DELFT UNIVERSITY OF TECHNOLOGY

DEPARTMENT OF AEROSPACE ENGINEERING

Report LR-276

FATIGUE PROPERTIES OF ADHESIVE-BONDED LAMINATED SHEET MATERIAL OF ALUMINUM ALLOYS

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ABSTRACT

Comparative fatigue tests were carried out on centrally cracked specimens and lug type specimens, both made from solid sheet and laminated sheet, consisting of five 1 mm sheets of 2024-T3 Alclad material bonded by FM 123/5. Most tests were carried out under constant amplitude loading but growth delays due to peak loads were studied also. Observations are made for through cracks and part through cracks. The significance of the results for application in aircraft structures is analysed.

NOTATIONS AND UNITS

a	crack length
da/dn	crack rate
K	stress intensity factor
R	stress ratio = S_{min}/S_{max}
S	stress
t	thickness

1 MPa = 1 MegaPascal = 10^6 N/m^2 = 0.145 ksi = 0.1020 kgf/mm^2

1 MPa$\sqrt{m}$ = 31.62 N/mm$^{3/2}$ = 0.910 ksi$\sqrt{in}$ = 3.255 kgf/mm$^{3/2}$

1 µm/c = 10^{-3} mm/cycle

1 mm = 0.0394 inch

1 INTRODUCTION

Laminated sheet material is produced by adhesive-bonding of a number of thin sheets to obtain a larger thickness, see Figures 1 and 2. A superior behavior with respect to fatigue crack growth and fracture toughness is related to the following arguments.

1. Thickness argument (plane stress/ plane strain)
Usually fatigue crack growth in thin sheets is somewhat slower than in thick sheets and plates. The classical explanation refers to plane stress along the crack front in thin sheet and plane strain in the thick material. If a thick section is then built up from a number of thin sheets, bonded by adhesive layers with a low stiffness, it is expected that the superior growth properties can be maintained partly or fully in the laminated material.

2. Peak load delay argument
The thickness argument should apply under constant-amplitude fatigue loading. However, it could be much more significant under variable-amplitude loading if high peak loads occur. Such loads produce larger plastic zones in thin sheets and longer crack growth delays can be expected. Under variable-amplitude loading the laminated material may be even more superior to solid (or monolithic) material than under constant-amplitude loading.

3. Crack arrest argument
Many cracks in service start as part through cracks and the initial growth also occurs in the thickness direction. Obviously this will be difficult in laminated material, because the adhesive layers will arrest crack growth for some time. As a result crack growth in the width direction will also be reduced as a consequence of restraint on crack opening by adjacent non-cracked laminas.

4. Fracture toughness argument
It is welknown that K_c for a thin sheet(plane stress)is higher than a valid K_{Ic} for a thick plate (plane strain) and this adventage can apply to laminated material.

In the past few years comparative fatigue tests on laminated sheet material and solid material of 2024-T3 Alclad were carried out in the Department of Aerospace Engineering of the Delft University by Van Lipzig (Ref. 1), Van Gestel (Ref. 2) and Hoeymakers (Ref. 3). Selected results will be recapitulated here, while results of some additional tests on the effect of peak loads will be presented also. Part of the results were published before, but got a limited circulation (Refs 4-7). Some older results of full-scale tests (Refs 8 and 9) on wing panels, related to the merits of laminated material, will be discussed briefly. The paper is completed by a general discussion and a number of conclusions.

2 CRACK PROPAGATION TESTS ON LAMINATED AND SOLID MATERIAL

Materials

All specimens were produced from 2024-T3 Alclad material. The thickness of thin and thick sheets were 1 and 5 mm respectively. The laminated material was obtained by bonding 5 sheets of 1 mm thickness with FM 123-5 as an adhesive (curing temperature 125° C). Bonding was carried out at the Fokker Aircraft Factories with anodizing as a pretreatment. The average thickness of the adhesive layers was 0.24 mm. In tensile tests (specimen width ∿ 16 mm) the following static properties were obtained (loading direction = rolling direction)

Thickness	S_a(MPa)	$S_{0.2}$(MPa)	δ = Elongation (%) L = 22.6	L = 50.5 mm
1 mm	452	340	24.1	
1 mm (anodized)	442	349	20.4	
5 mm	479	369		20.4
5 x 1 mm (anodized)	453	346	(26.1)	20.8

In several European countries the gage length (L) for measuring the elongation (δ) is normalized on 5 x diameter for a round specimen or the equivalent $5 \times \sqrt{A/\frac{\pi}{4}}$ (A = cross section) for a rectangular cross section. The geometrical similarity then allows a direct comparison of the elongations for different thicknesses. As a result L = 22.6 mm applies to a thickness t = 1 mm and L = 50.5 mm applies to t = 5 mm. It then turns out that δ is the same for the solid material (t = 5 mm), the laminated material (5 x 1 mm) and the anodized thin material (t = 1 mm). However, if compared for the same gage length L = 22.6 mm the elongation for the laminated material (26.1 %) is much larger than for a single sheet (20.4 %). Consequently the elongation results indicate that the plastic deformations in the necked region for t = 5 and t = 5 x 1 mm were quite similar. In other words necking in the laminated material did not occur as if the five sheets were fully separated sheets.

Through cracks

The dimensions of the crack propagation specimens are shown in Figure 3 while test conditions for the constant-amplitude tests are given below.

S_m (MPa)	S_a (MPa)
120	60
100	50
80	40
60	30

R = 1/3

Frequency 12 - 25 Hz

Three tests at each stress level

As an illustration average crack growth curves for one stress level are given in Figure 4. Scatter between three similar tests was very low. Crack propagation rates (da/dn) for all stress levels were plotted as a function of $\Delta K = C\Delta S\sqrt{\pi a}$ with C is the width correction factor. This resulted in very narrow scatter bands for each type of specimen indicating that ΔK correlated the data extremely well. The average curves are presented in Figure 4 which shows similar growth rates for the thin sheet and the laminated material and somewhat higher rates for the thicker material.

The results confirm the trend that crack rates are faster in thick solid material. This trend is supposed to be normal in view of a more predominant plane strain situation at the crack tip in thicker material. However, it should not be overlooked that other causes can also contribute to the thickness effect. The production history of the 1 mm material is not exactly the same as for the 5 mm material. Differences can occur with respect to strain hardening, heat treatment, residual stresses, inhomogeneities, etc. Some systematic work on the thickness effect is now carried out in our laboratory.

Figure 5 indicates practically similar crack rates for the thin sheet material and in the laminated material built up from 5 thin sheets. This suggests that crack growth in each of the 5 lamina was not affected by the presence of the other bonded lamina. However, fractographic observations revealed a different picture. The transition from the tensile mode to the shear mode started in the outer sheets, and as shown in Fig. 1, the transition in these sheets can be complete at a moment that the three inner sheets are still in the tensile mode. This is further illustrated in Fig. 6 where the average crack length at transition has been plotted. The corresponding ΔK-ranges are indicated in Figure 5. The results show that the transition behavior of the laminated material is not fully similar to that of the solid material, but it deviates considerably from the single thin sheet behavior. Consequently lateral contraction along the crack front in the laminated material cannot fully develop to the same extend as in the single thin sheet, in spite of the low elastic modulus of the adhesive.

The lateral contraction along the crack front in the laminated material will load the adhesive layer in tension. A reduction of the contraction should be expected depending on the elastic modulus and the thickness of the adhesive. A simple estimate of this effect on the plastic zone has been made in the Appendix of this paper and the result is shown in Figure 7. Apparently, the adhesive layers have a non-negligible effect on the plastic zone which was confirmed by the transitional tensile/shear mode behavior. The effect implies a tensile stress (S_z) in the adhesive layer. This might lead to an adhesive failure at the interface between adhesive and metal or a decohesion failure in the adhesive, but microscopical evidence of such a failure was not found. Apparently the quality of the bonding was good. It is then interesting to note that a poor bond quality could induce a local delamination around the crack front, implying a better approximation of the plane stress situation and possibly an improved crack growth resistance.

One comment on Figure 5 has still to be made. Crack growth in the laminated material was slower than in the solid material. For the laminated material ΔS employed for calculating ΔK was based on the cross section of the 5 metal sheets only, assuming that the thin adhesive layers will hardly contribute to load transmission in view of the low elastic modulus ($\sim$ 4000 MPa as compared to 72000 MPa for the Al-alloy). However, the 4 adhesive layers (average thickness 0.24 mm, specific gravity about 1 gram/cm^3) have increased the weight with some 7 per cent. The difference between the curves for t = 5 x 1 mm and t = 5 mm in Fig. 4 is of the same order (between ΔK for same crack rate). In other words on the basis of equal weight the crack rates in the laminated material and the solid material were practically similar.

Part through cracks

Part through cracks (see Fig. 2) were made as semi-elliptical cracks a/b = 1.5) in the thick material to a depth (b) of 1 and 3 mm respectively. In the laminated material initial cracks were through cracks in 1, 2 or 3 of the five sheets. The cracks were made by spark erosion which required some special arrangements for the deeper sheets. The majority of tests was carried out at 80 $\pm$ 40 MPa (20 Hz) and results for this stress level will be shown only.

An overall impression of the comparison between the various configurations is most easily obtained from Figure 8, which gives the full crack growth life until failure of the specimens. Each result is the average life of three similar tests. More detailed results are shown in Figures 9 and 10 by plotting the crack growth rate as a function of crack length. In the solid material the part through crack is initially growing at a slower rate then the through crack. However, after the crack has penetrated through the full thickness the growth rate rapidly approaches the crack rate of the initial through cracks. The behavior of part through cracks in the laminated material is completely different as should be expected. Penetration through the thickness is considerably retarded by the adhesive layers. This is especially evident if the initial crack is present in the front sheet only (lower curve in Fig.10). In the beginning of the test crack growth is accelerated from a = 3 to a = 5 mm but since the second sheet is still uncracked the growth rate slows down for a considerable period.

Crack growth in the internal sheets cannot be observed directly. However, the length of the invisible cracks was measured indirectly by the electrical-potential method (Ref. 10) in the second and the fourth sheet. This can successfully be done because the adhesive layer is a good electrical insulator. Illustrative results for specimens with an initial through crack in one sheet only are shown in Figure 10. Apparently a crack in the second sheet started when the crack in the first sheet had grown to a length of 13.5 mm. No observations were made for the third sheet, while in the fourth sheet a crack started when the crack length in the first and second were 25.5 and 22 mm respectively. Crack growth is significantly retarded if adjacent sheets are still uncracked. Obviously this implies a severe restraint on crack opening. Crack propagation rates of the individual sheets can be used to derive effective stress intensity factors from the curve for thin sheet in Figure 4. These empirical factors were compared to the theoretical values for an unrestrained through crack with the same crack length. The ratio

$$C_s = \frac{\Delta K_{empirical}}{C \Delta S \sqrt{\pi a}}$$

has the character of a tip stress reduction factor. Results plotted in Figure 11 show that reductions in the order of 50 per cent applied to the major part of the crack propagation life.

In the laminated specimens with part through cracks some debonding was microscopically observed on the fracture surface. The extent could not be determined, but probably it was fairly limited.

3 CRACK GROWTH DELAYS DUE TO PEAK LOADS

A rather small number of laminated specimens was still left after completing the tests of Refs 1 and 2. Some single thin sheet specimens were also available, but unfortunately the thicker material ($t = 5$ mm) was fully exhausted. Constant-amplitude tests were carried out at $S_m = 125$ MPa and $S_a = 25$ MPa, while subsequently similar tests were interrupted at $a = 10$ mm to apply a single peak load up to 200 MPa, see Figure 13. Similar tests were carried out previously on sheet specimens of 2 and 10 mm thickness (thesis work of Arkema reported in Ref. 11). Crack growth delays as observed from crack propagation curves are plotted in Fig. 14 which indicates that the delay period of the laminated material is significantly longer than expected for solid material of the same total thickness. However, it is considerably shorter than for the single thin sheet material, from which it was built up. In other words the behavior for $t = 5 \times 1$ mm is intermediate between $t = 1$ mm and $t = 5$ mm.

4 FATIGUE LIFE AND CRACK GROWTH RESULTS OF LUGS

Tests were carried out on lug type specimens shown in Figure 15, which were produced from the halves left from the crack propagation test reported before. Lugs made from the thicker sheet (t = 5 mm) and the laminated material (5 x 1 mm) were tested under constant-amplituding loading at a constant stress ratio R = 1/3, four stress amplitudes (30, 40, 50 and 60 MPa) and a frequency of 20 Hz. The S-N data obtained for the crack initiation life (until a = 2 mm) are shown in Figure 16, which indicated practically similar fatigue lives for the solid material and the laminated material.

Crack growth observations were made at both sides of the hole and at both sides of the lug, i.e. at 4 locations of each specimen. For this purpose slots were made in the steel fork holding the loading pin of the lug, see Figure 15. Cracks initiated in the course of a normal fatigue tests are labelled here as "natural" cracks. Crack growth from "artificial" cracks was also studied, where an artificial crack refers to a crack started by a small initional saw cut. Both artificial through cracks and corner cracks at the edge of the lug hole were produced to a depth of 1 mm. A comparison between crack growth lives for all types of cracks in made in Figure 17, which indicate some remarkable and partly unexpected results.

(1) Scatter is large for the natural cracks and small for the artificial cracks.

(2) Natural cracks have significantly longer growth lives than artificial cracks.

(3) Crack growth lives for the laminated material are longer than for the solid material.

(4) A corner crack in a laminated lug has a much longer growth life than a similar crack in a solid lug.

Fractographic observations were the key to understanding the first and the second trend. Fatigue cracks in a lug hole are initiated by fretting corrosion. Cracks can start at both sides of the hole, while at each side of the hole several crack nuclei may be found. Nucleation can occur at different time intervals and at each side of the hole at different levels, see Figure 16. As a result radial ridges will occur where different nuclei overlap. Actually crack growth is a rather chaotic process, especially when cracks are still small. The crack front is"poorly organized" and it requires some growth before it is straightened to the simple shape assumed for a through crack in fracture mechanics. Under such conditions scatter should be expected for natural cracks. In agreement with the above reasoning scatter was small for the artificial cracks because the initiation from the saw cut immediately led to a well organized crack front. A well organized straight crack front without any steps (or ridges) will move onwards more easily due to the higher stress intensity associated with it. As a result the crack growth period of the artificial crack was shorter in addition to low scatter.

The laminated material showed superior crack growth periods in all comparable cases shown in Figure 17. The smallest difference was found for the artificial through cracks and the life ratio found here agrees reasonably well with the results found for the central cracked panels (Fig. 5). The cracks in these panels also started as through cracks from a saw cut and a similar result should have been expected. For the natural cracks the difference between solid and laminated material is larger. Reorganizing an irregular crack front to a straight one is more difficult in the laminated material due to the intermediate adhesive layers. In agreement with this argument the life ratio found for the artificial corner cracks is still larger. Before a corner crack in a laminated material has grown to a full through crack the growth has been retarded significantly by crack opening restraint of still uncracked laminas. A similarly large difference in life as previously found for the part through cracks in Fig. 8 is then observed.

6 TESTS ON A FULL SCALE STRUCTURE WITH LAMINATED DOUBLERS

Several years ago (Refs 8 and 9) tests were carried out on a series of premodification F 27 wing panels, including random load tests, programmed load tests, flight simulation tests and constant-amplitude tests. The tension skin had a length of 8.31 meter and it contained two identical skin joints (Port and Star board). The skin at the joint (see Fig. 19) was locally reinforced with three thin doublers, staggered in the spanwise direction for a smooth increase of the effective skin thickness. The joint consisted of a splice plate at the outside of the skin under the fairing, while at the inside the hat stringers were uninterrupted. Cracks were found under the splice plate at the outer row of the rivet holes, which is a classical location for fatigue cracks. The cracks first occurred in the outer doubler, later in the inner doublers and ultimately in the skin. Sketches of typical crack nuclei configurations revealed after dismantling are shown in Fig. 20. Larger cracks of this type were found in the constant-amplitude tests only and crack growth could easily be followed by taking X-ray pictures. The first X-ray indications were obtained after 40 - 50 % of the total life, while the end of the test was due to final failure at another location. This implies that cracking at the joint developed very slowly due to the laminated doublers. With an integral skin reinforcement made by machining the cracks might have been the most critical ones.

DISCUSSION

Fatigue crack initiation

The present results of lug specimens indicate similar lives for laminated and solid material, see Figure 16. It is difficult to see any reason why crack initiation life should be longer for laminated material. On the contrary, the laminated material had cladding layers on all sheets and this might even enhance crack nucleation inside the hole of a specimen. In this respect laminates of bare sheet seem to be more logical, since the cladding on the inner sheets is superfluous for corrosion protection.

Growth of through cracks

Artificial cracks, starting from saw cuts were growing in the laminated material about 1.5 times slower than in the solid material. This was found both for centrally cracked panels (Fig. 4, 5 and 8) and lugs (Fig. 17). Two comments have to be made. (1) The laminated material requires some extra weight for the adhesive, which makes the improvement rather small. (2) The transition from tensile mode to shear mode suggests that the five sheets of the laminated material did not behave like fully separated sheets. In the literature Pfeiffer and Alic (Ref. 12) reported on tests on 7075-T6 CTS specimens with 8 layers (0.89 mm each) and 22 layers (0.29 mm each). Crack rates obtained were near the lower boundary of the scatter band of monolithic material, which seems to be in agreement with the present findings.

A fully different and remarkable behavior was shown by the natural cracks in the lug specimens (Fig. 17). Growth was slower than for artificial cracks, more scatter was observed and a larger difference between laminated and solid material occurred. On the average crack growth in the laminated material was about 4 times slower than in the solid material. In a previous section this was explained by referring to fractographic observations, which indicated that an natural crack had a more chaotic start, leading to a poorly organized crack front.

According to the terminology used in the literature (e.g. Ref. 13) the through cracks in the laminated material were growing in the "crack divider" direction. It wants to suggest that the cracks in the various laminas are growing more or less independently. Here this seems to apply to the natural cracks, whereas for an artificial crack all cracks apparently were growing in a fully synchronized way.

Growth of part through cracks

For part through cracks the differences between crack growth lives of laminated and solid material were large, both for centrally cracked specimens (Figs 8-10) and for lugs (Fig. 17). Slow growth of a part through crack in laminated material was also reported by Smith, Porter and Engstrom (Ref. 14) for the face sheets of a sandwich panel, consisting of three layers each (7075-T6 and 7178-T6 alloy), by Johnson and Stratton (Ref. 15) for laminated 2024-T81 material (up to 4 laminas, thickness 2.5 mm each), by Ratwani (Ref. 16) for a two-ply laminate of 7075-T73 (thickness 1.6 mm each), and by Anderson, Chu and McGee (Ref. 17) for a two-ply laminate of 7075-T6 (thickness 2.6 mm each).

For a part through crack, growth in the front sheet with the initial crack occurs in the crack divider direction. Growth in the thickness direction is labelled as occurring in the "crack arrester" direction (Ref. 13). Growth in this direction is difficult since the crack has to be re-initiated in each subsequent sheet and in view of the low stiffness of the adhesive layer the stress concentration effect of the existing crack is fairly small.

It will be recognized that the adhesive layer can exert two counteracting effects. If the shear stiffness of the layer is low (low modulus, high thickness) crack opening will hardly be restrained by the adjacent non-cracked sheets. On the other hand, crack initiation in the uncracked sheets will be a slower process. However, if the shear stiffness of the adhesive layer is high (high modulus, low thickness) crack opening will be reduced more effectively, leading to lower K-values, but reinitiation in subsequent sheets will be enhanced. Unfortunately the picture is even more complex if delamination occurs, either by a decohesion failure of the adhesive or an adhesion failure at this interface between metal and adhesive. Progressive debonding around the tip of a fatigue crack growing in a sheet under an adhesively bonded doubler was observed ultrasonically by Anderson, Chu and McGee (Ref.17). Apparently the bond quality is involved then and it is not at all obvious that a good bond quality implies slower crack growth. On the contrary it is easily speculated that either a poor bond strength or a thick adhesive layer will restrict crack growth from surface damage to the front sheet only.

The reduction of the apparent K-value due to non-cracked adjacent sheets was derived here from crack rate measurements. The results in Fig.12 show reductions to values in the order of 40 - 50 percent of K-values for a through crack. Ratwani (Ref.16) observed reductions to similar values in a two-ply specimen. He also calculated applicable K-values both analytically and with the finite element method, introducing a debonded area of an elliptical shape and dimensions as observed in the test and accounting for bending due to the crack in one of the two layers. A good agreement with the experimental values was found. Ratwani's work included the effect of thickness of the adhesive layer. Both tests and calculations indicated a somewhat faster crack growth for a thicker adhesive layer if a crack was present in one of the two sheets only. This seems a logical result. However, two adhesives (FM-400 and FM-73) with highly different elastic moduli produced almost similar crack growth rates. Ratwani notes that the critical failure strain of the high modulus adhesive was lower and about three times larger debonded areas were observed! More analytical work on the same probles is reported in the literature (Refs 17-19) which recognizes the significance of bond line stiffness and debonding. Unfortunately, the picture emerging is fairly complex, while bond line properties required are not readily available.

Peak load effects

As shown by Figure 14 crack growth delays after peak loads are larger if the material thickness is low. This trend is wellknown in the literature (Refs 20, 21). The predominant effect of peak loads in flight-simulation tests is also well documented (survey in Ref. 11). If higher peak loads are allowed in such a test (higher truncation level) crack growth is slower. Also this effect will depend on thickness and as a result crack growth under flight-simulation loading is slower in thin material (Refs 22-24). Consequently, the peak load effect shown in Fig. 14 suggests that lamination can lead to significant improvements under flight-simulation loading. Under constant-amplitude loading the improvement for a through crack (started by a saw cut) was fairly small and almost negligible if the weight of the adhesive was accounted for. However, this result is not relevant to more practical service load histories. Flight-simulation tests on laminated specimens should clarify this point.

Cracks in an aircraft structure

The results for the tension skin reported before illustrate the adventage of having laminated material. The structure just becomes more damage tolerant. McClaren and Ellis (Ref. 25) performed comparative tests on full-scale components produced either by machining a forging or by building up as a laminated sheet structure. The laminated components showed a superior fatigue behavior under some type of programmed block loading. An analysis of costs was made also which led to the conclusion that for many major components (citing these authors) "damage tolerant laminated designs can be achieved at competitive or lower weight and cost levels as compared to conventionnally machined parts".

Damage tolerant material

In practice cracks are rarely initiated as well behaved through cracks, but rather as a semi elliptical or quarter elliptical crack. According to the present results lamination of the material will then considerably improve the fatigue life. Initial crack growth will be slower and later macro growth will also be retarded depending on the load spectrum. Final failure will occur at a larger crack length, because K_c for a laminated material is significantly higher than for monolithic material as shown in several investigations (Refs 26-28). These features, slower crack growth and larger cracks tolerable, are attractive properties for designing a damage tolerant structure. Nevertheless a number of aspects requiring further study can be listed:

(a) the significance of the stiffness of the adhesive layer (including type of adhesive, curing and layer thickness),
(b) the effect of bond strength and local debonding around the crack tip
(c) omission of the cladding layer
(d) the behavior under more realistic load-time histories
(e) production problems, including quality control and economy.

CONCLUSIONS

Comparative fatigue tests were carried out on centrally cracked specimens and lug type specimens, both made from solid sheet and laminated sheet, consisting of five 1 mm sheets of 2024-T3 Alclad material bonded by FM 123/5. Most tests were carried out under constant-amplitude loading but some tests with a positive peak load were also done. Observations and conclusions are summerized below:

1. Crack initiation lives until a very small crack in the lug specimens were practically similar for solid and laminated material.

2. Through cracks in laminated specimens with a central crack were growing somewhat slower than in solid material (about 1.5 times). This also applies to lug specimens with artificially started cracks (i.e. started by a small saw cut).

3. For part through cracks a large difference was found between solid and laminated material for both types of specimens. Crack growth in laminated material was much slower.

4. Crack nucleation in lug specimens (without saw cut) did not occur simultaneously over the full thickness. As a result the initial crack growth was a somewhat erratic process with a poorly organized crack front. This caused more scatter in crack growth lives and considerably longer growth periods as compared to the behavior of artificial cracks.

5. After a positive peak load the crack growth delay period obtained for laminated specimens was significantly larger than for the solid specimens with the same thickness (t = 5 mm), although it was smaller than for the thin sheet material (t = 1 mm) from which it was built up.

6. The peak load results and observations on the transition from tensile mode to shear mode during fatigue crack growth (through cracks) indicate that the five sheets were not behaving like fully separated specimens. On the contrary, the results suggest that the adhesive layers did not fully obliterate the plane strain situation along the crack front.

7. An analysis of the present data, some older results of full-scale tests on wing panels and information from the literature indicates that laminated sheet material will significantly improve the damage tolerance quality of an aircraft structure. This conclusion is based on three facts. (a) Cracks in practice usually start as semi- or quarter elliptical cracks and initial growth will be much slower. (b) Subesquent macro crack growth under service loading with occasionally high loads should be expected to be slower also. (c) Due to improved fracture toughness a larger crack can be tolerated. However, practical application should be preceded by a series of carefully planned pilot tests.

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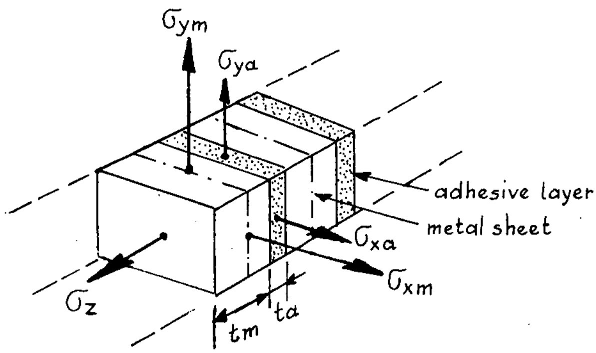
APPENDIX: PLASTIC ZONE IN LAMINATED SHEET MATERIAL

In this appendix (based on Ref. 4) an approximation of the plastic zone boundary will be made and compared with the boundary for a thin sheet (plane stress) and a thick sheet (plane strain). The approximation will be based on linear fracture mechanics formulas and von Mises yield criterion. In addition some simple assumptions are made. A mode I crack is loaded in the Y-direction, while it is growing in the X-direction. It is assumed that the deformations in XY plane are the same for metal and adhesive:



$$\varepsilon_{xm} = \varepsilon_{xa} \quad (1)$$

$$\varepsilon_{ym} = \varepsilon_{ya} \quad (2)$$

(m refers to metal, a to adhesive).

Secondly the stress in the thickness direction (σ_z) is assumed to be equal in the metal and the adhesive:

$$\sigma_{zm} = \sigma_{za} \quad (= \sigma_z) \qquad (3)$$

Finally it will be assumed that the crack tip area is in a quasi plane strain situation. Lateral contractions (in Z-direction) in each individual sheet will not be fully prevented by the weaker adhesive layer. The contraction in each sheet is assumed to occur symmetrically, i.e. the midplane of each sheet will have a zero displacement in the Z-direction. This quasi plane strain situation then leads to the condition:

$$t_m\, \varepsilon_{zm} + t_a\, \varepsilon_{za} = 0 \qquad (4)$$

Equations (1) to (4) and Hooke's law give:

$$\left[\sigma_{xm} - \nu_m\,(\sigma_{ym} + \sigma_z)\right] \Big/ E_m = \left[\sigma_{xa} - \nu_a\,(\sigma_{ya} + \sigma_z)\right] \Big/ E_a \tag{5}$$

$$\left[\sigma_{ym} - \nu_m\,(\sigma_{xm} + \sigma_z)\right] \Big/ E_m = \left[\sigma_{ya} - \nu_a\,(\sigma_{xa} + \sigma_z)\right] \Big/ E_a \tag{6}$$

$$t_m \left[\sigma_z - \nu_m\,(\sigma_{xm} + \sigma_{ym})\right] \Big/ E_m + t_a \left[\sigma_z - \nu_a\,(\sigma_{xa} + \sigma_{ya})\right] \Big/ E_a = 0 \tag{7}$$

Elimination of σ_{xa} and σ_{ya} from these equations leads to:

$$\sigma_z = C\,\nu_m\,(\sigma_{xm} + \sigma_{ym}) = \nu_{eff.}\,(\sigma_{xm} + \sigma_{ym}) \tag{8}$$

with

$$C^{-1} = 1 + \frac{ta\left[\frac{E_m}{E_a}\;\nu_m\,(1+\nu_a)\;\;(1-2\nu_a) - \nu_a\,(1+\nu_m)\;\;(1-2\nu_m)\right]}{t_m\,\nu_m\,(1-\nu_a) + t_a\,\nu_a\,(1-\nu_m)} \tag{9}$$

Since $E_m/E_a \gg 1$ and $\nu_a \sim \nu_m$ it follows from Eq. (9) that $C < 1$ as should be expected, because $C = 1$ would apply if $t_a = 0$ (metal only in plane strain).

Numerical values have to be substituted now, which for the metal are: $E_m = 72000$ MPa, $\nu_m = 0.33$, $t_m = 1.0$ mm. For the adhesive the elastic constants are somewhat uncertain. The most reasonable estimates for the adhesive FM 123/5 seems to be $E_a = 4000$ MPa, $\nu_a = 0.40$ while $t_a = 0.24$ mm. Substitution in (9) gives:

$$C = 0.424 \rightarrow \nu_{eff.} = C\nu_m = 0.140$$

Although this appears to be a fairly low effective Poisson ratio it should be recalled that the lateral contraction in the metal (ε_{zm}) requires a significantly larger ($t_m/t_a \sim 4$ x larger) lateral extension of the adhesive (ε_{za}) which causes a stress in the Z-direction (Eq. 8).

An estimate of the effect in the plastic zone will be derived by adopting the wellknown asymptotic stress field equations

$$\sigma_x = \frac{K}{\sqrt{2\pi r}} \cos\frac{\Theta}{2} \quad (1 - \sin\frac{\Theta}{2}\sin\frac{3\Theta}{2})$$

$$\sigma_y = \frac{K}{\sqrt{2\pi r}} \cos\frac{\Theta}{2} \quad (1 + \sin\frac{\Theta}{2}\sin\frac{3\Theta}{2})$$

$$\tau_{xy} = \frac{K}{\sqrt{2\pi r}} \cos\frac{\Theta}{2} \sin\frac{\Theta}{2}\sin\frac{3\Theta}{2}$$

$$\sigma_z = \nu_{eff.} (\sigma_x + \sigma_y) = \frac{2\nu_{eff.} K}{\sqrt{2\pi r}} \cos\frac{\Theta}{2}$$

with $K = S\sqrt{\pi a}$.

Then the three principal stresses are:

$$S_{1,2} = \frac{K}{\sqrt{2\pi r}} \cos\frac{\Theta}{2} (1 \pm \sin\frac{\Theta}{2}) \qquad (10)$$

$$S_3 = \frac{2\nu_{eff.} K}{\sqrt{2\pi r}} \cos\frac{\Theta}{2} \qquad (11)$$

Substitution of stresses (S_1, S_2, S_3) in the von Mises yield criterion gives the equation for the radius r where these criterion is satisfied:

$$\frac{r}{a} = \frac{1}{2} (\frac{S}{S_{yield}})^2 \cos^2\frac{\Theta}{2} \quad \left[3 \sin^2\frac{\Theta}{2} + (1-2\nu_{eff.})^2\right] \qquad (12)$$

Calculations were made for $\nu_{eff.} = 0.140$ (quasi plane strain), $\nu_{eff.} = 0.33$ (plane strain) and $\nu_{eff.} = 0$ (plane stress). The results for $S = \frac{1}{4} S_{yield}$ have been plotted in Figure 7.

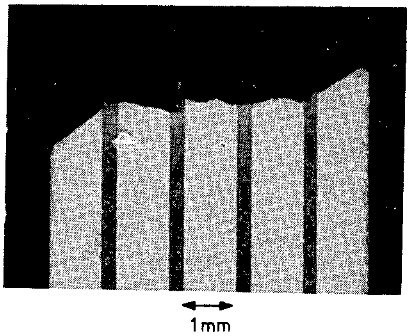


Fig.1: Cross section of fracture surface of laminated specimen. Outer sheets in shear mode, inner sheets in tensile mode (Ref. 1)

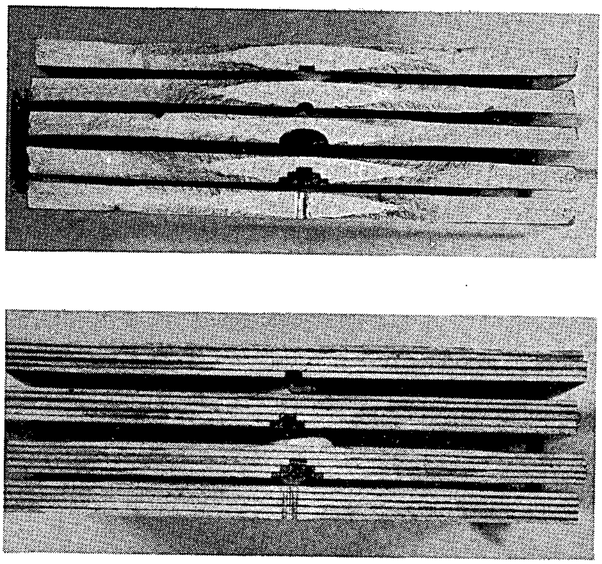
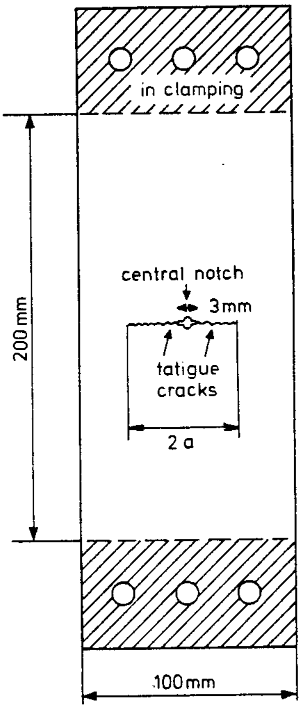


Fig. 2 : Fracture surface of solid and laminated sheet specimens with part through cracks and through cracks (Ref. 2)



Material 2024 - T3 Alclad
Thicknesses : 1 mm
5 mm
5x1mm laminated (Fig. 1)

Central notches : Through crack, width 3 mm
Part through cracks (see Fig 2)
with: width/depth = 3/1 and
9/3 millimeters

Fig. 3 : Crack growth specimen.

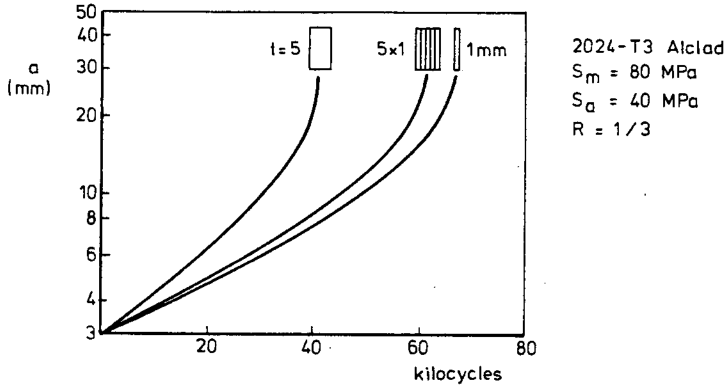


Fig. 4 : Propagation curves in solid and laminated panels with a central crack (Ref.1)

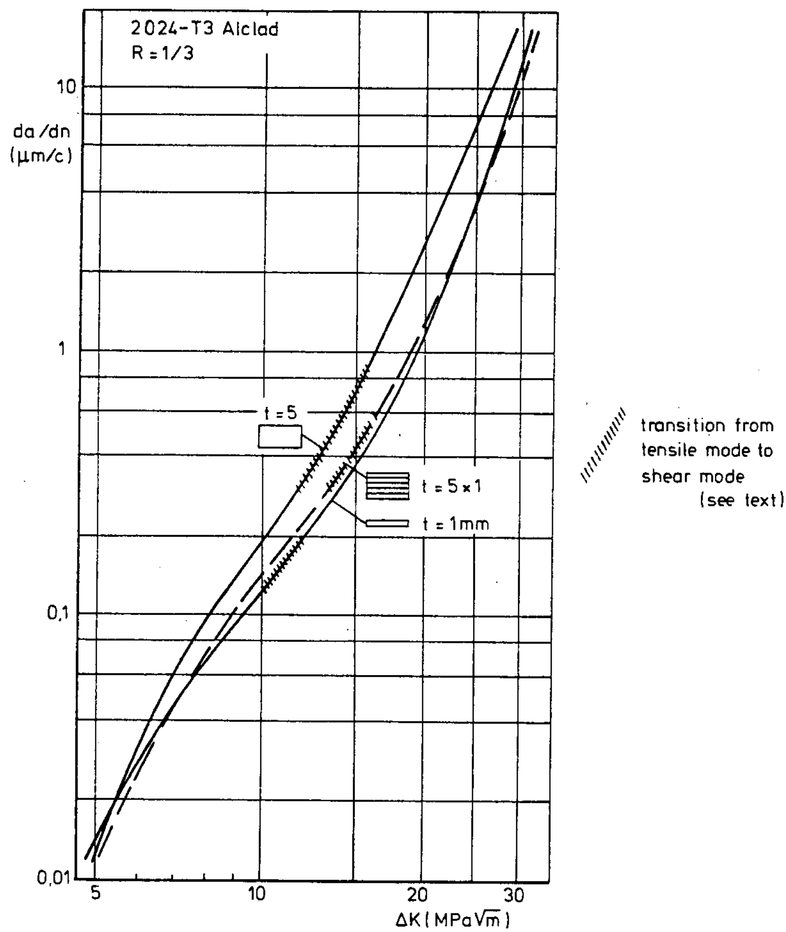


Fig. 5 : Crack growth rates for laminated and solid sheet material (Ref.1)

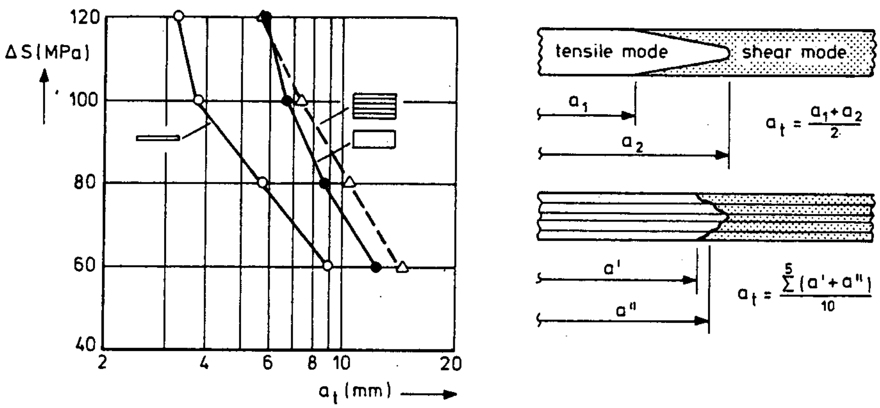


Fig. 6 : The transition from the tensile mode to the shear mode (Ref. 1)

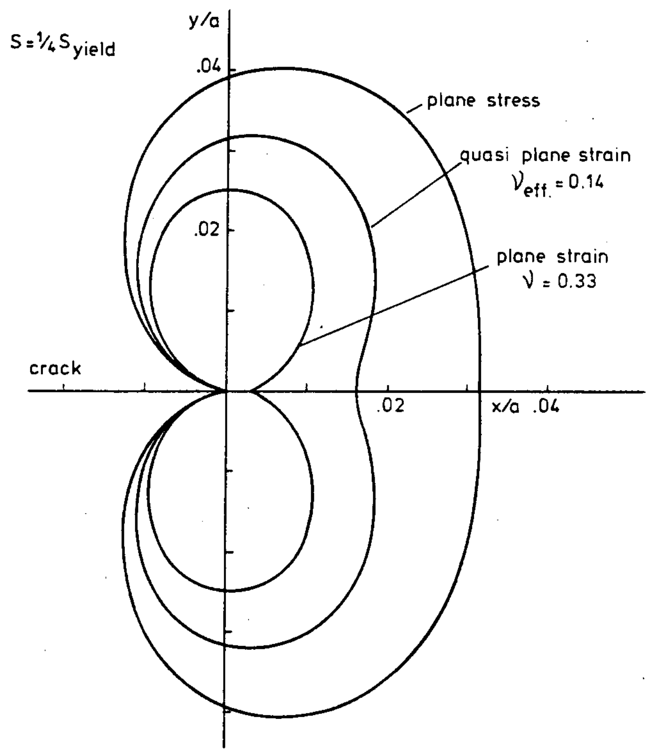


Fig. 7 : Crack tip zones with boundaries satisfying the von Mises criterion (Ref. 4)

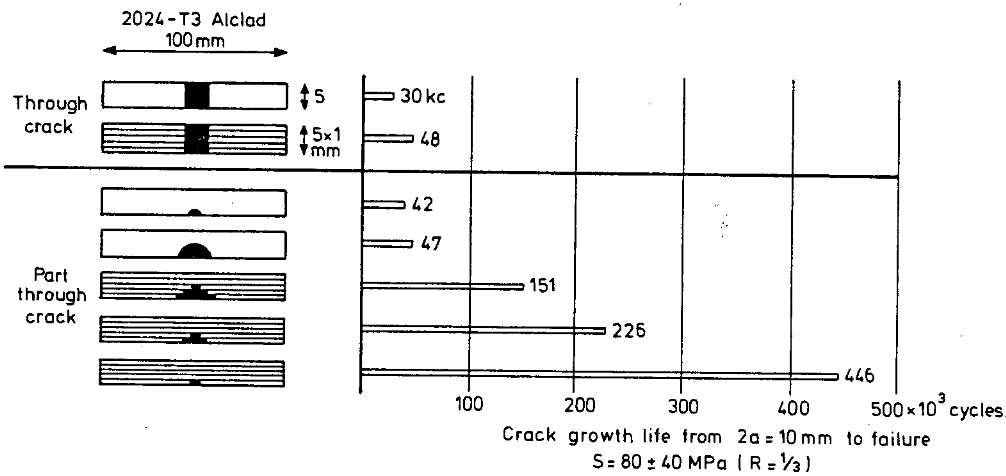


Fig. 8 : Comparison between crack growth lives in solid sheet and laminated sheet material (Ref.2)

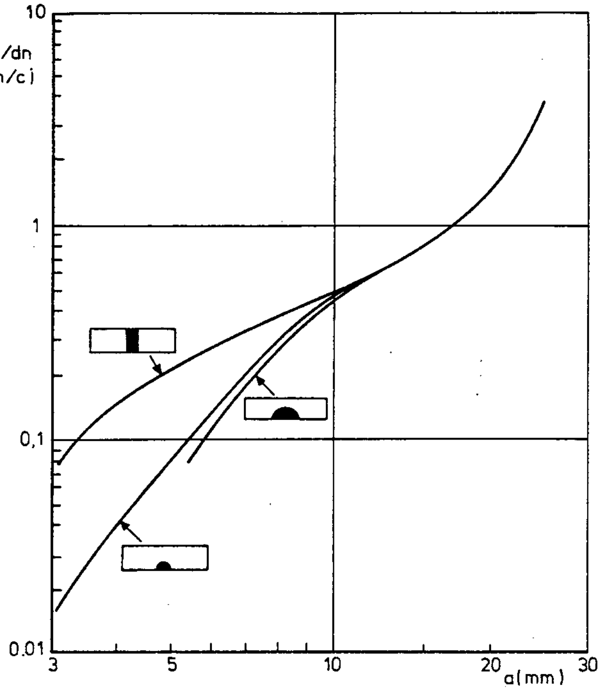


Fig. 9 : Growth rates for through cracks and part through cracks in solid material (Ref.2)

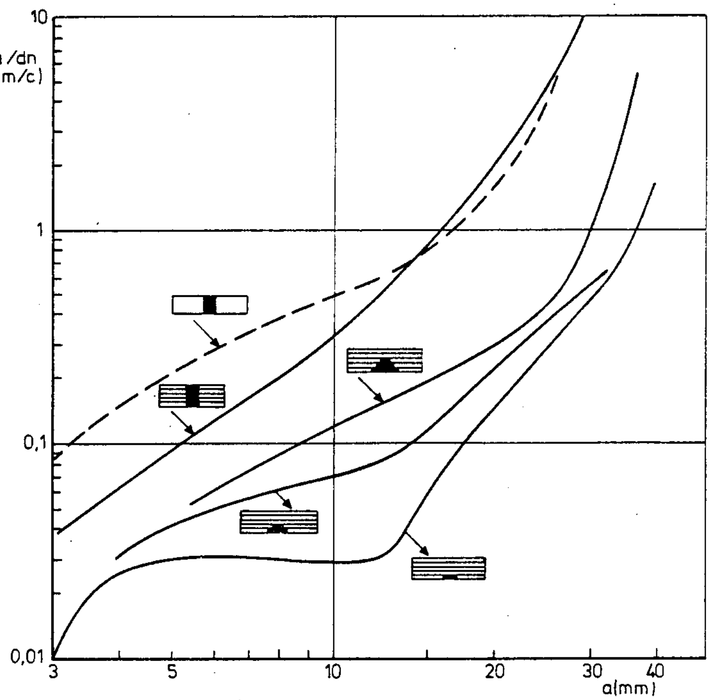


Fig. 10 : Growth rates for through cracks and part through cracks in laminated material (Ref.2)

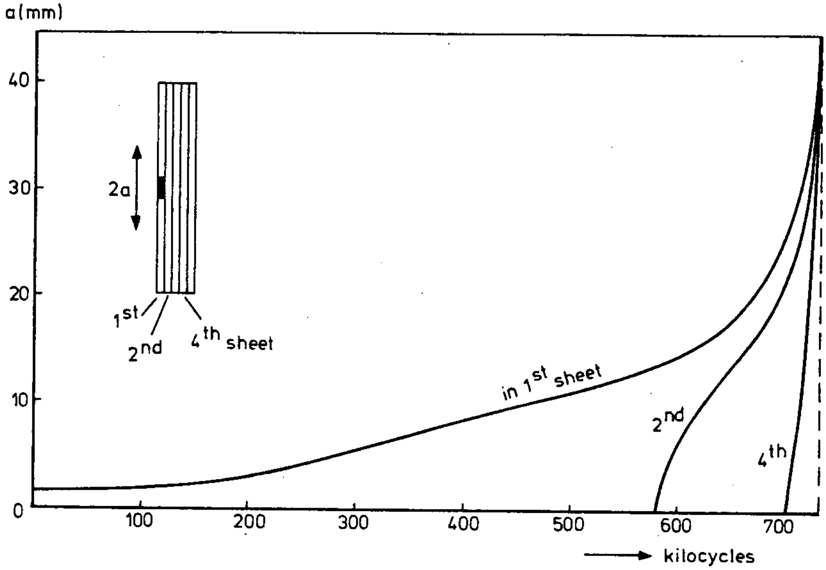


Fig. 11 : Growth of a part through crack in a laminated specimen (Ref.2)

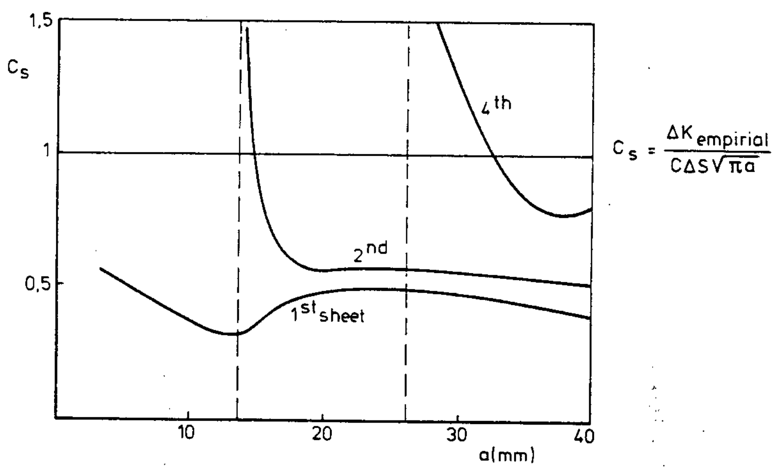


Fig. 12 : Tip stress reduction factors for cracks in Figure 11 (Ref. 2)

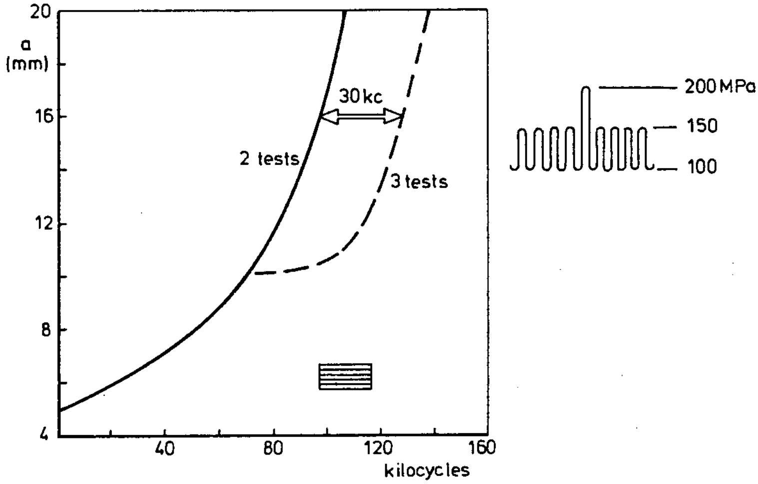


Fig. 13 : Crack growth delay in laminated material.

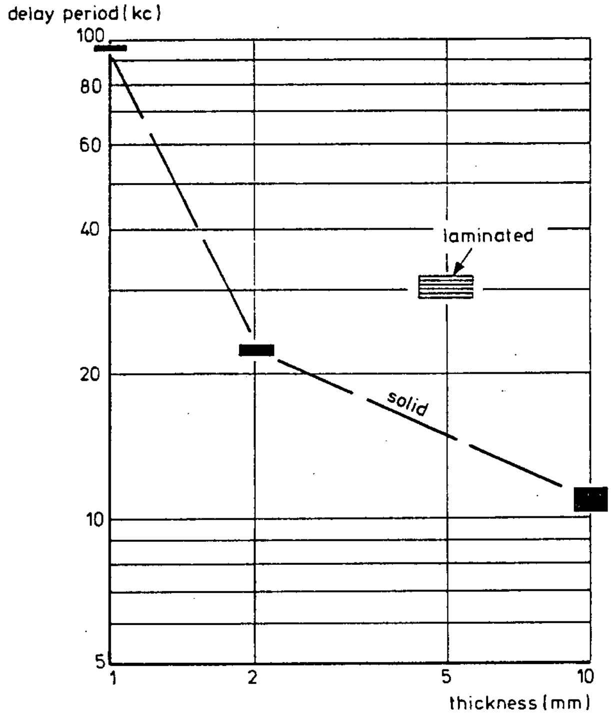


Fig. 14 : Effect of thickness on delays after a peak load (see fig. 13)

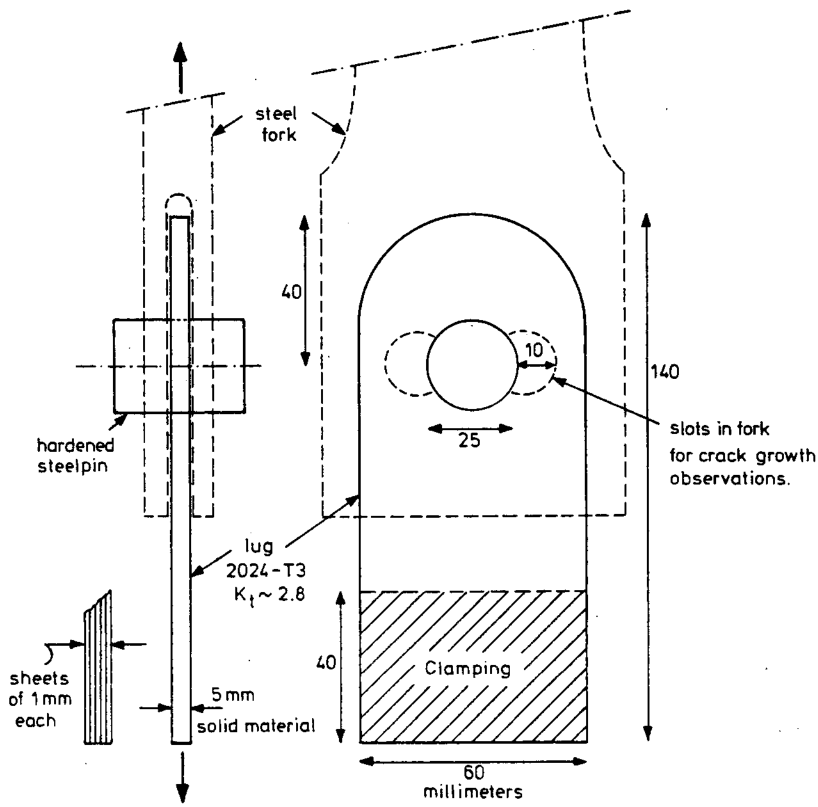


Fig. 15 : Dimensions of lug specimens of solid and laminated material (Ref.3)

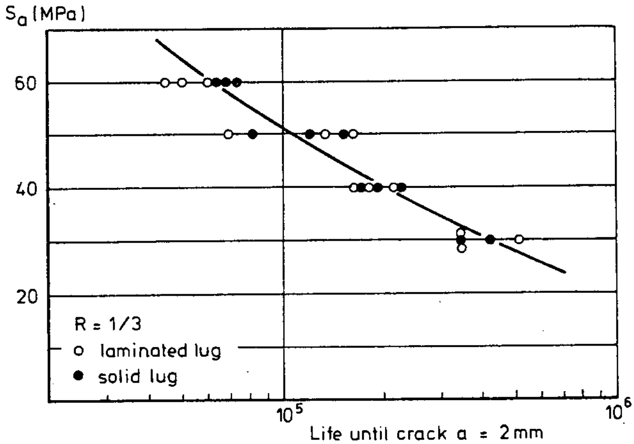


Fig. 16 : Crack initiation lives of laminated and solid lugs (Ref. 3)

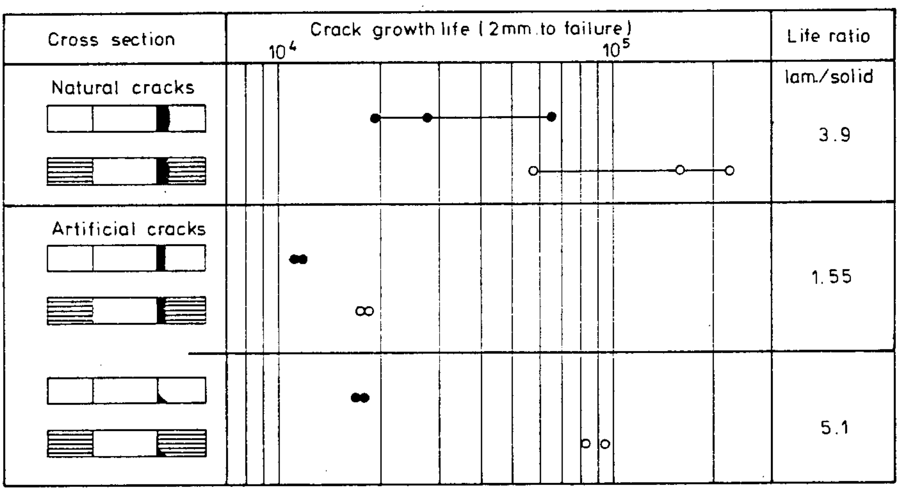


Fig. 17 : Comparison between crack growth lives of solid and laminated lugs, loaded at S_m = 80 MPa and S_a = 40 MPa (R = 1/3). (Ref. 3)

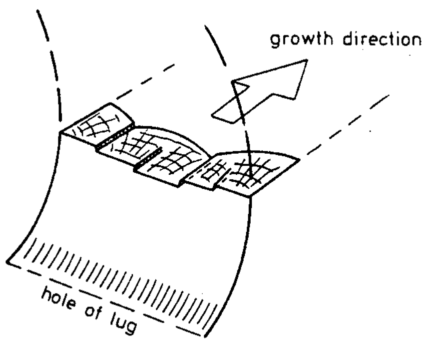


Fig. 18 : Poorly organized crack front due to multiple crack nucleation at slightly staggered levels.

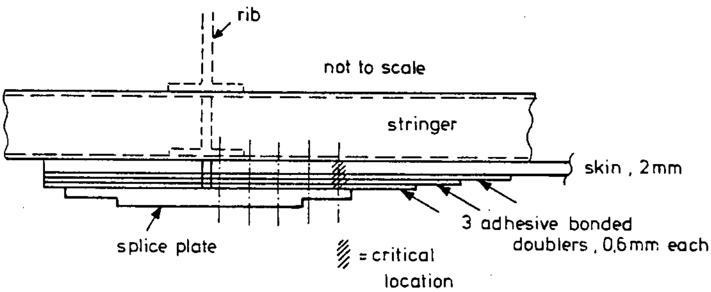


Fig. 19 : Wing joint with adhesive bonded doublers on skin (Ref. 9).

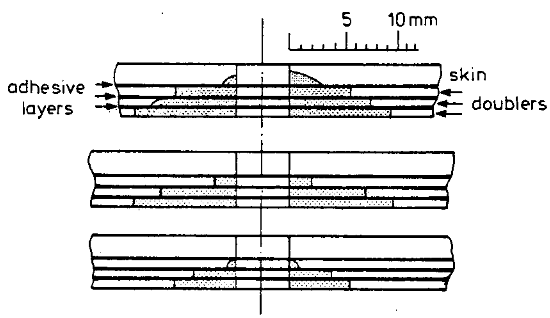


Fig. 20 : Fatigue crack nuclei in the critical section of the joint shown in Fig. 19 (Ref. 9)

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