FATIGUE LIFE UNTIL SMALL CRACKS IN AIRCRAFT STRUCTURES. DURABILITY AND DAMAGE TOLERANCE

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SUMMARY

Crack initiation in notched elements occurs very early in the fatigue life. This is also true for riveted lap joints, an important fatigue critical element of a pressurized fuselage structure. Crack nucleation in a riveted lap joint can occur at different locations, depending on the riveting operation. It can occur at the edge of the rivet hole, at a small distance away from the hole, but still with subsequent crack growth through the hole, and ahead of the hole with a crack no longer passing through the hole. Moreover, crack nucleation can occur in the top row at the countersunk holes (outer sheet) or in the bottom row at the non-countersunk holes. Fractographic evidence is shown. The initial growth of the small cracks occurs as an (invisible) part through crack. As a consequence, predictions on the crack initiation life are problematic. After a through crack is present, the major part of the fatigue life has been consumed. There is still an apparent lack of empirical data on crack growth and residual strength of riveted lap joints, five years after the Aloha accident. Such data are very much necessary for further developments of prediction models. Some test results are presented.

INTRODUCTION

Small fatigue cracks are of considerable interest for aircraft structures in view of a number of practical reasons:

- In general aircraft structures are not designed for an infinite life but for a finite life in view of weight considerations. It implies that small cracks will be initiated in service, hopefully after a sufficiently long service life.
- A practical definition of the fatigue life in service is not the life until failure, but the life until a small fatigue crack can be detected. This life should be called the *crack initiation life*. Obviously, the crack initiation life will depend on possibilities for crack detection and inspection procedures.
- Predictions on the crack initiation life should be most useful as a design tool in view of a sufficient durability of the aircraft structure.

With the last argument we enter a controversial problem setting. In the literature on fatigue prediction problems, one approach is based on the prediction of the stress-strain history at the root of a notch. It is supposed to include such a difficulty as variable-amplitude loading. Since no theory can be accepted without a proof by experimental evidence, a definition of the fatigue life is essential for the comparison between experiment and prediction. Usually this aspect is dealt with in a somewhat loose manner by assuming that predictions indicate some crack initiation life. It is tacitly assumed that the remaining crack growth life of a specimen will be rather short and can be ignored. In other words, the life until specimen failure is supposed to be approximately the crack initiation life. A weakness of several prediction models is that the large effect of material surface conditions on the crack initiation life is disregarded.

A second approach, which appears to be a more practical and safe conception, is based on assuming some small initial defect. The problem is then reduced to crack growth predictions for small cracks, starting from some very small initial defect. Fracture mechanics is supposed to be applicable. Two obvious questions are, what is the size of the initial defect, and what is the present state of the art of fracture mechanics, is it capable to predict the growth of very small cracks? The practical implication of these questions is not always realized. Small changes in the assumptions to be made can have a large quantitative effect on the predicted life. It is well known by now that small cracks can grow at ΔK levels significantly below the $\Delta K_{threshold}$ of large cracks. This observation can even be understood from a physical point of view, but it is of little help to arrive at accurate prediction methods for such small cracks.

In spite of the above observations, the aircraft industry is under pressure to give due attention to the crack initiation life. In the discussions between the industry, the airlines and the airworthiness authorities new life definitions have entered our jargon:

- operational life
- economical life
- crack free life
- threshold inspection

The arguments of the operators are evident. In view of reducing the operational costs, they want to spend very limited efforts on inspections for fatigue cracks. As a consequence, they highly prefer visible inspections, rather than more sophisticated inspection techniques. The aircraft industry can improve the fatigue quality of the aircraft structure. However, it will lead to a more expensive structure or to an increased weight. On the other hand, the aircraft industry wants to simplify the production of the structure to reduce the manufacturing costs, which can imply that stress allowables must be increased, or that more inspections in service are necessary.

In addition to conflicting economic arguments, safety aspects can also lead to controversial design options. In the arena of "durability" (economical aspect) and "damage tolerance" (safety aspect with economical consequences), the airworthiness authorities should take care of safety aspects by developing suitable requirements to be met by both the aircraft industry and the airlines. In the past, safety requirements were changed and extended, when experience and improved understanding showed it to be necessary. It should be admitted that some milestone accidents have highly influenced our philosophy with respect to safety, durability and damage tolerance, see the table below.

	Accidents	Lessons learned
1954	Two Comet aircraft exploded at cruising altitude, due to fatigue cracks and insufficient life.	Full-scale tests are essential to reveal fatigue critical locations and to show sufficient life. Such tests should not be done on a structure used for static tests before.
1969	F-111 wing failure due to a material flaw, slightly extended by fatigue in a material with low fracture toughness	Damage tolerance design is essential. Fracture toughness is important.
1977	Lusaka accident. A 707 lost a stabilizer, due to an unknown fatigue crack.	"Geriatric" aircraft require additional inspection procedures.
1988	Aloha accident. A 737 lost a large part of the fuselage skin due to many fatigue cracks in lap joints.	Multiple-Site-Damage (MSD) in "aging" aircraft can lead to extensive aircraft damage.

Over the past decades our understanding of the fatigue phenomenon in materials and structures has considerably increased. Qualitatively, our knowledge seems to be quite good, but unfortunately that knowledge has also made it clear that quantitative predictions should be expected to have a limited accuracy. That is an important argument for requiring full-scale fatigue testing, which now is part of a recently proposed revision of FAR AC 25.571-1A (Oct. 19, 1993).

With respect to predicting the durability and damage tolerance properties of an aircraft structure, there are three different problems:

1. Prediction of fatigue life until a small crack is present, i.e. the crack initiation life.
2. Prediction of fatigue crack growth following the crack initiation life.
3. Prediction of the reduced strength by a growing fatigue crack in order to know when a critical failure situation can arise.

The problems involved are essentially different for these three categories. For the second and the third one, fatigue crack growth resistance and fracture toughness are essential material properties respectively. They are supposed to depend on the "bulk" behaviour of the material (and the thickness as well). However, crack initiation life is highly depending on the material surface conditions at the potential crack initiation location. The crack initiation resistance is not a bulk property, but rather a surface property. Although the local stress level is a function of the geometry of the structure, there are several most relevant surface conditions, such as: material roughness, fretting corrosion (in joints), surface damage, residual stress in a surface layer, material quality and material defects in the surface layer. Those circumstances do not have such a significant effect on crack growth and residual strength. As a consequence, accurate predictions of the crack initiation life are still subjected to several uncertainties. This is especially true for joints, which unfortunately represent a major fatigue critical location in aging aircraft structures.

With the above general introduction in mind some more detailed attention will now be given to crack initiation and small cracks. More in particular, small cracks in riveted joints and related aspects of fatigue of lap joints in pressurized fuselages will be discussed.

CRACK INITIATION AND SMALL CRACKS

In the early sixties we made observations with binocular microscopes (30x) on small fatigue cracks in unnotched specimens and specimens with a central hole [1]. Tests were carried out under constant-amplitude loading ($R \approx 0$) at various S_{max}-values. The smallest crack that could be detected had a length in the order of 0.1 mm (0.004") (a length corresponding to 300 000 interatomic distances!). All cracks in the 2024-T3 specimens (bare sheet, t = 2 mm) were corner cracks. Results of central-hole specimens are shown in Fig.1. Instead of plotting the crack size ℓ as a function of cycles, ℓ is plotted as a function of the percentage of life until failure. A few observations can be made:

- After some 50% of the life small cracks were found. The results suggest that minute fatigue cracks must have been initiated early in the fatigue life.
- At low stress levels it takes a relatively larger period for cracks to be initiated. It indicates that there is a kind of a threshold stress level for crack initiation. This agrees with the definition of the fatigue limit as being the minimum cyclic stress level required to initiate a crack that will grow to failure.

In 1979 at an ICAF Symposium, Stone and Swift [2] reported on fractographic results of D.Y.Wang [3]. Flight-simulation tests with a severe cargo transport spectrum were carried out on 2024-T3 and 7075-T6 specimens (thickness 6.35 mm) with many open holes (diameter 6.35 mm). In the electron microscope (SEM) the most severe flights could be traced backwards. The crack growth curves clearly suggested immediate microcrack growth at the beginning of the fatigue life, starting with some initial flaw. The flaw was associated with either some kind of a tool marking or an intermetallic inclusion. The size of the flaws (EIFS = Equivalent Initial Flaw Size) was presented as a statistical distribution, see Fig.2. The average size was in the order of 20 μm. It was suggested by Stone and Swift, that the prediction of the complete fatigue life may well be considered as a prediction of crack growth only.

A similar fractographic investigation (optical microscope) was reported in 1983 by Potter and Yee [4], for 7475-T7651 specimens with two 6.35 mm holes with a fastener installed. The specimens were loaded by some kind of a programmed fighter manoeuvre spectrum. Crack growth curves of 37 similarly loaded specimens are shown in fig.3. Microcracks started from inclusions, tool marks and other damage. The initial flaw for the results in Fig.3 corresponds to an average EIFS of about 8 μm.

The present problem of aging aircraft is associated with fatigue of riveted lap joints in pressurized fuselage structures. In view of predictions of crack initiation life and crack growth, it is of great interest to have similar evidence on crack nucleation in riveted lap joints. Unfortunately, at the time of the Aloha accident, the available evidence was rather limited. In an older report (1963) by Walter Schütz [5], three crack growth curves were presented obtained in a large test program on riveted lap joints. They are reproduced in Fig.4.[a] In spite of the limited evidence, the crack growth curves illustrate two important aspects for fatigue of riveted lap joints:

- The crack initiation life is by far the major part of the fatigue life.
- After a crack became visible, the remaining life until specimen failure was relatively short. It implies that visible crack growth to cause ligament failure between the rivets covered a relatively small part of the life.

At the time of the Aloha accident, information on crack initiation in riveted joints was not collected systematically for the purpose of safety considerations. The effect of fatigue cracks on the residual strength of a lap joint was unknown. Actually empirical evidence on these issues is still rather scant more than 5 years after the Aloha accident.

SMALL CRACKS IN RIVETED LAP JOINTS

In my paper at the International Workshop on Structural Integrity of Aging Aircraft I have collected information on the failure modes of riveted joints [6]. Some useful descriptions were available from older reports. Moreover, riveted lap joint specimens tested in our laboratory by students were still available for examination. Pictures of characteristic fracture surfaces are shown in Fig.5.

Fig.5a shows the characteristic appearance of fatigue crack nuclei at both sides of a countersunk rivet. Crack initiation started at the edge of the rivet hole because of the local stress concentration. However, fretting corrosion will have contributed, due to fretting between either the two

[a] For a long time Schütz's data were the only results which I could find in the literature. Most probably, there is more in test reports of the aircraft industry, but unfortunately those are not published.

sheets or between the rivet shank and the wall of the rivet hole. Moreover, secondary bending, which can be quite high at the first rivet row, will also contribute to crack initiation.

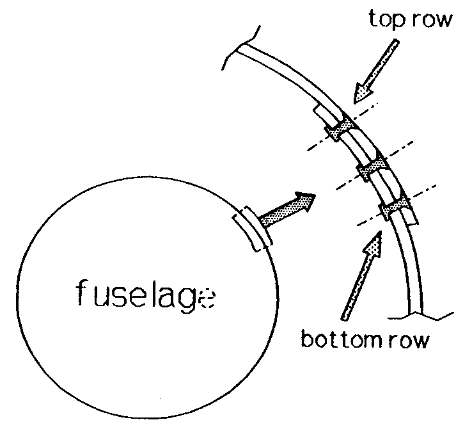
Fig.5b shows crack nucleation in the sheet material away from the rivet hole. This is a consequence of improved clamping, obtained by a more intensive squeezing of the rivets (higher squeeze force). It was recently shown by Slagter [7] that plastic deformation of the rivet and the sheet material, during the riveting operation, can leave residual tensile stress in the circumferential direction around the hole at a short distance of the edge of the hole. This residual stress, the improved filling of the rivet hole and the increased clamping, will remove the critical location from the edge of the hole.

Fig.5c shows a fatigue crack, which does no longer grow through the hole, but instead propagates around the hole This is the result of a still further increased clamping of the rivet. Apparently, the material around the rivet hole, together with the rivet itself, is now acting as a single continuous piece of material. The maximum stress concentration then occurs ahead of the rivet head, where secondary bending and fretting corrosion will cooperate to initiate fatigue crack nuclei. If this type of cracking occurs, more adjacent crack nuclei are created, which usually leads to a rough crack surface when the nuclei overlap, see Fig.5c. Crack growth around the rivet hole was found by Hartman [8] for huckbolt fasteners, which are known for a high clamping force.

Another typical and important feature of fatigue crack nuclei in riveted lap joints is that the cracks start as part-through cracks at the mating surfaces. It takes some time before the cracks have penetrated through the full sheet thickness. It is exactly for that reason that small cracks in riveted joints are invisible for a large part of the fatigue life. After breakthrough of the crack to the outer surface of the sheet, it rapidly becomes a crack with a length that is no longer small as compared to the rivet pitch. That explains why the visible crack growth period in Fig.4 was relatively short.

The significance of part-through cracks has recently been emphasized in test series carried out by Soetikno [9,10]. He interrupted fatigue tests after nx10^5 cycles for a static test until failure (n = 1,2,3,etc.). The purpose of the tests was to measure the decrease of the residual strength as a function of the previous fatigue loading. At the same time, the size and the shape of the fatigue cracks could be determined after the static tests. The fracture surfaces in Fig.6 clearly confirm the original part through character of the crack nuclei. Small crack nuclei could already be observed after 100 kc, which is about 20% of the fatigue life until failure. The initial reduction of the static strength is limited, but when linking up of cracks is imminent, the strength is rapidly decreasing. The same figure also shows residual strength results for a GLARE 3 riveted joint, based on 2024-T3 (3/2 lay-up)[(a)]. In this fiber-metal laminate (thickness 1.4 mm as compared to 1.6 mm for the 2024-T3 specimens; the GLARE specimens were 21% lighter) fatigue cracks for a long time occurred in one of the three layers only, which explains the superior residual strength behaviour.

Another complication of riveted lap joints should be mentioned here. In a lap joint with two or more rows of rivets, there are two fatigue critical rows, the outer rows, see sketch. The upper sheet is fatigue critical in the top row with the countersunk holes. In several tests on riveted lap joints failure occurred in this row. The lower sheet is critical in the bottom row at the non-countersunk holes. Fatigue cracks have been found in the bottom row, both in service and in specimens. Fig.7 shows some small fatigue cracks in the bottom row of a specimen that failed in the top row. Apparently, the bottom row was less critical, but not non-critical. Müller recently observed a tendency for cracking at the bottom row with the non-countersunk holes [13], if riveting is done more intensively (e.g. by a higher squeeze force, which increases the fatigue life considerably).



FATIGUE OF RIVETED LAP JOINTS IN PRESSURIZED FUSELAGES

Fatigue cracks in riveted lap joints of pressurized fuselages have occurred in service, but also in full-scale fatigue tests, including extensive cracking with significant decompression failures. The classical case is the Aloha accident. In view of monitoring crack growth, it is extremely important whether such cracks occur in the top row or in the bottom row. If it occurs in the top row, visual inspections can be sufficient. If it occurs in the bottom row, a visual inspection can only be made during a major overhaul (D-check). Improved riveting can increase the fatigue life, and thus alleviate the MSD problem, but instead of the top row being fatigue critical, it is possible that the bottom row becomes fatigue critical. *The paradoxical question is whether we prefer the longer fatigue life at the cost of an inferior detectability?*

[(a)] GLARE 3 is one type of the new fiber-metal laminates developed for high fatigue resistance [11,12]. The fiber-metal laminates are built up of thin Al-alloy layers with intermediate unidirectional fiber layers in an adhesive as a matrix. GLARE 3 has fiber layers in two perpendicular directions for application as fuselage skin material.

As pointed out in [6], the analysis of fracture surfaces of riveted lap joint specimens very often shows fatigue cracks at many rivets. Examples were already shown in Fig.6. Several other examples are documented in [13,14]. In other words, multiple-site damage (MSD) seems to be a normal pattern for riveted lap joints, if loaded under constant-amplitude fatigue loading (R=0). Actually, this is not a surprising behaviour. If the life is finite, crack initiation at a rivet hole is not a problem, it will occur. A conclusion postulated in [26] is: *If the fatigue life of a riveted lap joint is limited, cracks will occur at many rivets in the same row, and a potential MSD problem is present.* A practical problem is that cracks can remain small and hidden for quite a long time. As a consequence, there will be several small cracks in one row. In our laboratory Müller [13] carried out cyclic pressurization tests in a small barrel test set up (diameter 1200 mm). He also found MSD in a longitudinal lap joint of the biaxially loaded cylindrical specimen. At the ends of the cylinder (length 420 mm) the hoop stress was lower, due to the steel pressure bulkheads. As a result, cracks did not occur at the last few rivets close to the ends of the specimen, but they were found at all other rivet holes.

A similar phenomenon occurs in a pressurized fuselage. Due to pillowing of the skin (Fig.8), the hoop stress distribution in a fuselage is inhomogeneous with a lower hoop stress at the frames and a higher hoop stress between the frames. The inhomogeneity is caused by the restraint on radial expansion of the skin at the connections to the frame. The variation depends on the stiffness of the skin/frame connection. Differences of about 25% are mentioned in [15], but for a low stiffness skin/frame connection it may be lower. Mayville and Warren [16] refer to MSD found in an aging 727 aircraft with multiple cracks in three bays between the frames, but no cracks at rivet holes at the frame location, see the results in Fig.9. Similar crack patterns were predicted by a computer model developed by Broek [17].

Interesting results were published by Goranson and Miller [18,19], Gopinath [20] and Maclin [21] obtained in fatigue tests on two old fuselages of Boeing aircraft, purchased by Boeing from the airlines; viz., a 737 fuselage (59,000 pressure cycles in service) and a 747-100 fuselage (20,000 pressure cycles in service). A new 747-400 forward fuselage (new) was also tested to check redesigned parts of the structure. Small cracks in the lap joints were found in the 737 fuselage after 79,000 flights ($\approx$ 24 years). Linking up of cracks occurred at a later stage, with a decompression failure (flapping, total crack length $\approx$ 0.8 m) at 100,600 flights ($\approx$ 30 years). In the 747-100 cracks in some lap joints occurred after $\approx$ 30,000 flights onwards, with linking up at only one location before the test was stopped at 40,000 flights. In the new 747-400 forward fuselage cracks were detected from 33,500 pressure cycles onwards, with some linking up after an additional 10,000 to 20,000 cycles. By the end of the test (60,000 flights) the largest crack was about 0.48 m with some cracking in an adjacent bay, but without any unstable crack extension. The tests confirmed that the cracks started predominantly at rivets midway between the frames, due to a higher hoop stress (fuselage pillowing). However, the most important finding was that the damage development was rather slow in terms of crack growth per year. It thus should be possible to detect the cracks during scheduled inspections. Probably not everybody will be prepared to generalize this conclusion. There is at least one good reason for some concern. In the Boeing tests the crack occurred in the top row, where crack detection is relatively easy. However, if cracks initiate in the bottom row, the inner sheet is critical. As said before, visual inspection then is rather problematic.

A second matter of concern has been indicated by Swift [22] and Broek [23,24], and it was emphasized again by Swift [25] in the aging aircraft meeting in Hamburg last year. It is the problem that MSD can lead to a significantly reduced crack arrest capability of the structure. Crack arrest is necessary if large incidental damage occurs (rotor burst) or if inspection is extremely poor.

The above references were discussed in some detail in [26]. It appears that some remarkable inconsistencies are entering our philosophy on durability and damage tolerance of pressurized fuselages. In general inhomogeneous stress distributions are considered to be uneconomical, and also in this case it was not the aim of the designer. It was simply unavoidable. However, thanks to the inhomogeneity, fatigue cracks in the lap joints start between the frames, and not all together along the entire longitudinal lap splices. *The paradox is that an inhomogeneous hoop stress distribution will be favourable for safe crack detection, i.e. for damage tolerance.*

Another controversial aspect is offered by the tear straps. High strength and stiff crack stoppers should prevent extensive multi-bay crack extension. However, it is a kind of an "emergency" solution, which should not make us happy. Moreover, effective crack stopper straps will amplify the inhomogeneity of the hoop stress distribution[a]. The production shop anyhow dislikes tear straps because it implies a more complex and costly production. *The paradox is that a structure without tear straps, and with a more homogeneous hoop stress distribution, can lead to an increased durability at the cost of less damage tolerance.* Or the complementary paradox: *The crack stopper bands as a medicine can make the MSD-illness acceptable, if not stimulate it.*

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[a] It is noteworthy that the tear strap designs of several aircraft industries are all different with respect to material (2024-T3, Ti-6Al-4V), the connection to the skin (riveted, bonded), location of the straps (at the frames, between the frames), and the extent of the straps (full circumferential straps, local patches at the longitudinal lap joints, and no straps at all).

SUMMARY AND CONCLUSIONS

In the previous chapters observations on crack initiation and initial crack growth in riveted lap joints were discussed. It has led to the following conclusions:

Observations

1 Crack nucleation can occur at several locations around the rivet hole, at the edge of the hole, and away from the hole. In the latter case the crack initiation life is usually superior. Crack nucleation quite often occurs in the critical row with countersunk holes (top row), but it can also occur in the bottom row at the non-countersunk holes.

2 The occurrence of MSD in riveted lap joints under constant-amplitude loading is a normal cracking pattern, that should be expected, also under biaxial load conditions.

3 Small cracks initially grow as part through cracks, either in some quarter-elliptical shape from the edge of the hole, or in a semi-elliptical shape away from the hole. These cracks are still invisible. Only after penetrating the sheet thickness the cracks can be detected visibly.

4 In a lap joint the top row with countersunk rivets may be more fatigue critical than the bottom row, but the bottom row can also be critical. Even if the top row is critical, the bottom row is not necessarily non-critical.

Predictions

5 Predictions of the crack initiation life on the basis of material properties are not realistic. Fatigue data of riveted joints have to be used.

6 Prediction models for the propagation of through cracks in a riveted lap joint are now developed. Unfortunately, the empirical verification is still very limited due to the lack of crack growth data from riveted lap joints.

7 Prediction on the residual strength of riveted lap joints with MSD is another problem of current interest. Also in this case available evidence is quite limited. Curves as shown in Fig.6 (strength reduction during the fatigue life) are extremely scarce, although such a curve is frequently used in papers on Damage Tolerance Principles, however, without having test results available.

Design aspects

8 An inhomogeneous hoop stress distribution with a maximum between the frames is favourable for damage tolerance (more time for crack detection), but unfavourable for durability (shorter crack initiation life).

9 The nature of tear straps is an emergency design for the case that large cracks occur. The straps can indeed be favourable for the damage tolerance behaviour, while they can also be

favourable to achieve flapping as a final failure mode. They are undesirable from a production point of view.

10 The escape from the controversial issues mentioned in 8 and 9 is to design for fully avoiding the occurrence of MSD, i.e. to design for a sufficiently long crack initiation life, in other words lap joints, which will not become fatigue critical during the operational life of the aircraft.

Recommendations

11 The lack of experimental data on crack growth and residual strength, more than 5 years after the Aloha accident, is surprising. There is indeed a great need for systematic experimental investigations on these issues. For the growth of small cracks some kind of marker loads should be used.

REFERENCES

[1] Schijve,J. and Jacobs,F.A., Fatigue Crack Propagation in Unnotched and Notched Aluminium Specimens. National Aerospace Laboratory, TR-2128, Amsterdam, May 1964.

[2] Stone,M. and Swift,T., Future damage tolerance approach to airworthiness certification. 10th ICAF Symposium, Brussels, 1979, paper 2.9.

[3] Wang,D.Y., A study of small crack growth under transport spectrum loading. AGARD-CP-328, paper 14, 1983. Shorter version in ASTM STP 761, 1982, pp.191-211.

[4] Potter,J.M. and Yee,B.G.W., Use of small crack data to bring about and quantify improvements to aircraft structural integrity. AGARD-CP-328, Paper 4, 1983.

[5] Schütz,W., Zeitfestigkeit einschnittiger Leichtmetall-Nietverbindungen. LBF, Bericht Nr.F-47, 1963.

[6] Schijve,J., Multiple-Site-Damage fatigue of riveted joints. Durability of Metal Aircraft Structures. Proc. of the Int. Workshop Structural Integrity of Aging Airplanes. Atlanta, 31 Mar.-2 April, 1992, pp.2-27. Also: Fac.Aerospace Eng.,Report LR-679, Delft 1992.

[7] Slagter,W.J., Static strength of riveted joints in fibre metal laminates. Doctor thesis, Delft University of Technology, 25 April 1994.

[8] Hartman,A., Jacobs,F.A. and van der Vet,W.J., Constant load amplitude and programme fatigue tests on single lap joints in Clad 2024-T3 and 7075-T6 aluminium alloy with two rows of rivets or huckbolts. Nat.Aerospace Lab. NLR, TN M.2147, 1965.

[9] Soetikno,T.P., Residual strength of the fatigued 3 rows riveted GLARE3 longitudinal joint. Master thesis, Faculty of Aerospace Eng., Delft, May 1992.

[10] Schijve,J., Fatigue of riveted lap joints of GLARE 3. Fac.Aerospace Eng., LR-Doc. b2-93-6, Delft, 6 Dec.1993.

[11] Gunnink,J.W., Vogelesang,L.B. and Schijve,J., Application of a new hybrid material (ARALL) in aircraft structures. Proc. 13th I.C.A.S. Congress, August 1982.

[12] Gunnink,J.W.and Vogelesang,L.B., Metal Laminates - The Advancements in Aircraft Materials. Conf. Advanced Materials, 35th Int. SAMPE Symposium, Anaheim, April 1990

[13] Müller,R., Fatigue crack initiation in riveted lap joints and in pressurized fuselages. SAMPE European Conference, Birmingham, 19-21 October 1993. Report LR-725, Delft Un. of Tech.,

Fac. of Aerospace Eng., 1993.

[14] Wit,G., MSD in fuselage lap joints. Report LR-697, Delft Un. of Tech., Fac. of Aerospace Eng., July 1992.

[15] Molent,L. and Jones,R.: Crack growth and repair of multi-site damage of fuselage lap joints. Eng.Fracture Mech., Vol.44, 1993, pp.627-637.

[16] Mayville,R.A. and Warren,T.J.: A laboratory study of fracture in the presence of lap splice multiple site damage. Structural Integrity of Aging Airplanes, Springer Verlag, Berlin 1990, pp.263-273.

[17] Sampath,S. and Broek,D.: Estimation of requirements of inspection intervals for panels susceptible to multiple site damage. Structural Integrity of Aging Aircraft. Springer Verlag, Berlin, 1991, pp.339-389.

[18] Goranson,U.G. and Miller,M., Aging jet transport structural evaluation programs. Proc. 15th ICAF Symp.,Jerusalem, 1989, EMAS Warley, 1989, pp.319-351.

[19] Goranson,U.G., Continuous airworthiness of aging jet transports. Proc.2nd Int.Conf. Aging Aircraft, Baltimore, Oct.1989. FAA, U.S. Dept. of Transportation, 1990, pp.61-89.1990.

[20] Gopinath,K.V., Structural Airworthiness of Aging Boeing Jet Transports - 747 Fuselage Fatigue Test Program. 1992 Aerospace Design Conference, Feb.3-6, Irvine, 1992. AIAA 92-1128.

[21] Maclin,J.R., Performance of fuselage pressure structure. Paper 3rd Int.Conf. on Aging Aircraft and Structural Airworthiness. Washington,D.C., 19-21 Nov.,1991.

[22] Swift,T.: Damage tolerance capability. Fatigue of Aircraft Materials. Proc. Specialists Conference, Delft, 14-15 Oct.1992. Delft University Press 1992, pp.351-387.

[23] Broek,D.: Fatigue and damage tolerance, the Schijve era and beyond. Fatigue of Aircraft Materials. Proc. Specialists Conference, Delft, 14-15 Oct.1992. Delft University Press 1992, pp.19-41.

[24] Broek,D.: The effects of multiple-site-damage on the arrest capability of aircraft fuselage structures. FractuREsearch TR 9302, June 1993

[25] Swift,T.: Widespread fatigue damage monitoring issues and concerns. Paper 5th Int. Conf. on Structural Airworthiness of New and Aging Aircraft. Hamburg, 16-18 June, 1993

[26] Schijve,J., Multiple-site damage in aircraft fuselage structures. Faculty of Aerospace Engineering, Report LR-729, Delft, July 1993.

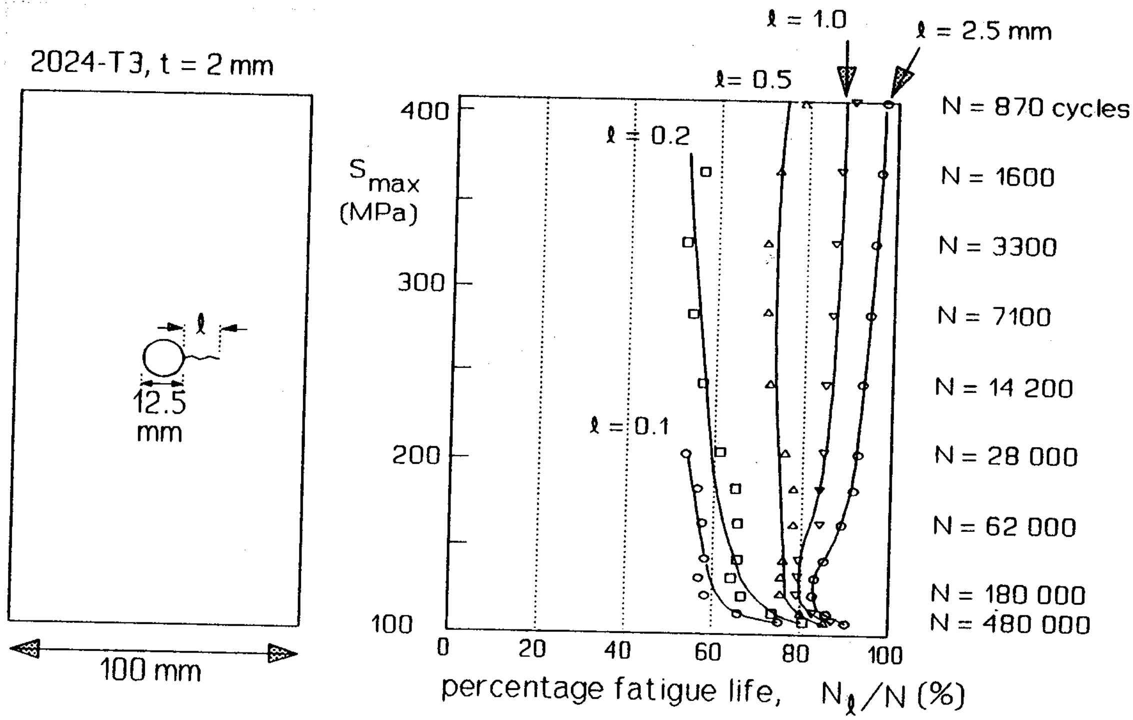

Fig. 1 Small cracks in an open hole specimen [1]. Microscopical surface observations.

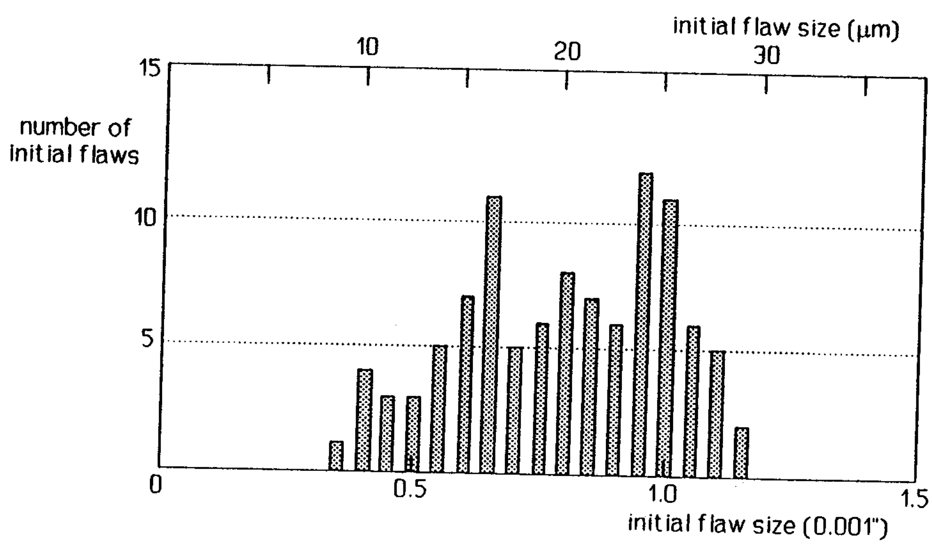

Fig.2 Equivalent initial flaw sizes observed in flight-simulation tested 2024-T3 specimens [2].

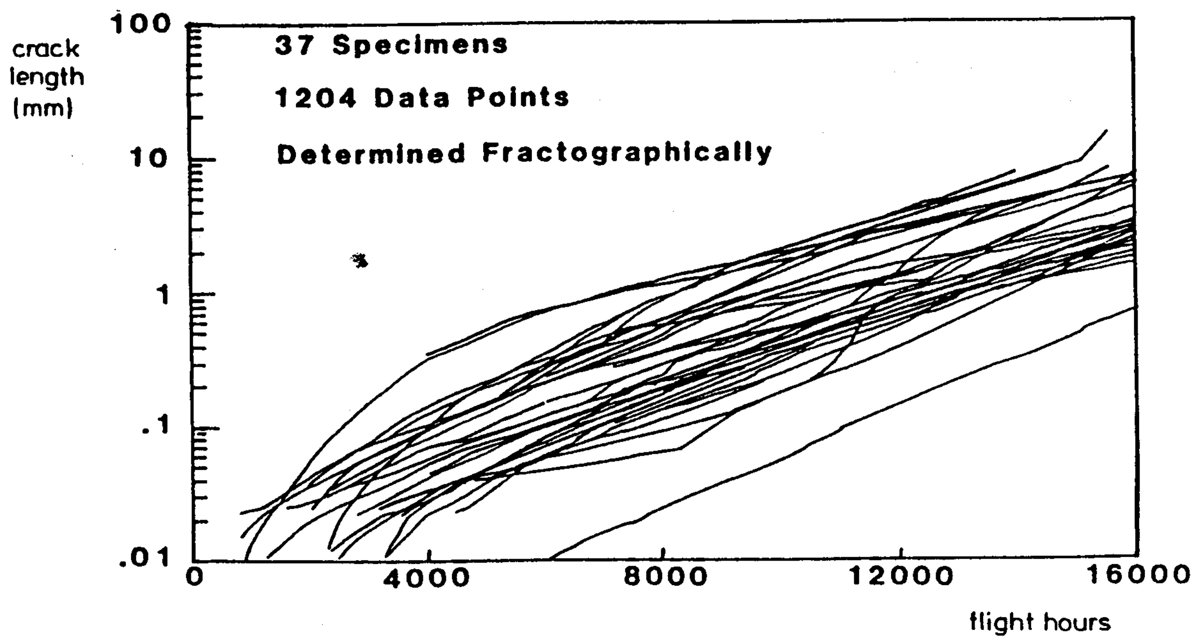

Fig.3 Crack growth curves of small cracks in service simulation test (manouevre spectrum) as derived from fractographic observations [4].

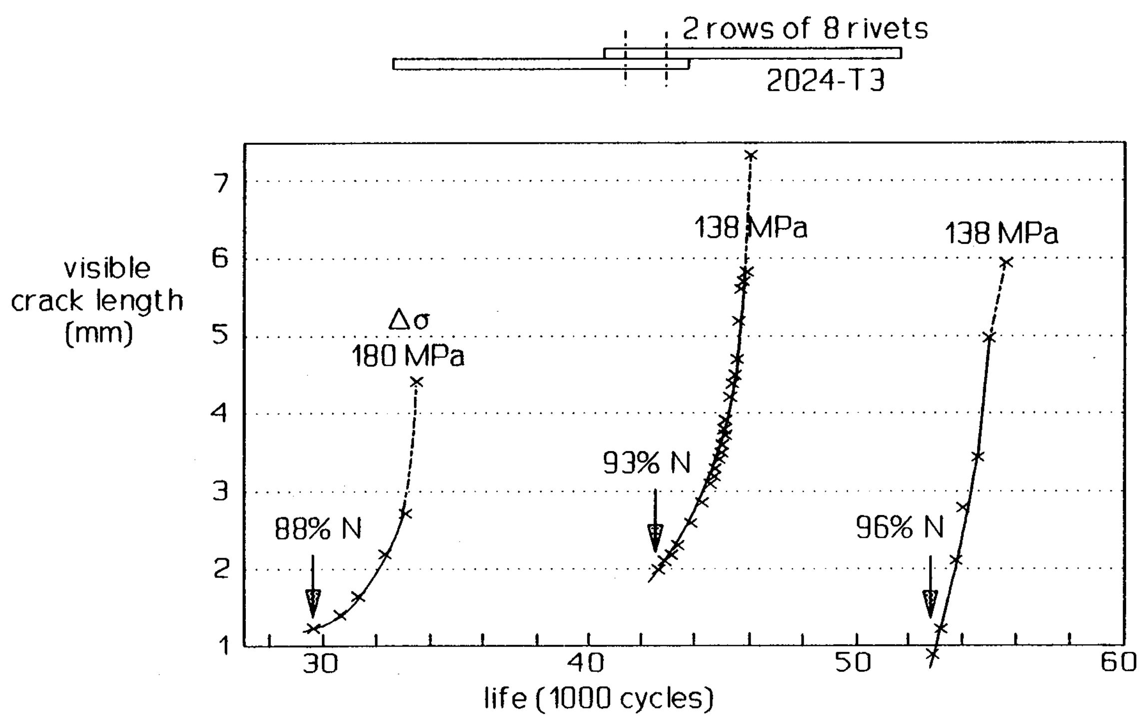

Fig.4 Crack growth curves for riveted lap joints obtained by Schütz [5].

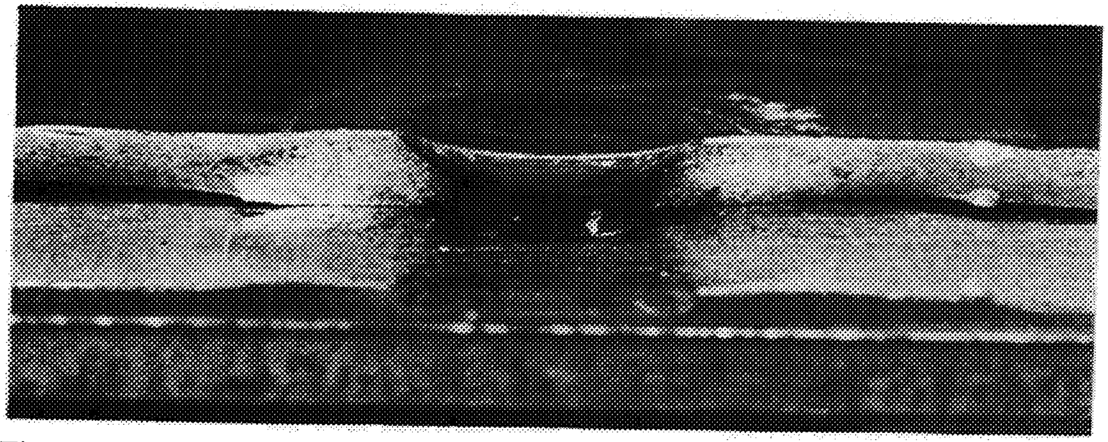

Fig. 5a Cracks at both sides of a countersunk hole.

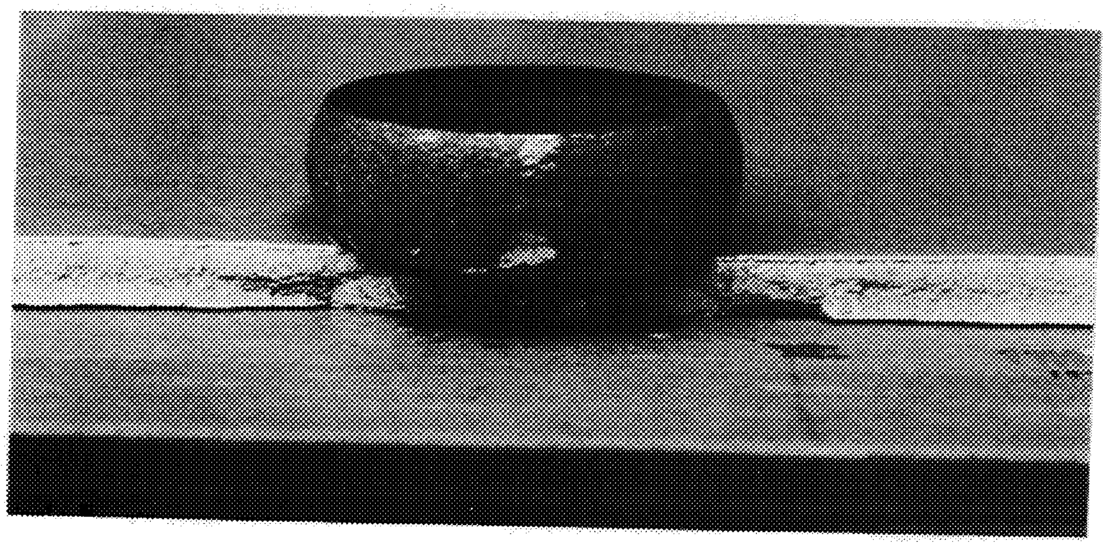

Fig. 5b Semi-elliptical crack nuclei, initiated away from the rivet.

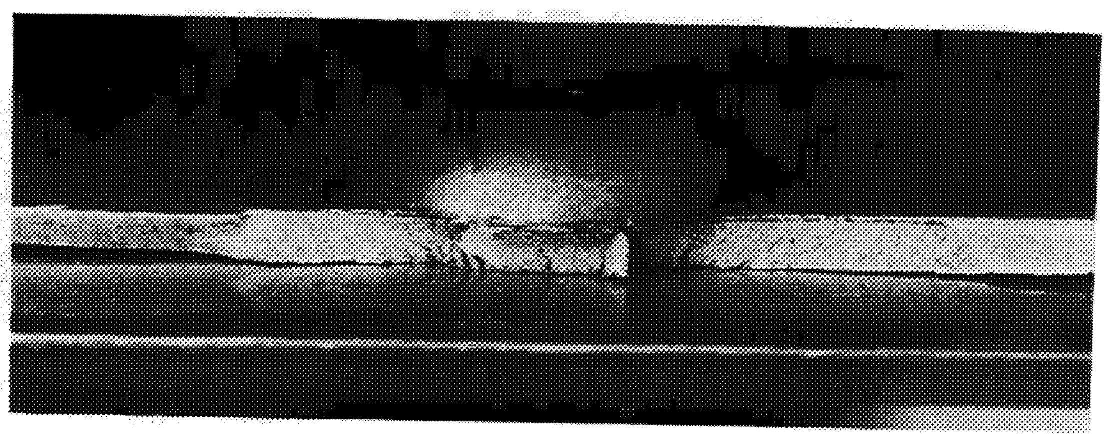

Fig. 5c Cracks started ahead of the rivet. Crack path no longer through the hole (good clamping).

Fig.5 Different types of fatigue crack nuclei in riveted joints [6].

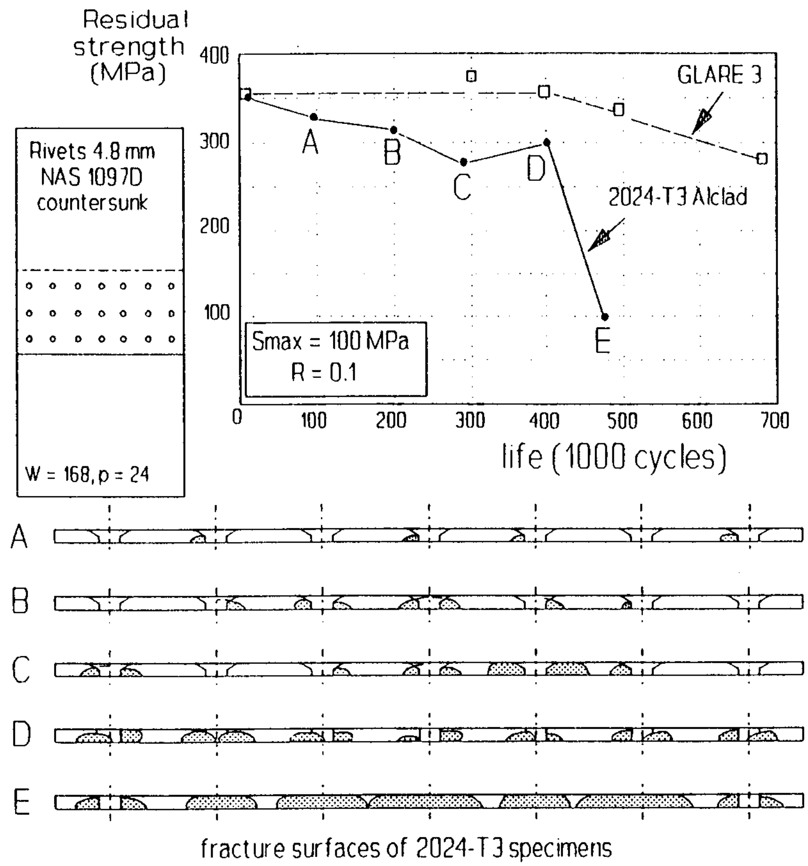

Fig.6 Reduction of static strength of a riveted lap joint due to fatigue. Results for 2024-T3 Alclad (t = 1.6 mm) and GLARE 3 (t = 1.4 mm) based on 2024-T3 [9,10].

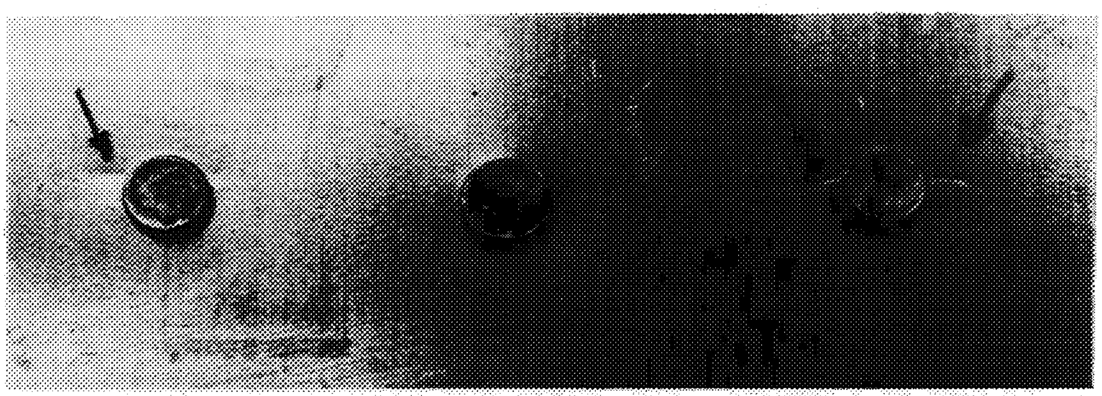

Fig.7 Small cracks at non-countersunk rivet row. These cracks were present after failure occurred at the other critical row with countersunk rivets. (2024-T3 Alclad, t = 1 mm, 3 rows of 7 rivets, d = 3.2 mm, S_{max} = 106 MPa, R = 0, N = 182000 cycles).

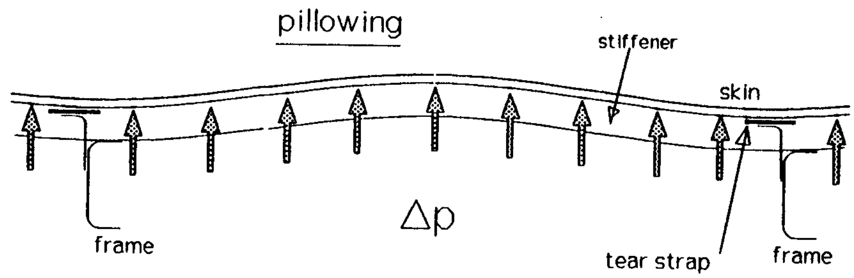

Fig. 8 Pillowing of an aircraft fuselage.

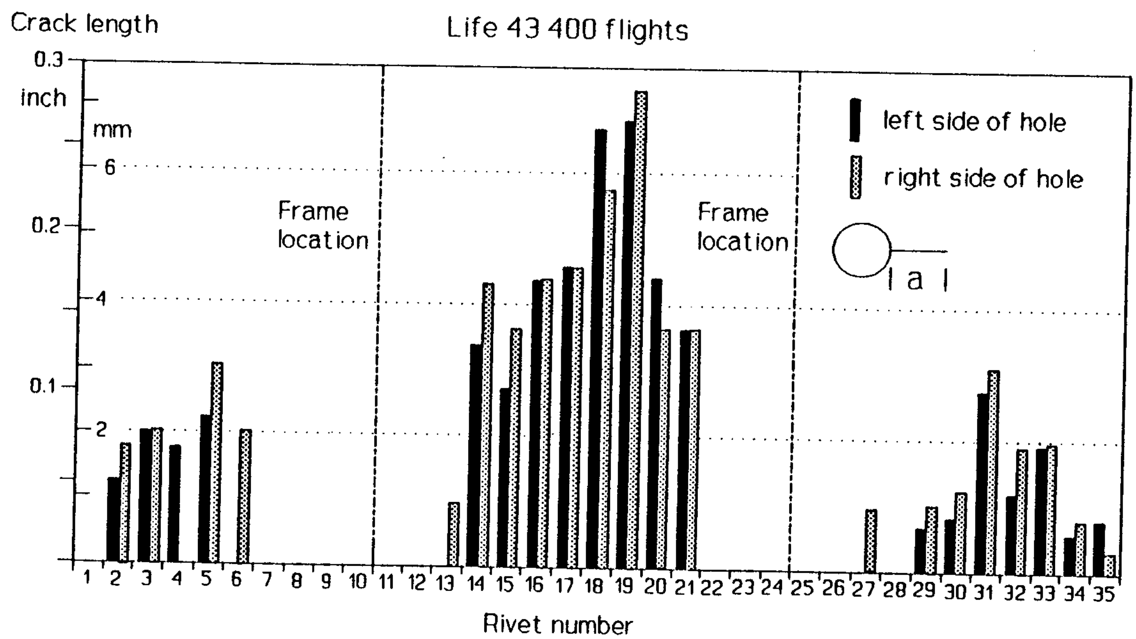

Fig.9 Fatigue cracks in a riveted lap joint of an aging 727 fuselage [14]