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Multiple-site Damage in Aircraft Fuselage Structures

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Symbols and abbreviations

C_L	center line
d	hole diameter
L	specimen length
MSD	Multi-site fatigue damage
p	rivet pitch, or cabin pressure
r_p	plastic zone size
R	fuselage radius
σ_{res}	residual strength
t	specimen thickness
W	specimens width
WFD	Widespread fatigue damage (in the literature also WSFD)

Units

1" = 25.4 mm
1 ksi = 6.90 MPa
1 ksi$\sqrt{in}$ = 1.099 MPa$\sqrt{m}$

MULTIPLE-SITE DAMAGE IN AIRCRAFT FUSELAGE STRUCTURES

Abstract: Since the Aloha accident the MSD problem of riveted lap joints in aircraft fuselages has drawn much attention. The failure scenarios for a lead crack and more small MSD cracks as discussed by Broek and Swift are summarized, including recent results of relevant test series of Broek. It shows that small MSD cracks can significantly reduce the load for unstable crack extension. Prevention of catastrophic consequences requires crack arresting capability of the structure. Related aspects of the problem setting are discussed with reference to failure criteria for ligament failure, the MSD problem for existing and new aircraft, and different options for crack stopper bands. Recommendations for future research are made.

1 INTRODUCTION

After the Aloha accident in April 1988 the significance of MSD and WFD has been generally recognized. It has led to an international working group, which now under the name Airworthiness Assurance Working Group (AAWG) is active in defining the WFD conditions and parameters involved. It also prepares guidelines for maintenance programs [1]. The basic ideas are illustrated by Fig.1 taken from the recent Plantema Memorial Lecture by Goranson [2]. Due to fatigue cracks the residual strength of the structure degrades during the life of the aircraft, and it is doing so more rapidly if WFD occurs. That requires more frequent special inspections.

The Aloha accident also triggered a lot of work on fracture mechanics studies to develop models for the prediction on fatigue crack growth rates of MSD cracking patterns. Two obvious problems are involved:

1. How to model the aircraft structure.
2. How to obtain relevant K-factors.

For fatigue crack growth in a fuselage structure the once per flight pressurization is the predominant fatigue cycle. It might well be hoped that the classical da/dN-ΔK approach could lead to useful predictions. Unfortunately, fatigue cracks in aging aircraft usually starts in the joints, more specifically in the longitudinal riveted lap joints of the fuselage skin[1]. In general the material is low thickness 2024-T3 Alclad sheet (t = 0.8 to 2 mm, 0.032 to 0.080 inch). Fatigue of riveted lap joints implies that the two above questions are just difficult questions because:

1. The load transmission in riveted lap joints is a highly complex phenomenon (pin loaded hole by countersunk rivets, load transmission by friction, fretting, secondary bending, residual stresses, see [3]).
2. The relevance of K-solutions for small cracks in riveted joints is questionable in view of the complex nature or the load transmission and the failure modes.

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[1] Fuselage lap joints can be made by combining riveting and adhesive bonding. If a high quality adhesive bond can be guaranteed, this is by far the superior solution in view of substantially longer fatigue lives. Here we have riveted joints in mind only.

At the moment research programs on both issues are carried out in several laboratories, including effects of biaxiality. Empirical results are very much needed because data on early fatigue crack growth in riveted lap joints are rather scarce. This is equally true for results on the residual strength of the joints if MSD is present.

The MSD problem has also emphasized the significance of reliable inspection techniques for indicating small cracks in riveted joints. If this problem could be solved, taking into account the weakness due to the human factor, the MSD problem can probably be kept under control. Since we may not be convinced that we have reached that situation, attention must be focused on the presence of undetected MSD and its consequence for the damage tolerance of aging aircraft structures. The present paper is primarily a discussion on some aspects of this problem. It was stimulated by discussions during the last ICAF meeting in Stockholm (7-11 June 1993) and the 5th Aging Aircraft Conference in Hamburg (16-18 June 1993), noteworthy by papers of Goranson, Swift and Lincoln [2,4-6] (as well as by previous papers of these authors). Another significant stimulus was a recent report by Broek [7] on the analysis of residual strength tests, carried out by Foster Miller on 2024-T3 Alclad sheet specimens with MSD. It included uniaxial tests on flat sheets and on large curved sheets with crack stopper bands under pressurization cycles. Since the results analyzed by Broek are of paramount significance for statements earlier made by Broek [8] and Swift [9], the present report starts with a survey of Broek's report in chapter 2. It is then followed by discussions on the failure scenario for stiffened skins and on failure criteria (chapters 3 and 4). In chapter 5 comments are made on MSD in a fuselage structure (existing aircraft and new aircraft) and on crack stopper bands and their alternatives. Recommendations for investigations are made in chapter 6. The paper is summarized in a number of conclusions.

2 SUMMARY OF REPORT BY BROEK

Broek recently published a report [7] under the title: "The effects of multi-site-damage on the arrest capability of aircraft fuselage structures". He analyzed the results of an experimental program performed by Foster Miller. Three tests series were carried out to determine the residual strength behaviour of 2024-T3 Alclad sheet panels (t = 0.04" = 1.0 mm).

1. Flat panels, W = 20", L = 40" (508 x 1016 mm) loaded uniaxially.
   3 panels with a single central crack.
   9 panels with a central lead crack and additional small MSD cracks.
2. Unstiffened curved panels (R = 75" = 1905 mm) biaxially loaded by internal pressure.
   3 panels, W = 120" (3000 mm), 2 panels with a single central crack, 1 panel

with a central lead crack and additional MSD cracks.

3. Curved panels with arrester straps, loaded by internal pressure. The panels with a central lead crack and a broken strap were provided with additional MSD cracks.
   4 panels with light straps (width = 2", t = 0.04", 51x1.0 mm) and a small strap spacing (10", 254 mm), strap material 2024-T3 Alclad.
   3 panels with heavy straps (width = 2.5", t = 0.08", 63.5x2.0 mm) and a 2x larger strap spacing (20", 508 mm), strap material 2024-T3 Alclad.

The last part of the cracks was made with a jeweler saw.

Results of the flat panels

The cracking patterns are shown in Fig.2. Because all patterns were symmetric with the same cracks at both sides of the center line (C_L), only half of the cracked cross section is shown. The MSD cracks are in line with the lead crack. Crack edge buckling was prevented only in this test series on flat panels.

The results of the 3 single-crack specimens are presented in Fig.3, together with results of 3 MSD specimens (P5, P8, P11), which did not immediately fail after absorbing the MSD cracks. According to Broek the data are well represented by a net section failure criterion, σ_{net} = 37.5 ksi (259 MPa), which is somewhat below $\sigma_{0.2}$ = 43.65 ksi (301 MPa). Broek prefers to refer to Fig.3 as the net section collapse line.

Absorption of an MSD crack did not always directly lead to complete failure (A in Fig.2). The crack was temporarily arrested, and a further increase of the load was necessary for specimen failure. In other cases absorption of an MSD crack immediately caused final failure (F in Fig.2). It is important to see how the MSD cracks ahead of the lead crack have reduced the residual strength. This is shown by Fig.4. Apparently a significant loss can occur depending on the MSD configuration. The loss is almost 50% for specimen P11 . As shown by Fig.2 this specimen has only a small ligament between the lead crack and the first relatively large MSD crack. A very small increase of the load is then sufficient to pick up the 2nd crack and to cause final failure.

Reference may be made here to a recent report by Moukawsher [10], sent to me by professor Grandt,Jr. of the Purdue University. Moukawsher tested 2024-T3 flat panels (W = 9" = 229 mm, t = 0.09" = 2.3 mm) with a relatively large central lead crack along a series of 10 or 11 holes (d = 0.16" = 4.1 mm). Three configurations were tested, specimens with holes and MSD, specimens with holes without MSD, and specimens without the extra holes. The configurations are presented in Fig.5 together with the residual strength obtained in static tests. The latter results are collected in Fig.6. Although the specimens were relatively narrow with a larger thickness than used by Broek, it confirms that the MSD ahead of a lead crack can significant reduce the residual strength. Even holes alone already have some effect.

Results of the unstiffened curved panels loaded by pressure

Tests were carried out on two panels with a single crack (2a = 11" and 16" respectively, 140 and 203 mm) and one panel with a lead crack (2a = 11" = 140 mm) and two MSD cracks at each side of the lead crack. The crack patterns are shown in Fig.7, together with the test results. The nominal residual strength values in this figure are pretty low. Broek refers to a K-amplification by crack edge bulging.

The panel with MSD cracks showed the first link up at a rather low stress. Linking up of the second one was immediately followed by failure. It occurred at a 14% lower stress level than the second panel (P21). Apparently, the 2nd MSD crack has further reduced the strength.

Results of the curved panels with arrester straps loaded by pressure

The cracking patterns in Fig.8 show 4 panels with light straps and a small strap spacing (10", 254 mm), and 2 panels[2] with heavy straps and a 2x larger strap spacing (20", 508 mm), strap material 2024-T3 Alclad. Detailed information about crack extension and absorption of the MSD by the lead crack are presented in [7]. Here a brief summary should be sufficient.

- In the four panels with light straps (spacing 10") unstable crack extension occurred, which was stopped at the strap.
- MSD cracks just ahead of the lead crack stimulated unstable extension at a lower stress level (compare P25 to P26).
- Strap failure occurred at a lower stress if MSD cracks in the next bay near to the strap were present (compare P24 to P23).
- The two panels with the heavy straps confirm that more MSD cracks will reduce the stress level for unstable crack extension.
- When the crack reached the heavy strap in panel P27 it was arrested, but after unloading (because of a technical problem) and reloading , failure occurred by circumferential tearing along the rivets of the strap (see Fig.13b to be discussed later). In the second panel with heavy straps the unstable crack extension was immediately followed by the same circumferential tearing along the rivets of the strap. Strap failure did not occur in spite of significant MSD cracks just at the other side of the strap.

3 THE STIFFENED SKIN FAILURE PHENOMENON

The residual strength of a stiffened skin structure depends on different possibilities for local failure:

(1) The skin itself can become critical.

(2) A stiffener (or a frame or a crack stopper strap) can become critical, because it has to carry more load due to the crack in the skin.

(3) The same is true for the fasteners (rivets) connecting stiffeners to the skin.

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[2] Results of the 3rd panel with heavy straps are not presented in [7].

These failure modes have been recognized for a long time, see e.g. [11,12]. Seemingly simple analyses were based on displacement compatibility calculations, which can include certain plasticity effects (fastener flexibility, plastic deformation of stiffeners). A usual picture to illustrate the significance of different local failure modes is presented in Fig.9. It shows the failure loads for a skin crack and for a stringer as a function of the crack length. The skin curve goes through a minimum, because if the crack approaches the stiffener, the displacement restraint of the stiffener reduces the crack driving force. If the crack has sufficiently passed the stiffener, the restraint becomes less and the crack driving force becomes smaller again.

As an example Fig.9 shows the crack extension of a crack with an initial (semi) length a_i. It starts with some stable crack extension in the skin until it becomes unstable. It then is arrested at B, because the crack driving force is not large enough. A new piece of stable crack extension occurs until stiffener failure occurs at C. This behaviour is extensively explained in the books by Broek [13,14] including the effects of stiffening ratio, stringer spacing and strength, fastener spacing and flexibility, and the skin material. Actually a third curve should be added to Fig.9 for fastener failure, which for clarity has not been done here.

Figure 9 applies to a panel with a single crack. The picture becomes different if there are more cracks present. The most simple case is a lead crack with additional cracks along the same line. This is the MSD situation discussed by Broek [7,8] and Swift [4,9]. The extra failure mode to be considered is failure of ligaments between the lead crack and adjacent MSD cracks. Figure 10 and the following reasoning is mainly based on a paper by Broek [8].

If the lead crack wants to grow, the skin failure curve will be affected by the presence of the MSD cracks. The skin failure curve must obviously run to the tip of the 1st MSD crack (curve 1). If the 1st MSD crack would not be present the skin failure curve would be reduced to curve 2. As a consequence, as soon as the 1st MSD crack is absorbed by the lead crack, curve 2 is applicable. If that crack is absorbed curve 3 is applicable, etc. An example of a scenario for crack growth in the configuration of Fig.10 is shown in the graph. The lead crack after some stable crack extension becomes unstable at A. Without MSD cracks it would be stopped at A^*. However, the 1st and the 2nd MSD crack are absorbed during the unstable crack extension. Then curve 3 becomes applicable, and the load now is still below that curve. The load must be increased to B after which the crack becomes unstable again. After having absorbed the 3rd MSD crack the load must be increased to C of curve 4 (if stiffener failure would not occur). The crack again becomes unstable, but after having absorbed the 4th MSD crack the load is above the kin failure curve 5. Complete failure will occur. It can be concluded that:

- without MSD: panel failure will be induced by stiffener failure at load level D.
- with MSD cracks but without failure of the stiffener: failure will occur in the skin at the lower load level C.

- with MSD cracks and stiffener failure: the residual strength is further reduced to load level B (failure at F).

Broek [7,8] and Swift [4,9] made calculations essentially based on the above scenario. They concluded that MSD can significantly stimulate unstable crack extension, and that under those circumstances weak crack arresters can lead to a low residual strength, i.e. to a catastrophic failure. It is supported by the test results of Broek summarized in the previous chapter.

4 THE FAILURE CRITERIA

Four different types of failure have been mentioned in the previous chapters.

- skin crack failure
- stiffener failure
- fastener failure
- ligament failure

It can not be questioned whether they can occur and whether a failure scenario according to Fig.10 is possible. However, it may well be asked how accurate the predictions can be made for the failure modes. It can also be asked whether the failure models used are plausible and consistent. Brief comments will be presented.

4.1 *Skin failure*

As said before, the skin material of a pressurized fuselage structure usually is 2024-T3 Alclad sheet. During static loading until failure large plastic zones and stable crack extension must be expected. Both will depend on the geometry of the structure and the type of loading. Significant differences should be expected for flat stiffened panels (wing panel simulation e.g.) and for fuselage structures loaded by internal pressure. In order to get better information than obtained now by simplified models, extensive FEM calculations will be necessary. The relevance of the results still depends heavily on the question, whether the FEM model is a realistic simulation of the structure. Assuming that extensive computer calculations can give us the answers to our questions on deformations and displacements in the structure, the result is still restricted to the "crack driving force". In order to arrive at predictions on stable crack growth under a monotonically increasing load, some characteristic material property must be employed, again in broad terms, the "crack growth resistance" of the sheet material.

A most simple material characteristic is the fracture toughness K_c,but it is of little help for stable crack growth predictions. A more promising characteristic might be the R-curve (static crack growth resistance curve), because it is based on the results of stable crack extension in unstiffened sheets. It still is not clear whether such R-curves can be applicable to stiffened structures. The same possibilities for developing similar plastic zones as in the unstiffened sheet, are not present, due to stiffening elements. Another complication is the curved, biaxially loaded skin of a pressurized fuselage[3]. It should be admitted that we are not really in good shape for accurate predictions on stable crack growth in a fuselage structure, starting from simple material data. Quite a lot of work has still to be done, but as pointed out by Broek [8], the problems should be defined first before expensive efforts are started.

In the opinion of the author a failure criterion for stable crack extension in thin sheet material should be based on the plastic deformations around the crack tip, because I believe that similar to fatigue crack growth, static crack extension is also closely linked up with the local plastic deformation at the crack tip. Some characteristic description of the plastic deformation should be adopted. It should not be a globally defined parameter, but rather a locally defined parameter. Of course it will depend on all relevant geometrical aspects. The "crack tip opening angle" proposed by De Koning [15] may be a useful proposition. It was also successfully used by Newman [16], and recently [17] for the flat MSD panels tested by Broek as discussed in the previous chapter (panels P1-P12, Fig.2). His predictions were promising, but it should be recognized that it is an application on uniaxially loaded flat panels. It is also no longer a simple procedure, since it requires extensive elasto-plastic FE calculations.

4.2 *Stiffener failure and fastener failure*

If stiffeners or straps have a simple geometry and a simple connection to the skin, the prediction of stiffener failure will not be the weakest link of the analysis. Obviously, it should be based on static tests on the stiffeners itself. There is nevertheless a problem to simulate the load transmission from the skin to the stiffener in such tests.

Fastener flexibility and fastener failure strength were determined in empirical programs for the purpose of application to residual strength problems [?]. Although, it also might not be a critical problem, it should be kept in mind that fastener flexibility does not only depend on the type of fastener and its dimensions, but for rivets also on the squeezing force (more generally; the riveting technology). Relevant simulation tests can be done if the significance of the relevant variables is understood.

4.3 *Ligament failure*

The prediction of the ligament failure is a problem for which Swift [18] proposed an interesting failure criterion ("an intuitive link-up criterion"). It is also used by Broek [7]. The criterion implies that a ligament will fail if the sum of the sizes of the two

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[3] It may be noted here that Broek in [7] used his data from curved unstiffened sheets with a single crack, loaded biaxially under pressure, to obtain basic material data for the prediction of MSD configurations in curved sheets without an with strap arrestors. It implies that he wanted to make the extrapolation from basic data to the prediction problem as small as possible. In other words, if we are not capable to solve our complex problems, let us be clever enough to start from realistic information obtained under realistically simulated conditions, rather than trying to start from first principles.

plastic zones of the two crack tips involved becomes equal to the ligament size,see Fig.11.

$$r_{p1} + r_{p2} = lig \tag{1}$$

The approach seems to be logical and comparable to a net section yield criterion. We are familiar with the net section yield criterion for unstiffened sheets with a single crack if the material is a ductile alloy such as 2024-T3. The question is rarely raised why the net section yield criterion is logical in that case. Moreover, there is a difference between a plastic collapse of a ligament and the failure of a sheet with one central crack. If the ligament is plastically yielding, it can do so with a decreasing $\int \sigma_y dx$ integrated over the ligament length. That need not imply that it will immediately fail because load shedding to the uncracked ligaments is possible. Such a load redistribution can hardly be estimated by simple models. FE calculations with a good model could shed some more light on this issue.

It is also remarkable that the plastic zone size equation used both by Swift and Broek is the Irwin equation, uncorrected for the local stress redistribution:

$$r_p = \frac{1}{2\pi}\left(\frac{K}{\sigma_{0.2}}\right)^2 \tag{2}$$

If such a correction is made it leads to a constant $1/\pi$ in the above equation instead of $1/2\pi$ [13], i.e. the plastic zone size is twice as large. Broek was very much aware of the plastic zone size effect on the estimations of the residual strength when using the ligament failure criterion of Eq.(1). In addition to Eq.(2) he used three other r_p equations, viz. the Dugdale equation and two equations developed by himself, which should be improvements if the plastic zone size is large. However, it turned out that using the uncorrected Irwin equation led to reasonably good predictions for his test results discussed in chapter 2. The other equations gave larger plastic zone sizes, and as a result systematic underestimations of the test results. It is suggested here that the original Irwin equation underestimates the plastic zones in the ligament, but a possible overestimation of the residual strength is reduced by the occurrence of some stable crack extension, which reduces the ligament size. Moreover, the ligament failure criterion of Eq.(1) is a simplification as pointed out before. In addition, the 1-dimensional r_p estimates require some yield stress. Broek adopted $S_{0.2}$. Swift [4] quotes $S_{0.1}$ = 51 ksi as a good value for Moukawsher's results, and he used S_{yield} = 50 ksi in his own calculations. There is no objection to use a suitable value in an engineering model, but the choice has still an arbitrary character, unless it can be justified by a generally good fit of predictions to empirical results.

One final comment on the ligament failure criterion is made here. The criterion is essentially based on yielding in the ligament. However, for failure of the skin, a crack driving force criterion is used. In other words, two different criteria are used for failure in the same skin material.

5 Some more comments on MSD in a fuselage structure

The calculations carried out by Broek and Swift are instructive to recognize how small MSD cracks can significantly degrade the residual strength of a fuselage structure. They also showed, at least in a qualitative way, the effect of design parameters. In spite of the limited validity of the simplified models, the scenarios of possible failure sequences are correct. The same is true for the message that good crack arresting properties are essential to avoid critical situations if MSD is present. The configuration of a lead crack with small adjacent MSD cracks should be representative for a fatigue critical fuselage lap joint. A real threat according to Swift [4] is that the small cracks may well be below a realistic detection limit. An unstable crack extension might then occur quite unexpectedly. At the same time, Swift also refers to the effect of MSD cracks on the residual strength capability if discrete source damage does occur, such as a non-contained disintegration of an engine disk (rotor burst). For military aircraft the possibility of battle damage is another source of discrete damage. The conditions are rather different for civil airlines and military air forces. The airlines have considerable economical restraints. They feel obliged to keep inspections for cracks to the lowest acceptable minimum, while grounding a hole fleet, if some fatigue safety issue occurs, is very unpopular. Another difference is that air forces are more ready to do fatigue load monitoring. Here the discussion will be restricted to civil transport aircraft. However, there are two different questions:

- what should be done for existing aircraft ?
- what should be recommended for fuselage structures still to be designed?

5.1 MSD and existing aircraft

Since the Aloha accident several forecasts for the numbers of aging aircraft have been presented. The example shown in table 1 (1989) suggests very large numbers in the future. Whatever the accuracy of such predictions is, it is sure that there will be large numbers of old aircraft flying around. Moreover, there are no arguments to retire an aircraft from service if the airlines by correct maintenance have kept the aircraft in good condition. We thus must realize that:

a. MSD cracks or Widespread Fatigue Damage (WFD) can occur to the old aircraft and unstable crack extension must be considered.
b. Discrete source damage can occur to old aircraft, but how likely is the coalescence of MSD and discrete source damage?

Different approaches to the first problem are:

(i) The fatigue life is large enough to make the occurrence of MSD very unlikely.
(ii) MSD may occur but it will be detected before a critical event will occur.
(iii) MSD may occur, but an unstable crack extension will not be catastrophic. The non-catastrophic nature can be the result of crack arrest or a safe decompression, e.g.by flapping.

Table 1 : Expected numbers of old aircraft [20]

Jet Transport aircraft 20 years old or more				
	Type	1989	1995*	2000*
Airbus	A-300	0	20	125
Boeing	707/720	337	397	435
	727	737	1123	1647
	737	221	419	699
	747	31	265	487
British Aerospace	BAC-111	159	196	211
Fokker	F-28	10	84	144
McDonnell Douglas	DC-8	299	342	342
	DC-9	511	745	894
	DC-10	0	202	328
	MD-80	0	0	7
All others	-	135	186	219
Total	-	2,440	4,102	5,731
* Without further attrition/retirement				

In all three cases a pertinent question is how the safety will be proven.

Ad (i): It should be pointed out that a well designed aircraft may have no MSD problems. It then would be unreasonable to burden that type of aircraft with the same restrictive rules as a more MSD sensitive aircraft type. Nevertheless, the question remains, how it will be proven that an aircraft type will have no (or less) problems. It is not realistic that such a proof can be given by analysis alone. Results of full-scale testing and service experience are essential. Tear down inspections of old aircraft, including opening of critical joints, and/or full-scale fatigue testing of old aircraft are highly recommendable. However, even if there is relevant evidence that significant MSD is highly unlikely, inspections can not fully be eliminated. Without inspections the structure becomes a safe-life item, with statistical aspects to be considered. Nevertheless, if more crack free fatigue life is available, the safety is less depending on the quality of the inspections.

Ad (ii): Results on crack growth in fuselage riveted lap joints were obtained by Boeing in full-scale fatigue tests on an old 737 and an old 747-100, retired from service [19]. MSD did occur, but crack growth was relatively slow. There was ample time for crack detection before link-up of cracks led to a larger lead crack. In other words, crack detection during regular inspections should be possible. The conclusion might even be that there is no catastrophic problem scenario for these cracks if adequate inspection procedures are adhered. Not every body 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 (Fig.12), where crack detection is relatively easy. However, if cracks initiate in the bottom row, the inner sheet is critical. Visual inspection then is rather problematic. Only during major inspections (D-checks) the interior of the aircraft is removed and the bottom row is accessible for visual inspections. During more regular inspections, the eddy current method can be applied to detect cracks from the outside in the second layer. It is rather tedious and the human factor can not be ignored. The magnetic-optical imaging (MOI) technique has been developed recently, and it may improve the situation. At the moment a large evaluation program is conducted by the Sandia National Laboratories [21]. It will be most interesting to see whether this will become a useful option. Until now, there is a general feeling, that we never can fully rely on inspections, the more so if small cracks have to be detected. During the last ICAF meeting (Stockholm June,1993) and the Aging Aircraft Meeting in Hamburg (also June,1993) it was clear that interesting developments of sophisticated NDI methods are going on, although they are not yet operative. At the same time the airlines expressed their doubts about new techniques. Apparently, the airlines are strong believers of the usefulness of visual inspections. They complain about neglecting visual inspection methods in the present research efforts.

Statistically, there are more possibilities for increasing the probability of detection. A fleet leader analysis could be helpful, i.e. selective inspections of older aircraft as discussed by Goranson [2]. In addition, inspections of selected areas may also be useful. The hoop stress distribution is inhomogeneous along the longitudinal lap joints. The stress is higher between the frames, and the fatigue cracks generally start at those locations [22,23]. Moreover, the nominal hoop stress is not constant over the whole fuselage. Selected locations for inspections can be indicated, based on stress analysis of the structure, and preferably on a full-scale test survey. Obviously, this kind of improvements of crack detection depends very much on the airlines and the aircraft industries and their cooperation.

Ad(iii): In the third approach MSD may occur, but unstable crack extension will not be catastrophic. This appears to offer more confidence. However, for existing aircraft we can do something about inspection procedures only, while unfortunately crack stopper elements of the fuselage have to be accepted as they were built into the structure. It is noteworthy to see the different solutions adopted by different aircraft industries for crack stopper bands: different materials (2024-T3, Ti-6-4), different connections to the skin (riveting, bonding), different locations (at the frames, midway between the frames), local stopper (at the riveted joint only), or no crack stoppers at all.

The question now is : have we arrived in a critical situation with aging aircraft, knowing that the number of aging aircraft will increase[1]. As said before, two different situations have to be envisaged:

1. Increasing MSD and WFD, which leads to unstable crack extension.
2. Rotor burst damage to an aircraft with undetected MSD and WFD.

The first situation occurred in the Aloha accident. Such accidents can be prevented

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[1] Aging aircraft of airlines, which do not adhere to good maintenance and inspections will always be in danger. Although this problem should be considered by airworthiness authorities, this aspect will not be discussed here.

only if adequate inspections are carried out, and when preventive repairs are made on suspect joints, such as done now to the 737. The second situation can be more serious, but the probability of occurrence will be low. It should be recognized that rotor bursts during take off and landing are not part of the discussion here, since the fuselage then is still unpressurized. Secondly, the rotor burst damage will be limited to that part of the fuselage, where the non-contained disintegrating parts can hit the structure. The local damage can be severe. It appears that also here the risk should be minimized by concentrating efforts on MSD detection at an early stage by selective inspection scenarios as discussed before. Of course, stringent inspection requirements for jet engines are also part of the problem.

5.2 *MSD and new fuselage structures*

In view of the MSD problem two philosophies can be adopted:

I. Design for a sufficiently long fatigue life, free from MSD cracking problems.

II. Design for improved damage tolerance.

Nowadays the useful economical service life of new civil transport aircraft will be relatively high, say 50,000 to 100,000 flights and beyond. In implies that careful design procedures must be used. At the same time, there is a constant pressure on designing for cheap production and a low weight. It easily leads to well known conflicting arguments related to economy and safety. Longer fatigues lives can be obtained in various ways, such as:

- a lower design stress level (more weight)
- a more fatigue resistant material (usually more expensive)
- improved detail design (usually more expensive production)
- improved fastener systems (more expensive).

As long as the skin of a pressurized fuselage will be made of sheet material, potential alternatives for the present 2024-T3 Alclad are other Al-alloys, such as the Al-Li alloys, the 6013 alloy and the new C188 alloy. These alloys are attractive for one common reason: sheets of these new materials can simply replace the good old 2024-T3 material. There is no need for different design concepts, and the same tooling can be used. The investments for the introduction of the new materials would be relatively low. The Al-Li alloys are even advertised as 2024-T3 or 7075-T6 replacement alloys. Of course all properties of new alloys, including the corrosion resistance, should be equivalent or superior to those of the classical Al-alloys. That explains why a large scale application has not yet been achieved.

Similar economical arguments also affect the introduction of improved detail design. The aircraft industries prefer to develop "families" of aircraft types, which can take a large number of years. Detail design characteristics adopted many years ago for the older members of the family are again applied without significant modifications to later ones. It can be motivated by past experience, and also by avoiding the risk and the costs of new concepts. Unfortunately, essentially new improvements are not stimulated in this way, which is considered as a frustrating condition for developments in a high-tech industry.

Another illustration of possibly conflicting arguments was recently encountered in our laboratory. It was shown before [3] that increasing the squeezing load during riveting has a large favourable effect on the fatigue life of lap joints. In recent experiments of Richard Müller (not yet published) he observed that doubling the squeezing load increased the fatigue life of riveted lap joints approximately 3 times! If the increased squeezing force is applied in production, we have to reconsider the quality control procedures. It probably will increase the cost of production, but there can be another economic profit. However, the experiments have led to another remarkable observation. If the normal squeezing force was applied, cracks originated in the countersunk holes of the critical top row in Fig.12. In view of the larger and different stress concentration of the countersunk hole, this is what we expect. Unfortunately, for the increased squeezing force the crack initiation site moved to the non-countersunk bottom row, which is difficult to inspect in a fuselage. Consider the question: "is a threefold increased life desirable if the crack detection becomes relatively more difficult?".

5.3 *Crack stopper bands and alternatives*

As said before, 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 related to 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 [23], but for a low stiffness skin/frame connection it may be lower. There is a remarkable inconsistency in our appreciation of the inhomogeneous stress distribution. 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. It greatly favours possibilities to detect fatigue cracks before there is a general cracking pattern at all rivets.

Crack stopper bands will have a small effect on the hoop stress variation. They are supposed to arrest growing fatigue cracks by the restraint on crack opening, which implies a ΔK reduction. The bands should also contain unstable crack

Table 2: Material comparison.

	specific mass	S_{ult}		E	
	g/cm^3	MPa	ksi	MPa	ksi
2024-T3	2.77	445	66	72000	10400
Ti-6-4	4.46	1070	155	117000	17000
Arall 3 (3/2)	2.33	821	119	68000	9900
Glare 2 (3/2)	2.52	1214	176	65000	9600

extension and thus increase the residual strength to a sufficient level. For the latter purpose bands with a high strength and a high stiffness have advantages. Materials to be considered are 2024-T3, Ti-6-4 and fiber metal laminates (Arall and Glare). In a preliminary investigation [24] crack stoppers of 2024-T3, 7075-T6, Ti-6-4 and Arall and integrally machined bands were compared under fatigue loading. The best behaviour was shown by bonded crack stopper bands of Ti-6-4 and Arall. For riveted bands the Arall strips were better than the Ti-6-4 bands in view of crack initiation in the bands. The fiber-metal laminates are then superior due to their high crack growth resistance. The combination of Ti-6-4 bands with a 2024-T3 skin has disadvantages in the production, while corrosion problems should also be considered. Until now tests with Glare crack stopper bands have not yet been carried out. They should be preferred to Arall bands in view of the significantly higher strength, see the table 2. The stiffness of the fiber-metal laminates is lower, but since the specific mass is also lower, a larger cross section can be used for the same weight. It expected that Glare could be an extremely good crack stopper material.

The question of crack stopper bands with a larger cross section (heavier) is a design option. The test results of Broek in chapter 2 indicate that the heavier bands did not fail. In this particular case the cross section of the bands was 2.5"x 0.08" (63.5x2.0 mm²) compared to a cross section of the skin between the frames of 20"x0.04" (508x1.0 mm²). The cross section of the bands thus is 25% of the cross section of the skin. If the material of the bands had been used to increase the skin thickness, the hoop stress would be reduced by a factor of 1.25. The life increase factor might be approximately $1.25^4 = 2.4$! An additional advantage is that production without the extra bands is significantly more simple in the production. Actually, the solution without bands is more elegant, since the crack stoppers are in fact an emergency solution. However, omitting the bands implies that we rely very much on crack detection. It can be justified only if either a lower stress level or a more fatigue resistant material is adopted. It is expected that the latter solution can be achieved with Glare 3 as a skin material. This application is now experimentally studied in barrel tests.

There is still another solution, which has been labelled as "flapping". If an unstable one-bay flapping crack occurs, see Fig.13a, a safe decompression might be possible. It appears that a thin skin will flap more easily than a thick one [25]. It anyhow is a valid question whether a structure can be designed to stimulate flapping if a growing midbay crack is present. It may well be expected that a locally high circumferential stiffness at the frames will favour flapping. Broek obtained a noteworthy result in the tests with the heavy straps. When the (unstable) crack reached the strap, the crack path continued along the circumferential rivet row of the strap, see Fig. 13b. It is also a kind of flapping where the crack followed a less resistant crack growth path. It might be questioned whether we should design to obtain a flapping fracture mode. Perhaps it is not an elegant solution, but it could be useful. At the moment deviating crack growth directions leading to flapping in fuselage structures are studied [26-28], which could lead to a better background for designers.

6 Research on MSD

Because many aircraft become old, continued research on the aging problem can well be justified. Recommendations for research on some topics are presented below.

Barrel tests

The understanding of possible failure scenarios is essential to plan any useful research programs. Although the present understanding is rather qualitative in nature[1], it is obvious that the residual strength of a fuselage is a complex phenomenon. As a result, quantitative predictions are difficult. The qualitative message of Fig.1 is clear, but it is most remarkable that so little experimental results are available about the deterioration of the residual strength of riveted lap joints. Some recent results of Roebroeks [29] are presented in Fig.14 in a comparison between riveted lap joints of 2024-T3 and Glare 3 (3/2). Apart from the fact that the fiber-metal laminate was highly superior to the Al alloy, the figure shows that the strength of the 2024-T3 lap joint after 400,000 cycles started to decrease rapidly. Fractographic observations showed that it was associated with MSD at most rivet holes. There is indeed a need for this type of data, obtained on larger specimens under realistic conditions. That should include testing of curved panels under biaxial loading. The need for barrel test equipment is evident, and several test set-ups have been made (e.g.[25,30-32]).

It must be emphasized that results of residual strength tests should be well documented with respect to stable crack extension, crack path, fracture mode (stable crack extension in thin sheet does not occur in the tensile mode), crack opening and out-of-plane deformation (bulging), both near the crack tip and along the crack edges. Such information for cracks in fuselage lap joints is essential as a description of the real life phenomenon. If that is not available, there is no relevant guidance for theoretical analysis[2]. Unfortunately, our picture of the failure phenomenon is insufficiently complete on essential details. It can well be extended in barrel tests in order to get a better idea about the crack driving mechanism.

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[1] Understanding per definition should always start in a qualitative way.

[2] Quoting Broek [8] "we will have to tell them what we want, to avoid getting what we don't want".

Fracture criteria

Research on fracture criteria is justified only if it is supported by experiments. As a first start, the experiments can be carried out on flat uniaxially loaded specimens, but the specimens should be relatively large to avoid edge effects. In a recent study in our laboratory [33] MSD was obtained at 16 of the 20 rivets of the top row in 500 mm (20") wide specimens. The edge effect (i.e. crack initiation predominantly at the edge rivets) was avoided by a simple clamping device on the edges to reduce the secondary bending at the edges.

Observations on the failure phenomenon are again essential for the study of failure criteria. Fatigue crack initiation, crack growth and residual strength should be studied in experiments on realistic riveted lap joints. Starting with artificial crack starter notches should be avoided in view of the complex nature of crack initiation. For residual strength tests it might be justified to start from artificial cracks, provided that it is understood why it can be justified.

If the picture is sufficiently complete, FE calculations can be used. Complex FE models will be necessary to simulate the crack growth behaviour. The calculations must lead to a characterization of the crack driving force. Failure criteria (crack growth resistance) can then be considered. As pointed out before, a local deformation parameter is recommended for residual strength predictions.

The experiments should be extended to biaxial loading and loading under internal pressure.

Fatigue of riveted lap joints

It is almost embarrassing to recognize that the fatigue behaviour of a simple riveted lap joint is quantitatively poorly documented. Apparently we have been living with S-N type information for a long time. Because aircraft now become old, S-N data are no longer sufficient to deal with the MSD problem. The number of variables of a riveted lap joint, including the production variables, is quite large. Useful research can be done in this area, in the first place experiments with detailed observations on what happens in the joint.

NDI

As said before, the MSD problem in service might be kept under control if reliable NDI procedures are available. Hopefully, the Sandia program [21] will give useful information. Statistical methods on the probability of detection [2,34,35] should be adopted to analyze the potential usefulness of NDI. It is believed that an adequate NDI system, based on selective inspection procedures as discussed before, can largely solve the MSD problem, provided that information about cracks in the fuselage structure obtained in full-scale tests and in service is available.

7 Some summarizing conclusions

The conclusions are related to MSD problems in riveted joints of civil transport aircraft fuselages.

1. Since the Aloha accident the MSD problem of riveted lap joints in aircraft fuselages has drawn much attention. The problem is complex for a number of reasons: (i) The load transmission in riveted lap joints is a complex phenomenon. (ii) In view of this complexity, the relevance of available K-solutions is questionable. (iii) The failure criteria for residual strength predictions are not well developed.
2. The failure scenarios for a lead crack and more small MSD cracks as discussed by Broek and Swift are instructive to show that small MSD cracks can significantly reduce the load for unstable crack extension. Their failure scenarios also indicate that prevention of catastrophic consequences requires crack arresting capability of the structure. The ideas have been confirmed by recent experimental evidence analyzed by Broek summarized in the present report.
3. The engineering criterion for ligament failure between crack tips, based on touching of estimated crack tip plastic zones, is simple. It can be used to indicate possible failure scenarios. However, it can not be expected to be correct for more accurate predictions. The failure criterion then should preferably be based on local crack tip deformation characteristics, such as the CTOA.
4. The inhomogeneous hoop stress distribution is favourable for the possibility of MSD crack detection in the riveted joints of the aircraft fuselage.
5. Knowledge about MSD cracks occurring in service and/or full-scale fatigue tests is essential for the specification of relevant MSD inspection procedures. In this respect it is important to know whether cracks will occur in the top row or the bottom row of a longitudinal lap joint.
6. There are several design options to cope with future MSD problems of new aircraft fuselages. Designing for a longer riveted joint fatigue life is a good approach. Tear straps are another solution, where Glare will be an excellent material choice. A fuselage without crack stopper bands is attractive for obvious production arguments . This can only be justified if the stress level is sufficiently low, or if the material has a high fatigue resistance, such as the fiber-metal laminates (Arall and Glare).
7. There is still surprisingly little experimental information on MSD crack growth and residual strength. Some recommendations for investigations have been made. The experimental approach is emphasized in order to obtain a more detailed picture of the phenomena involved, which is necessary for guidance of analytical studies.

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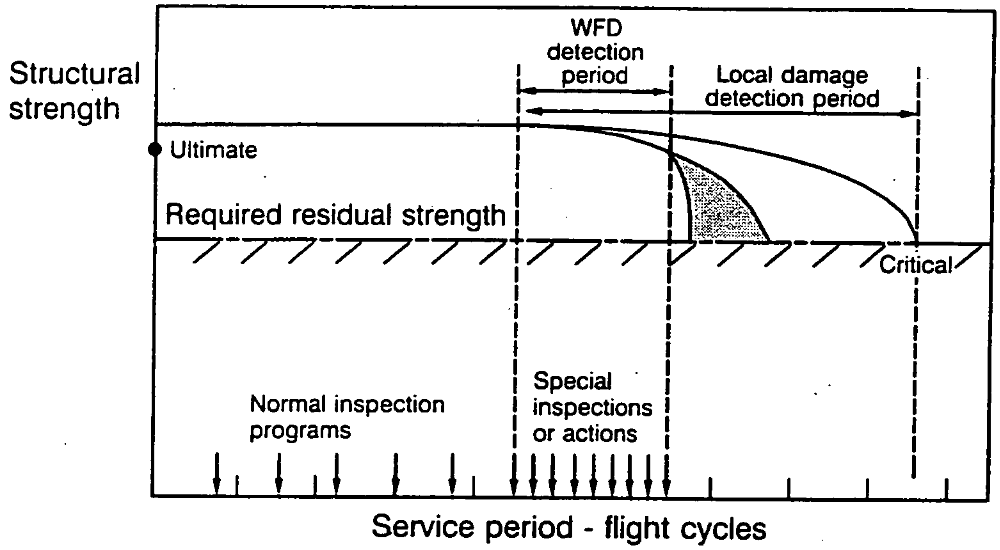

Fig. 1 Strength degradation of aging aircraft [1].

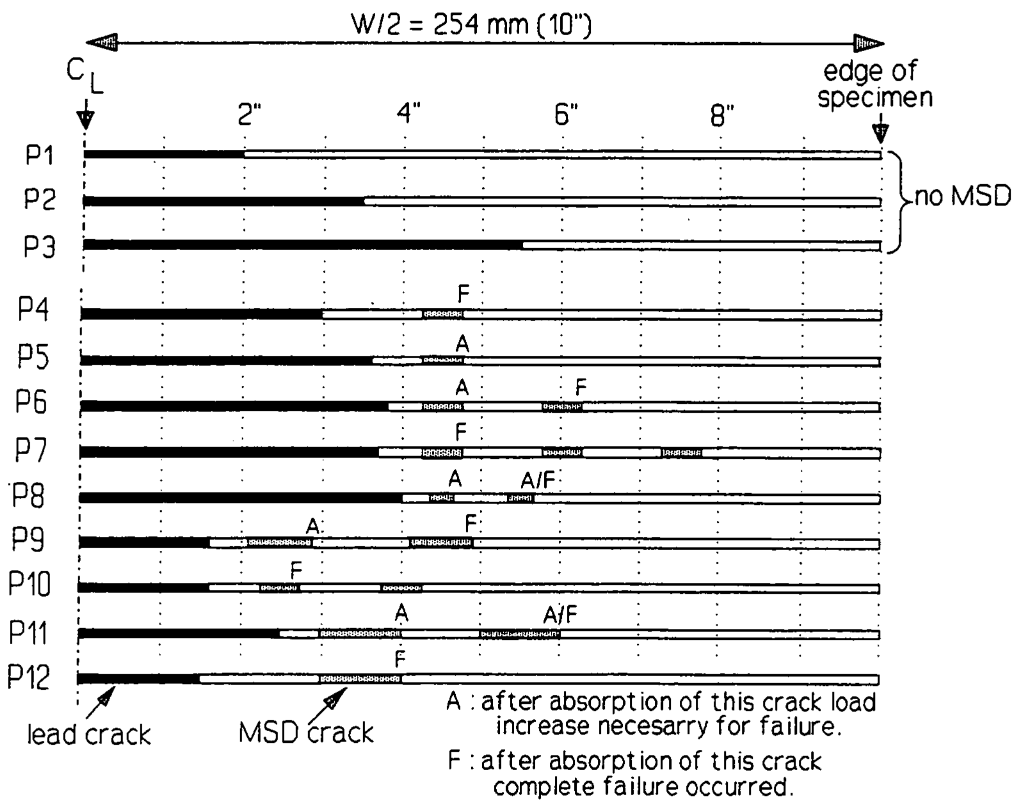

Fig. 2 Crack configurations of flat panels. Results of Broek [7].

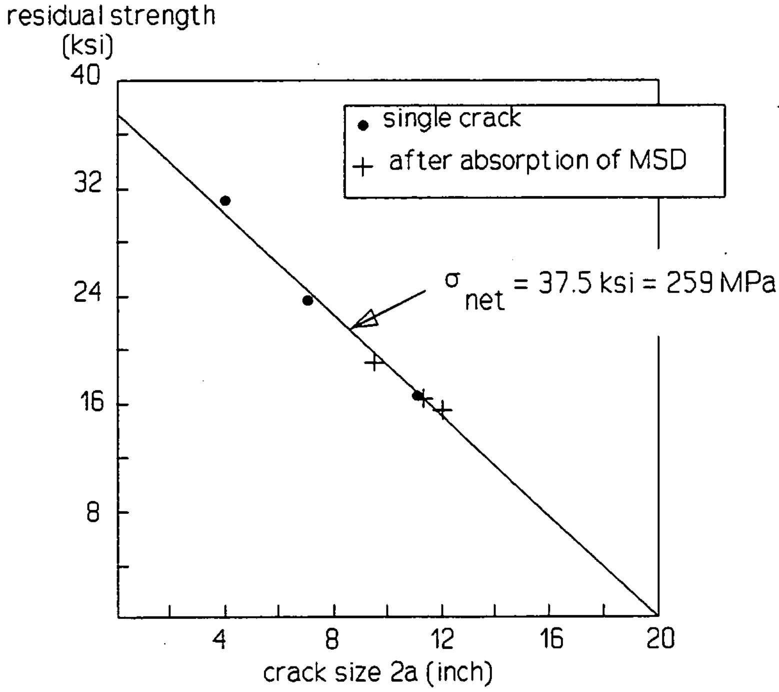

Fig. 3 Residual strength of flat panels.
Results of Broek [7]

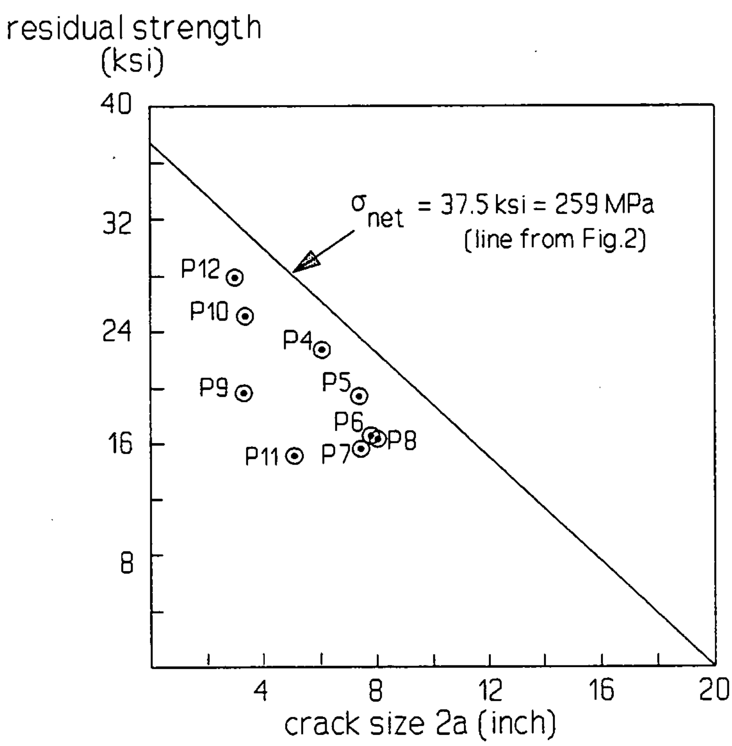

Fig. 4 Residual strength of flat panels with MSD.
Results of Broek [7]

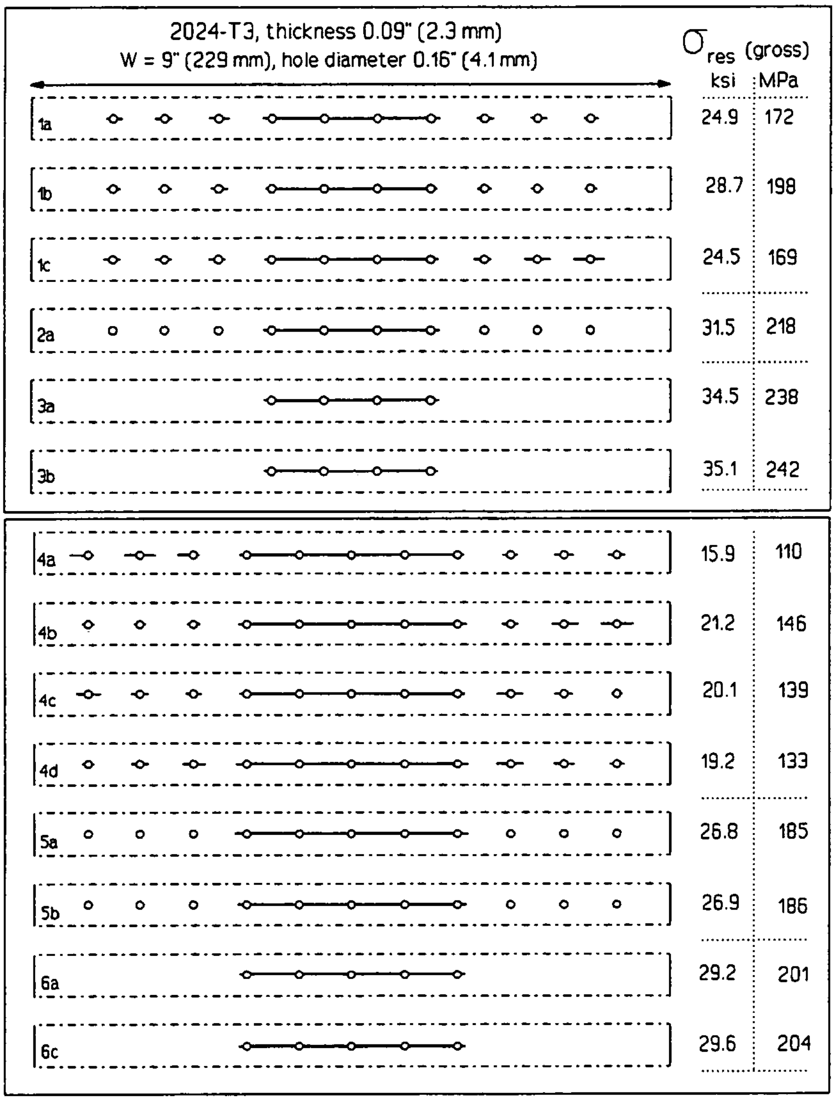

Fig. 5 Cracking patterns adopted by Moukawsher [10].

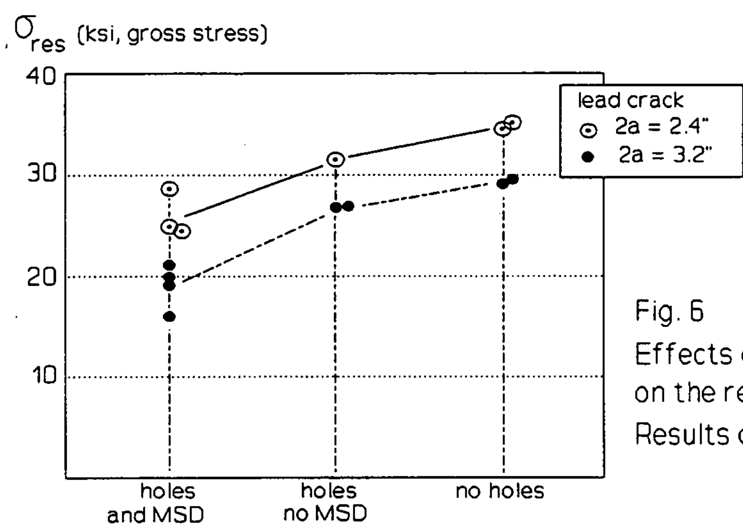

Fig. 6
Effects of MSD and holes on the residual strength.
Results of Moukawsher [10]

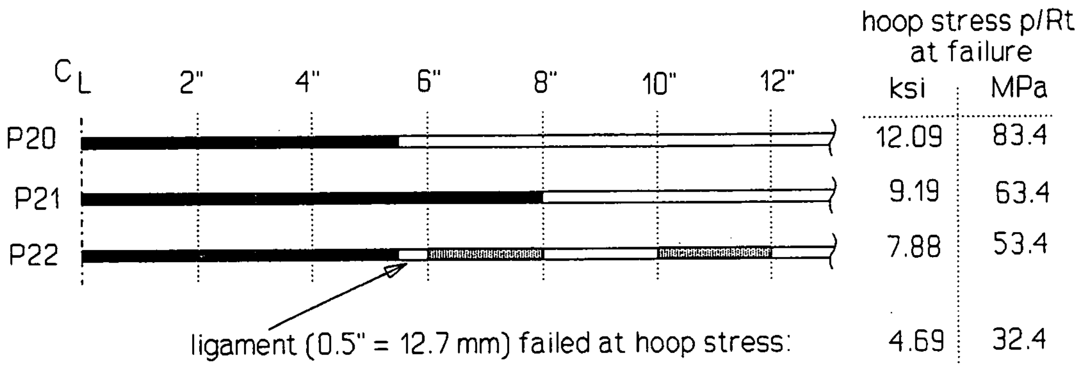

Fig. 7 Crack configurations of curved panels loaded biaxially under pressure. Results of Broek [7].

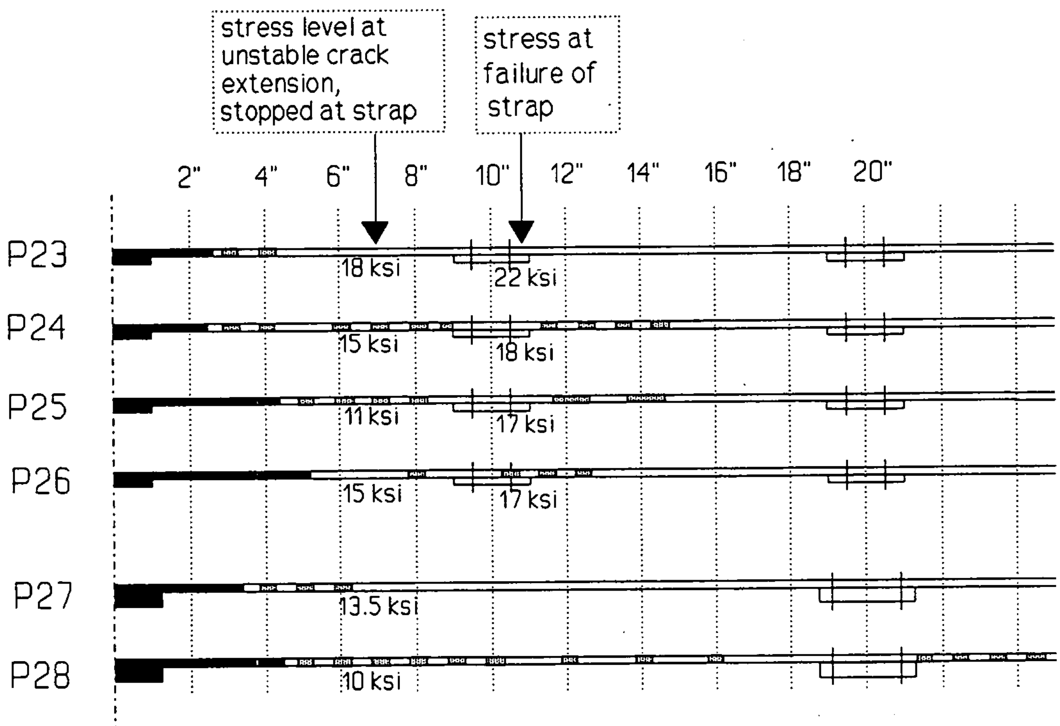

Fig. 8 Crack configurations of curved panels with crack stopper straps, loaded biaxially under pressure. Results of Broek [7].

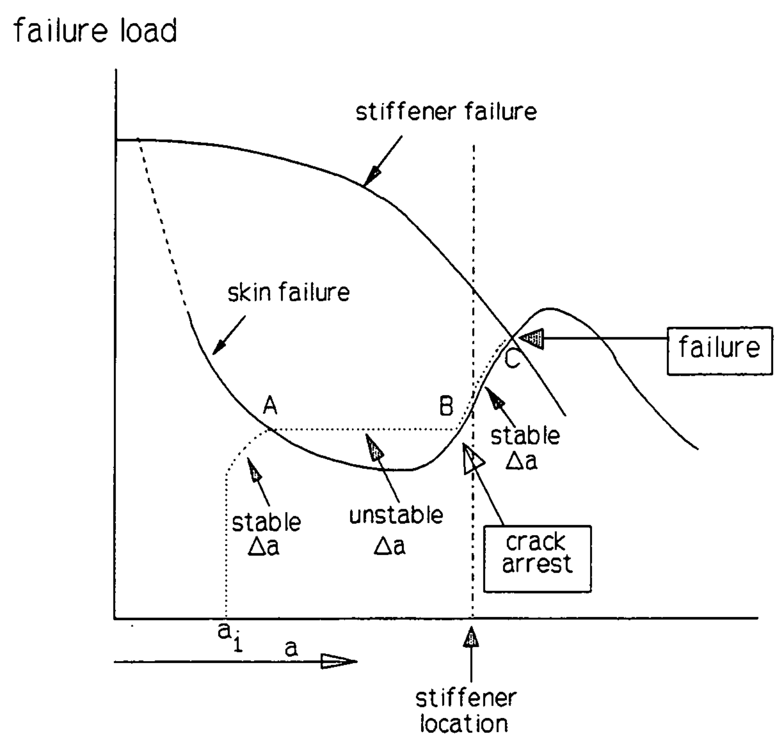

Fig. 9 Residual strength diagram.

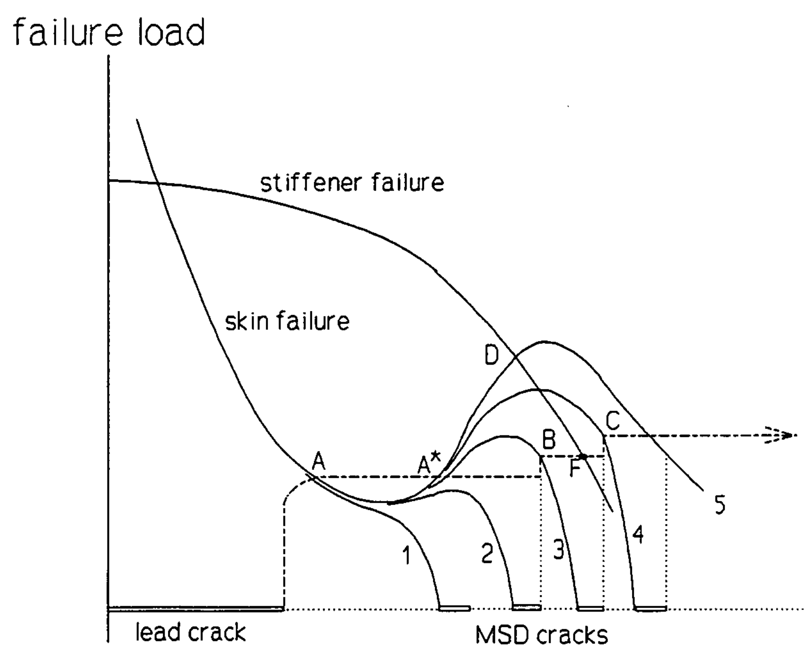

Fig. 10 Effect from MSD cracks on failure of lead crack.

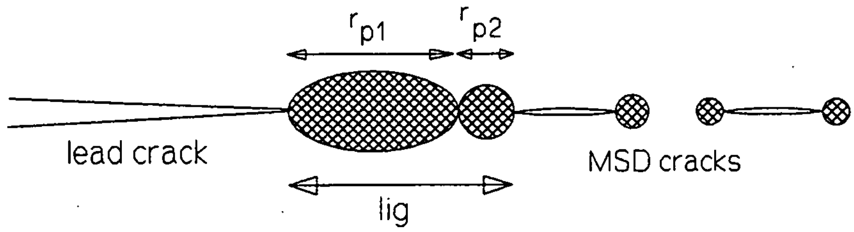

Fig. 11. Touching plastic zones as a criterion for ligament failure.

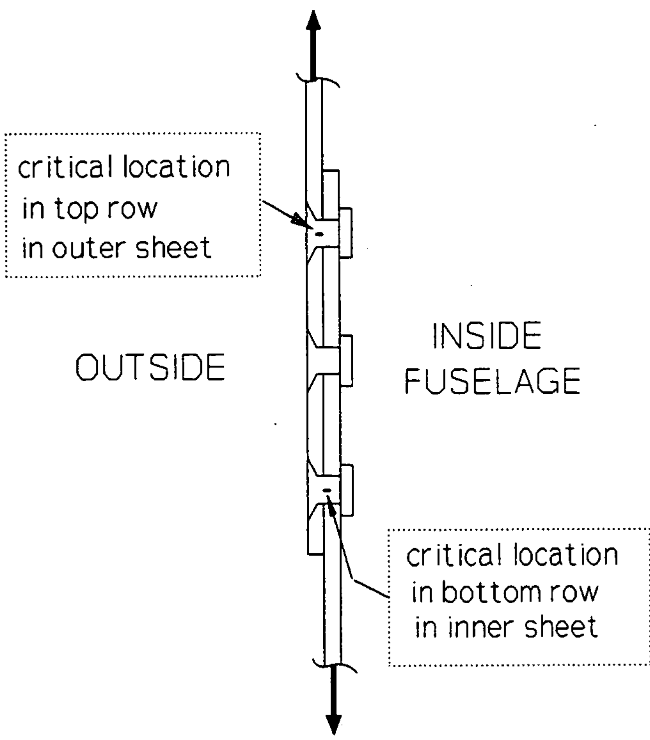

Fig. 12 Critical locations in outer sheet and inner sheet of fuselage lap joint.

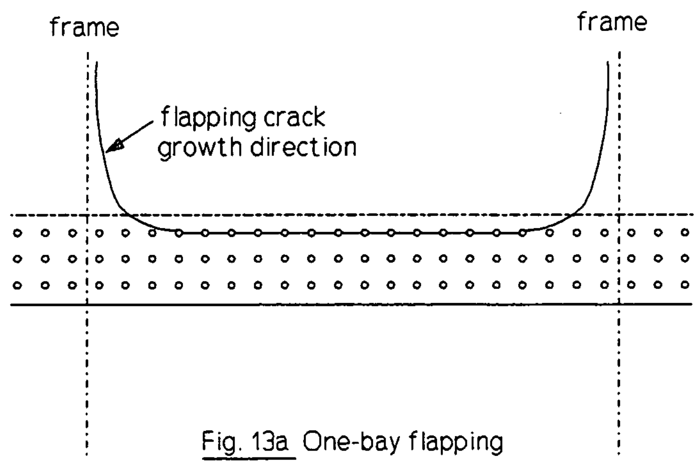

Fig. 13a One-bay flapping

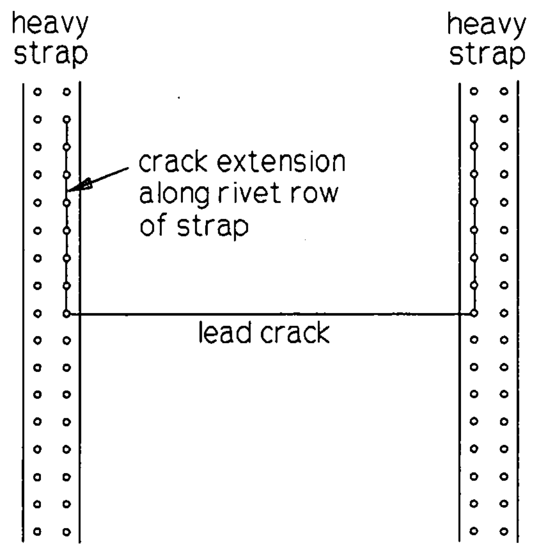

Fig. 13b Observation of Broek in
curved panels under pressure load

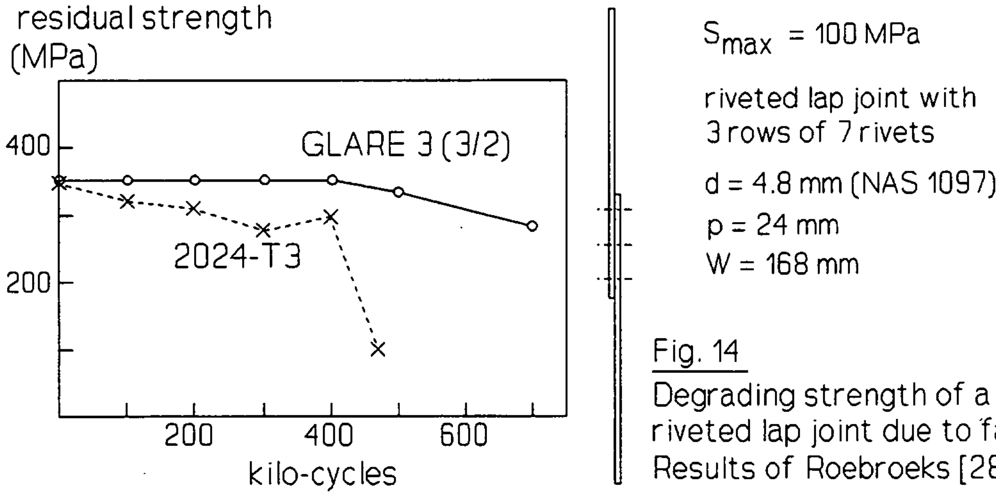

Fig. 14
Degrading strength of a
riveted lap joint due to fatigue.
Results of Roebroeks [28]

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