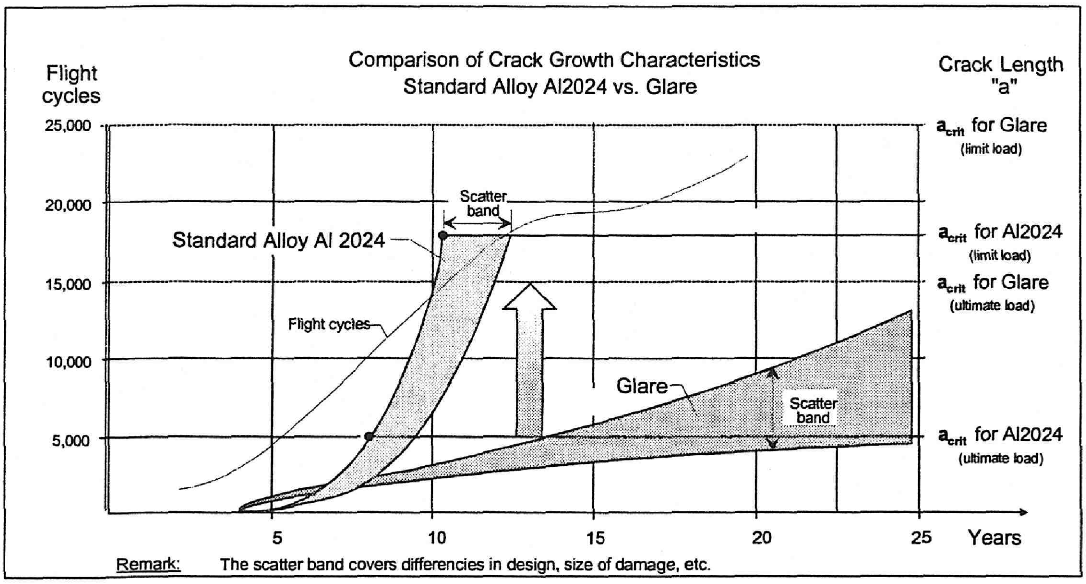
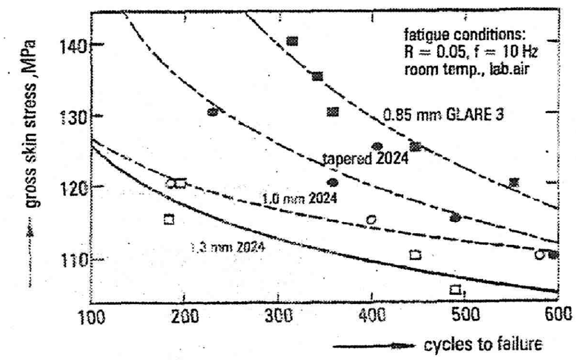


Fig. 20: Outstanding crack growth resistance of GLARE

Damage-tolerant repairs of a FML structure can be accomplished with similar procedures and techniques as used for monolithic aluminium alloy structures. Conventional riveted or bonded patch techniques have been shown to perform very well.



Source: Fredell (1994)

Fig. 21: Beneficial effect of riveted GLARE patches on the fatigue lives of repaired monolithic aluminium skins

An interesting sideline of the repair techniques for aircraft structures is the promising application of GLARE 3 and 4 as patch material to obtain damage tolerant repairs of cracked aluminium skins. The high blunt notch strength, moderate stiffness and excellent fatigue resistance of cross-ply GLARE laminates implies that the riveted "soft patch" of GLARE is an ideal solution for the repair of incidental damage in fuselages of monolithic sheet material [10]. Fatigue lives of the surrounding skins are increased by the more favourable load transfer that occurs into the soft patch of GLARE as shown in Figure 21.

Another interesting repair technique is offered by adhesively bonded patches of unidirectional GLARE 1 and 2 laminates. Up to now, very high modulus materials, especially boron-epoxy, were considered for bonded crack patching. However, the moderate stiffness and the small thermal mismatch of GLARE with the aluminium structure make GLARE a better solution which has already been applied in service, see Figure 22. Bonded GLARE patches are very well suited for life extension programs of ageing transport aircraft suffering from multiple site damage.

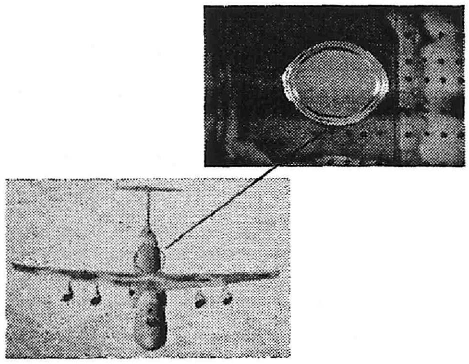


Fig. 22: Bonded GLARE patch repair on crown of C5 Galaxy

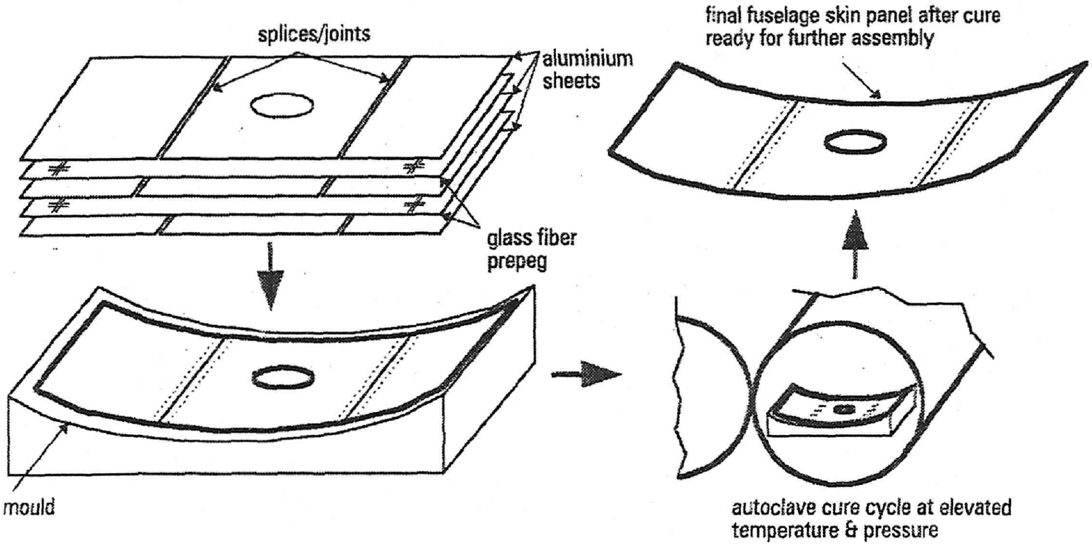


Fig. 23: Production of a curved panel with splices

MANUFACTURE OF LARGE GLARE PANELS

Research and development on FML projects in The Netherlands are planned, co-ordinated and carried out by the Fibre Metal Laminate Centre of Competence (FMLC). The Fokker Aircraft Industry, the National Aerospace Laboratory (NLR) and Delft University of Technology are represented in this foundation. An important project of FMLC was the production technique of large GLARE panels for fuselage structures. Some aspects will be summarised.

The fact that thin aluminium sheets (0.2 – 0.4 mm) are used in combination with thin pre-pregs (adhesive layers with unidirectional fibres) has one general advantage: it is possible to cure in an autoclave a large panel in a mould with a smooth radius, thus producing a curved (or even double curved) panel as needed for fuselage sections. In order to be able to produce very large GLARE panels without joints, FMLC developed the splice concept [11]. The size of the panels can be increased by having the aluminium sheets overlap one another. Since there is no size limitation of the pre-preg layers, this results in a "continuous" laminate. The splicing concept is illustrated in Figure 23. The size limiting factor now is the size of the autoclave. The maximum panel size was increased to 16 x 6 meters^2. As part of this production process local stiffening elements and cut-outs can be included simultaneously. The integration of the production steps and the possibility to produce large panels reduces the manufacturing and assembly costs. As a result, it is possible to produce more (structurally) efficient fuselage panels in a more cost-effective way, resulting in less expensive panels (see Figure 24).

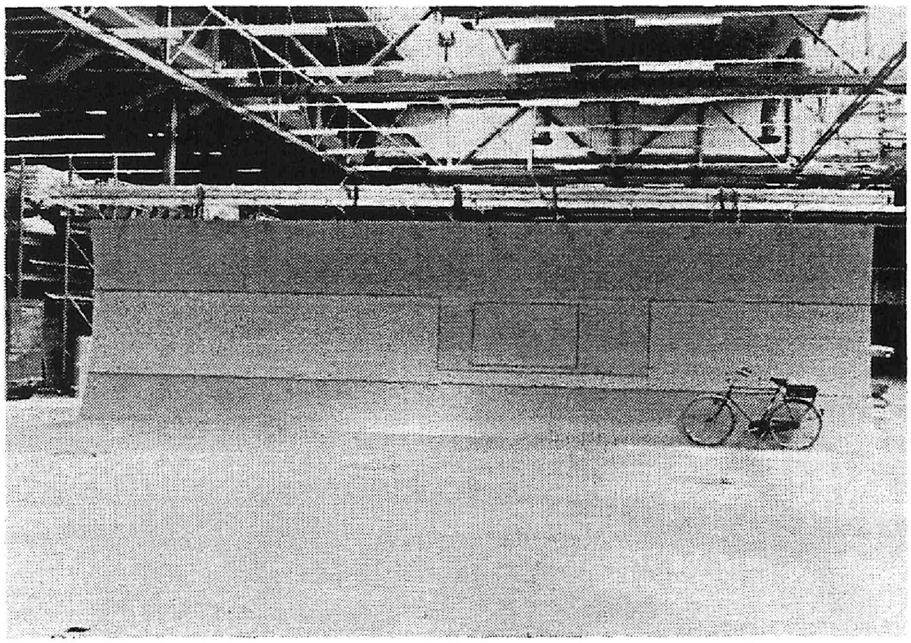


Fig. 24: The second single curved panel (GLARE 3-4/3-0.4) including 3 doublers

SUSTAINABLE DEVELOPMENT

The contribution of GLARE to sustainable development was recently studied [12]. Life cycle analysis on a fuselage section including pre-treatment and bonding of the laminates in comparison with conventional monolithic aluminium structures proved a 27 % total (i.e. manufacture and use phase) lower environmental impact of the GLARE structure. The manufacturing phase of the GLARE structure has a 50 % lower environmental impact, primarily due to the lower aluminium content of the laminate and the lower buy/fly ratio that is possible with the bonded structure compared to the milled aluminium equivalent. Recycling techniques for the scrap material was developed. The aluminium is separated from the fibre/epoxy fraction by granulation at very low temperatures that makes use of the difference in coefficient of thermal expansion, followed by separation by an eddy current sorter.

FURTHER PROSPECTS OF THE FML CONCEPT

The present GLARE variants have proven to be a promising family of structural materials for aerospace applications. GLARE was developed for aircraft structures as a fatigue insensitive material with a low density. But as part of the GLARE development programme, it was also shown that the fibre-metal laminates have a range of other promising properties. The concept of FML's can easily be extended to other structural components. New laminates can be developed by introducing other metals, fibres and adhesives. Some proposals which are now considered or already under way are mentioned below.

1. New aluminium alloys for FML's can enhance the mechanical properties: the static properties with alloys of the 7000 series and the damage tolerance properties with the 2524 alloy (Fig. 25).

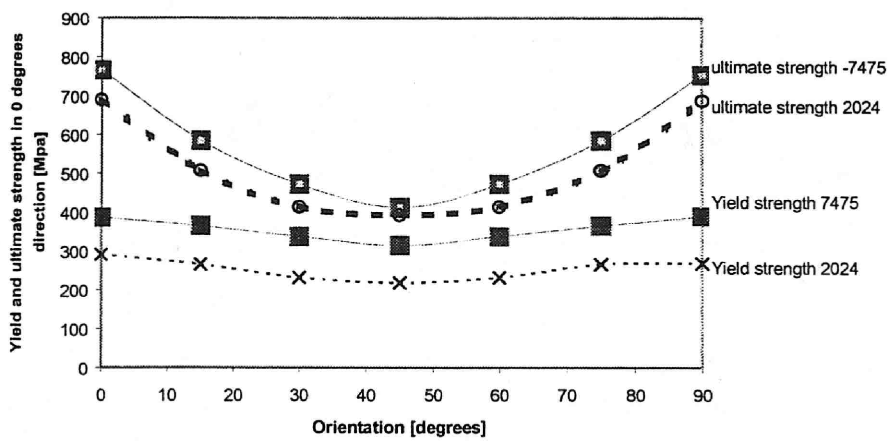
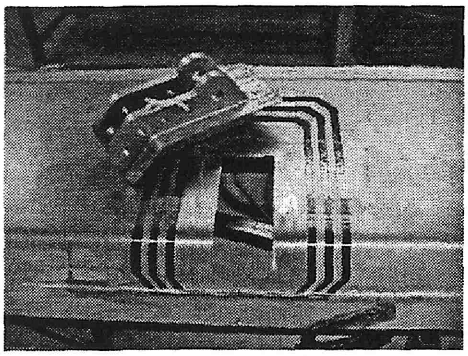


Fig. 25: Yield and Ultimate strength of two variants of Glare 3 (with 2024 alloy and 7475 alloy) under off-axis loading

For certain applications in aircraft components an improvement of the shear strength is important. The shear strength depends on the behaviour of the aluminium alloy. An aluminium alloy with higher yield stress can improve the shear properties of the laminate up to 15% if compared to the 2024 alloy [13].



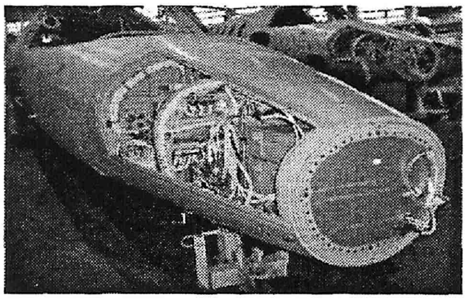


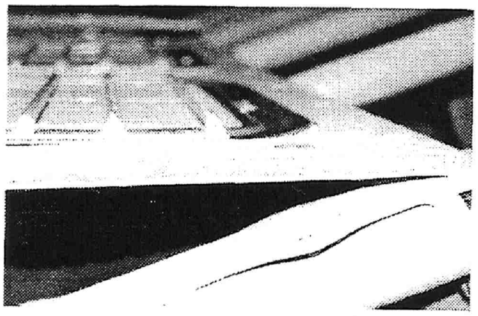
Fig. 27: SHORTS Lear 45 Radome Front Bulkhead

Fig. 26: Carbon-Titanium flap structure

2. Application of GLARE at elevated temperatures is possible but it requires a replacement of the standard epoxy adhesive with a 177°C curing epoxy system and metal sheets of 2024-T81 instead of 2024-T3. This GLARE grade can be used up to 180°C. In addition, most of the mechanical properties of this new FML are increased with 15% [14].

   Application of FML's at still higher temperatures are considered, but then the aluminium alloy sheets have to be replaced by more temperature resistant titanium alloys or steel. Carbon and M5 fibres can replace glass fibres and other resin systems (Peek, Pei, Phenolics etc.) can replace the epoxy based adhesive. The advantage of these laminates is the combination of high stiffness, high yield strength, good fatigue and impact properties at elevated temperatures (see Fig. 26).

3. Special GLARE grades for high impact resistance. GLARE type 5 has been optimised for floors in passenger and cargo areas [15]. It outperforms other solutions such as aluminium alloys. An increasing number of aircraft operators fly with GLARE 5 floor material. Other applications coming into perspective for reasons of impact damage are cargo barriers, leading edges and bulkheads (see Figs.27 and 28).



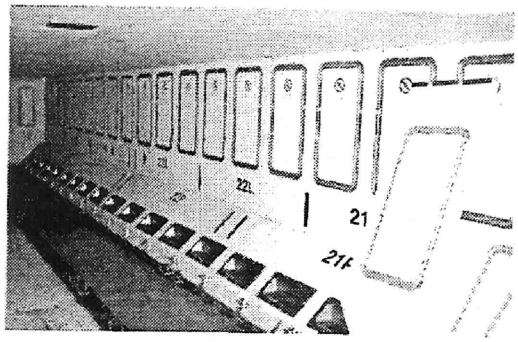


Fig. 28: Garuda A330 cargo bay floors and liners made of GLARE

4. Firewall and fire- and blast-resistant cargo containers. Several GLARE grades have shown fire resistance in a number of qualification tests. This is combined with significantly better impact properties (in comparison with glass composite firewalls) and more simple attachments to the aircraft structure. Galaxy Aviation Security designed and tested an advanced GLARE container (see Fig.29). This cargo container is able to resist the blast of two times the amount of semtex used by the terrorists for the Lockerbie disaster. After the blast the container withstood also the intense heat of the resulting fire.
5. Seamless FML tubes. FML tubes were originally intended to be used as impact energy absorbers for helicopter struts and aircraft seats. Other applications of interest were construction elements of space stations, floor struts in aircraft, bicycles and pipelines in the chemical industries. In the letter application, a through crack in one metal layer does still not lead to leakage. Different kinds of metal-fibre combinations can be thought of depending on the application. A special production technique has been developed. (see Fig. 30)

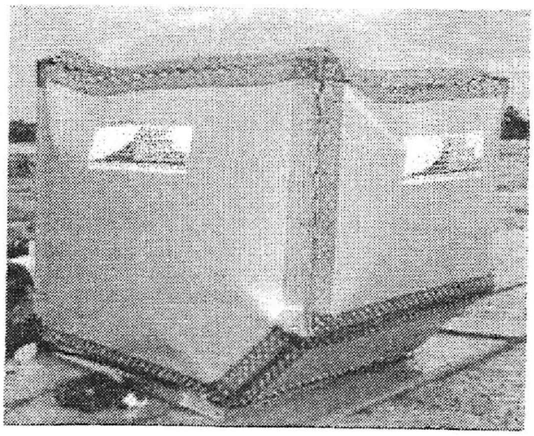


Fig. 29: Cargo container after explosion test

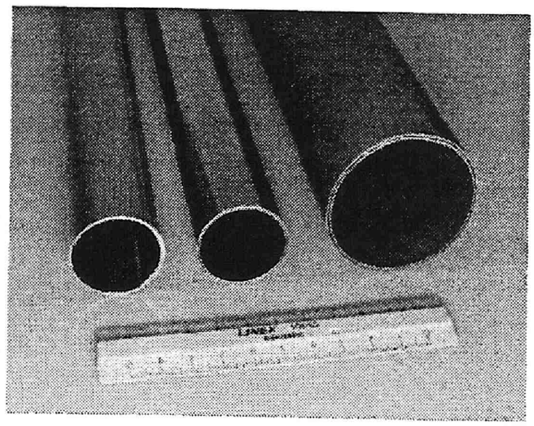


Fig. 30: Seamless tubes

CONCLUDING REMARKS

The introduction of a new material in aircraft structures is a problematic question for different reasons. It is remarkable that economical arguments are important for a new material for civil aircraft, whereas aircraft performance arguments are more dominant for military aircraft. If both economy and aircraft performance can benefit from a new material, the prospects look bright. Anyway, economic consequences must be evaluated. The two most important parties involved are the aircraft industry and the aircraft operator. Hesitation, not to say suspicion, to introduce a new material in a civil aircraft can be understood because of the risk associated with the new material. An illustrative example occurred several decades ago when the 8-1-1 Ti-alloy was chosen for a Mach-3 aircraft. Large investments were already made when it turned out that the stress corrosion resistance in salty environments was unacceptable. Other unfortunate cases happened in the past with composites for general aviation aircraft. Managers in the aircraft industries do not appreciate risks with unknown consequences. Well-established and conservative options are frequently preferred. Introduction of a new material may then be difficult.

In the civil aviation society, substantial hesitation comes from the airlines and the airworthiness authorities. A new material will be welcome to the airlines provided it can be guaranteed that maintenance will be cost-effective, i.e. reduced inspections, less and easy repairs, easy replacements, etc. Reduced weight in itself is not a straightforward argument for them, but reduced fuel consumption is significant. The airworthiness authorities, being responsible for safe aviation, are the other party asking for proven experience which by nature of a new concept is non-existing.

In the light of the above arguments, the development of the GLARE family of fibre-metal laminates is noteworthy. For the people involved in this development, it took years. However, developing GLARE in a period of some 15 years to become a mature option for aircraft material selection is a relatively short period, also in view of the fact that some service experience is already available. However, when starting the development originally in the Structures and Materials Laboratory of the Faculty of Aerospace Engineering, we have always realised that the broad range of problems had to be tackled right from the beginning, and not just one after the other one. This has led to investigating immediately all technological problems involved such as listed in Table 1. As an example, do not only determine the static properties (S_U, $S_{0.2}$, elongation), but also immediately the blunt notch strength and the residual strength in cracked condition. Also production aspects (machining, plastic forming, and quality control), impact and corrosion, and joints were considered from the early days of the development. Of course, some aspects were favourable for Delft. The knowledge of fatigue problems in aircraft materials, joints and aircraft structures was well developed in the Netherlands. Secondly, the culture of adhesive bonding was also present because of the wide experience of the Fokker Aircraft Industry. Furthermore, experience was already gained with ARALL. Last but not least, a close co-operation with the aircraft industry has significantly contributed to stimulate the development of GLARE.

GLARE is now available as a material for aircraft structures. However, the better statement is that GLARE can now be part of designing aircraft structures. The type of GLARE must be selected for the purpose and the function of the specific structural component. The selection for large parts of the Airbus 380 fuselage skin is a milestone, but it should not be overlooked that the GLARE concept can also be profitable for other type of structures. The luggage container of Fig.29 is just one example. Even for a small component as a lug-type connection between larger aircraft components, a GLARE component can be most attractive in view of the extremely slow crack growth, which will make the connection very much damage tolerant.

GLARE as a fibre-metal laminate is a hybrid material with many variants, a material, which is by definition much different from a monolithic material. As a result, designing in GLARE will be a more demanding and challenging job. Let it be known it can offer great satisfaction and a profit for the society.

REFERENCE LIST

(1) Vogelesang, L.B. and J.W. Gunnink, *ARALL: A material challenge for the next generation of aircraft.* Materials and Design, Vol. 7, No. 6, 1986.

(2) Vlot, A., *Glare - history of the development of a new aircraft material.* Kluwer Academic Publishers, Dordrecht, 2001, ISBN 1-4020-0124-X

(3) Vlot, A., *Low-velocity impact loading,* PhD Thesis report, Delft University of Technology, 1999.

(4) Schijve, J., H.T.M. van Lipzig, , G.F.J.A van Gestel and A.H.W. Hoeymakers, *Fatigue Properties of Adhesive-Bonded Laminated Sheet Material of Aluminium Alloys,* Engn. Fracture Mechanics, 12, 1979, pp. 561-579.

(5) Roebroeks, G.H.J.J., *Towards GLARE: The development of a fatigue insensitive and damage tolerant aircraft material,* PhD Thesis, Delft University of Technology, 1991.

(6) Marissen, R., *Fatigue Crack Growth in ARALL, a Hybrid Aluminium-Aramid Composite Material,* PhD Thesis, Delft University of Technology, 1988.

(7) Vries, T.J. de, *Residual Strength.* FML an introduction, A. Vlot and J.W. Gunnink (eds.), ISBN: 1-4020-0038-3, 2001, pp. 197-217.

(8) Vlot, A., et al, *Towards application of FML in large aircraft,* Aircraft Engineering, Vol. 71, No: 6, 1999.

(9) Coenen, R.A.M., *Design of a quality assurance system for structural laminates,* PhD Thesis, Delft University of Technology, 1998.

(10) Fredell, R.S., *Damage tolerance repair techniques for pressurized aircraft fuselage,* PhD Thesis, Delft University of Technology, 1993.

(11) Roebroeks, G.H.J.J., *The development of the splice geometry,* SLI Report TD-R-93, Delft, 1993.

(12) Tempelman, E., *Sustainable transport and advanced materials,* PhD Thesis, Delft University of Technology, 1999.

(13) Auffret, A. and A. Gennai, *New aluminium alloys for GLARE,* FML an introduction, A. Vlot and J.W. Gunnink (eds.), ISBN 1-4020-0038-3, 2001, pp. 79-97.

(14) Boer, T. de, *New generation FML,* FML an introduction, A. Vlot and J.W. Gunnink (eds.), ISBN 1-4020-0038-3, 2001, pp. 39-51.

(15) Roebroeks, G.H.J.J., *Features FML,* FML an Introduction, A. Vlot and J.W. Gunnink (eds.), ISBN 1-4020-0038-3, 2001, pp. 23-37.