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Strength and Fatigue Life Enhancements of Cracked Metal

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Airframes and Engines Division
Aeronautical and Maritime Research Laboratory

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ABSTRACT

Measures to reclaim both the static strength and the fatigue life of plates with cracks are examined experimentally on 2024 aluminium alloy specimens. The simple measure of stop drilling the crack tips restored most of the tensile strength of the plate. To enhance the fatigue life of a cracked plate the process of stop drilling the crack tips, cold expanding the resulting holes and inserting interference-fit steel plugs has dramatically increased fatigue resistance. For the same spectrum fatigue life the stresses in the life-enhanced plates need to be 2.6 times those in the plain cracked plates. This factor remains practically constant in the presence of secondary bending when the average bending-stress/axial-stress ratio is 0.5. Surprisingly, the secondary bending reduced the fatigue life by only about 25% for both unenhanced and enhanced specimens. These results have obvious application to the lower wing skin cracking in the F-111 aircraft.

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Strength and Fatigue Life Enhancements of Cracked Metal

Executive Summary

The detection of fatigue cracks in RAAF aircraft structures and components usually raises the question of the adequacy of residual strength and remaining fatigue life. When replacement of the cracked region is unfeasible, attention is often given to means of reclaiming the strength and the life. It is obvious that to reduce crack growth rates some changes are required to alter the stress or the material environment at and ahead of the crack tip. When a crack is well established, local stress changes appear more effective than material changes in slowing crack growth. The present study arose in the context of fatigue cracking in the aluminium alloy lower wing skin of the F-111 aircraft, but has been conducted in a generic fashion.

Removal of the high stress concentration by crack tip drilling is an obvious method of increasing residual strength and fatigue life, which has been well documented in the literature. Such a process then allows two other means of enhancement. First, plastic expansion of the hole which, due to residual compression upon elastic recovery, gives a large beneficial decrease in the mean of the cyclic loading. Second, the use of interference-fit bushes, plugs, or fasteners, which significantly decrease the alternating component in the plate by providing an alternative load path, notwithstanding an increase in mean tensile stresses. These two approaches have had very little experimental verification in the context of crack tip stop drilling, in particular for bending conditions.

In this report, an experimental investigation has been carried out on cracked plates representative of the F-111 lower wing skin to address the following: (i) the influences of stop-drilling and of cold expansion of these holes on the residual strength, and (ii) the influence of a combination of stop-drilling, cold expansion, and the application of interference-fit plugs for increasing fatigue life. Since the local detail in the lower wing skin is subjected to considerable secondary bending stress, the influence of this variable on life enhancement is also examined. In the experimental program solid plugs were used. Fuel leakage from the wing is thereby more easily prevented, and with plugs flush to the outside surface the added precaution of a bonded composite patch is feasible.

It has been shown that the simple measure of stop drilling the crack tips restored most of the tensile residual strength of the cracked plate. Also the process of stop drilling the crack tips, cold expanding the resulting holes and inserting interference-fit steel plugs has dramatically increased fatigue resistance. For the same spectrum fatigue life the stresses in the life-enhanced plates need to be 2.6 times those in the plain cracked plates. This factor remains practically constant in the presence of secondary bending when the average bending-stress/axial-stress ratio is 0.5. Surprisingly, the secondary bending reduced the fatigue life by only about 25% for both unenhanced and enhanced specimens. These results have obvious application to the lower wing skin cracking in the F-111 aircraft.

Authors

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Dr J.M. Finney has spent more than forty years researching fatigue and fracture. His work has ranged widely over this topic – from the study of dislocation behaviour under fatigue stressing, through the relation between fatigue properties and micro-structure, fatigue crack growth behaviour and modelling and life estimation techniques, to structural fatigue of service aircraft.

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Contents

1. Introduction

The detection of fatigue cracks in high-performance structures and components usually raises the question of the adequacy of residual strength and remaining fatigue life. When replacement of the cracked region is unfeasible attention is often given to means of reclaiming the strength and the life. It is obvious that to reduce crack growth rates some changes are required to the stress or the material environment at and ahead of the crack tip. When a crack is well established, local stress changes appear more effective than material changes in slowing growth.

Without adding material there are several well-founded ways of changing stresses around the crack tip to enhance life. Removal of the high stress concentration by crack tip drilling is an obvious method. Such a process then allows two other means of enhancement. First, plastic expansion of the hole giving residual compression upon elastic recovery. The residual compression adds algebraically to the cyclic stress, resulting in a large decrease in the mean stress of the cycle, thus increasing life. There is evidence [1] that hole cold expansion not only increases fatigue life, but markedly increases the critical crack length. Second, the use of interference-fit bushes, plugs, or fasteners is a well established method [2] of improving fatigue life in components with holes. The mechanics of this process is that the oversize bush or fastener experiences compression while the material surrounding the hole is put into circumferential tension (though radial compression). Local tensile stresses at the hole due to the remotely-applied cyclic loading are partly offset by the unloading of the bush or fastener (i.e. relief of the interference) with the net effect that, although there is an increase in tensile mean circumferential stress at the hole, there is a decrease in the alternating component and a consequent increase in life.

The present study arose in the context of fatigue cracking in the aluminium alloy lower wing skin of the F-111 aircraft, but has been conducted in a fairly generic fashion. To address residual strength concerns, the influences of stop-drilling and of cold expansion of these holes on failing stress have been examined. Fatigue life extension has been explored by a combination of stop-drilling, cold expansion, and the application of interference-fit plugs. Solid plugs were used for the obvious reason of effectively removing the hole. Fuel leakage is thereby more easily prevented, and with plugs flush to the outside surface the added precaution of a bonded patch is feasible. The technique of stop drilling combined with cold expansion and interference fits has been established previously [3], but always in symmetrically-loaded specimens. The local detail in the lower wing skin is subject to considerable secondary bending stress and the influence of this variable on life enhancement is examined.

2. Material and Heat Treatment

The lower wing skin of the F-111 aircraft is manufactured from 2024-T851 aluminium alloy. The -T3 version of this alloy in an appropriate thickness, 3.66 mm, was readily available as alclad sheet. A large sheet of this material, from which all specimens were manufactured, was given the -T851 elevated-temperature ageing treatment: 190°C for 12 hours. Tensile tests were made on the aged material (detailed in Section 3), and from their load/displacement curves a yield stress of approximately 435 MPa was determined. For comparison, average handbook values of yield stress for 2024 are as follows:

2024-T3	345 MPa
2024-T851	465 Mpa

It follows that the elevated temperature ageing treatment did produce material quite close to the -T851 temper, despite the material having been manufactured and aged to the -T3 condition some years earlier.

3. Residual Strength Tests

3.1 Specimens and Test Conditions

All specimens were manufactured with the longitudinal axis in the rolling direction of the sheet, and all were made from the one sheet. They each measured 400 mm long, 125 mm wide overall, and 3.66 mm thick. Figure 1 gives the geometry of the various specimen configurations. Tensile specimens were manufactured with a test section width reduced to 60 mm, their test results giving a residual strength baseline. The "fatigue-cracked" specimens all contained a central flaw totalling 40 mm in length, as shown in Fig. 1. Two main types were manufactured, those containing an actual fatigue crack, and those where the end of the flaw was terminated with a "stop-drilled" hole.

Specimens with flaws terminating in a fatigue crack were made with a 5 mm diameter central hole and a fine saw cut either side totalling 36 mm in length. These were then fatigued under constant-amplitude loading to give 2 mm crack growth at each end. The fatigue conditions were: $\Delta K = 11\ \mathrm{MPa}\sqrt{\mathrm{m}}$, $R = 0.1$, at 10 Hz. The average growth rate under these conditions was 2.35×10^{-7} m/cycle (which is close to the data in the Damage Tolerant Design Handbook [4]).

Stop drilled specimens were manufactured by drilling and reaming the holes followed by a fine saw cut to join the two holes. Again the total flaw length was 40 mm. The hole diameter and the degree of cold expansion of the hole were varied to produce six hole conditions as shown in Table 1. Specific mandrels were manufactured to give either 4% or 10% cold expansion. As with any mandrel technique of cold expansion, the out-of-plane displacements slightly distorted the specimens, and the magnitudes of the distortions are given in Table 2.

3.2 Results and Discussion

Five tensile specimens and four of each of the other types of specimen were tested to fracture in a 500 kN Instron servohydraulic test machine and using a load ramp rate of 50 kN/minute. Load/actuator-displacement plots were made for each specimen test.

Table 3 gives the individual test results and Table 4 summarises average behaviour. Residual strength, in an aircraft structure, is defined as the gross-area failure stress. For a small crack in a large structure the gross-area and net-area failure stresses are practically identical, but not so in small specimens. In the present tests it is believed that the proper basis of comparison is net-area stress.

Table 4 shows that the net-area failure stress for the fatigue-cracked specimens is only 56% of the baseline residual strength. It is clear also from this Table that simply stop drilling a crack restores most of the tensile strength of the specimen; for a 4 mm diameter hole the residual strength is 91% of the baseline, for the 10 mm diameter hole it is 94%. The degree of cold expansion also bears little on the residual strength. From Table 3 it is evident that there is little scatter in failure stress for any specimen configuration. The scatter in percentage residual strengths in Table 4 is, on average, just less than ±1%.

4. Fatigue Life Enhancement Tests

4.1 Purpose

Fatigue life extension of cracked components by using mechanical means to alter crack tip stresses is a well established practice. The present work aimed to determine the amount of life extension when the tips of a central through-crack in a plate are stop drilled and the holes are then reamed, cold-expanded, and fitted with interference-fit plugs. Solid plugs have been used for the dual purposes of producing the interference fit and geometrically removing the hole. This then would allow, if ever desired, the application of a bonded patch to one of the surfaces by having the plug flush with that surface. This extra measure has not been examined in the present work.

Secondary bending is common in highly stressed regions of aircraft structure, yet its effect on fatigue life, particularly for regions that have received some life enhancement treatment, has been little investigated. The present tests have examined, on simple cracked plates, the influence of substantial secondary bending on the improvements in fatigue life by the measures noted in Section 3.2. Elementary reasoning would indicate that secondary bending of cracked plates without any life enhancement measures should have a large influence on remaining fatigue life. It is not at all certain what the influence of secondary bending would be in the presence of measures to promote fatigue life.

4.2 Specimens and Test Conditions

The geometry of the baseline specimens, that is, those which contain a simple through-thickness fatigue crack, is identical to the static fracture specimens shown in Fig. 1(b), the only exception being that the specimen length was 350 mm instead of 400 mm. For these 'unenhanced' specimens the 40 mm long fatigue crack was generated by initially drilling a 5 mm diameter central hole, then making a fine saw cut either side to give a total flaw length of 30 mm, and finally applying constant-amplitude fatigue cycling to give 5 mm of crack growth at both ends. The fatigue cycling conditions commenced with $\Delta K = 9.8$ MPa$\sqrt{m}$, R = 0.1, 5 Hz, which gave an average growth rate of 2.85×10^{-7} m/cycle.

The 'life-enhanced' specimens also had a central through-thickness crack but with 10 mm diameter stop-crack holes at both tips. (Again, apart from the length of 350 mm, these specimens were identical with the static fracture specimens of Fig. 1(c). They were also manufactured the same way, namely, that the two holes were drilled and reamed first and then a fine saw cut made between them). The holes were given a 3.4% cold expansion and then fitted with a 0.38% interference-fit steel plug. Again the total flaw length was 40 mm.

These specimens were tested under a modified FALSTAFF sequence in order to determine the remaining fatigue life. The modification to FALSTAFF was simply to truncate all negative loads to zero to avoid the need for anti-buckling restraints. The truncation reduced the number of load levels from 32 to 26, and the number of turning points from 35966 to 35656 for the 200-flight block length. All tests were made at an average cyclic frequency of 5 Hz and maximum peak stress levels such as to give lives typical of fighter aircraft, ranging from about 3000 to about 15000 flights. Digitally-controlled servo-hydraulic test machines were used for all fatigue tests.

The specimens described above, namely, those 'unenhanced' and those 'life-enhanced', were tested in the usual symmetrical manner with no secondary bending for the purpose of determining experimentally the influence on fatigue life of the life enhancement measures. A further set of specimens was tested for determining the influence of secondary bending on the fatigue lives of both unenhanced and life-enhanced conditions. Again, the specimens were identical in all respects to those noted above, except that their length was 280 mm. They were fitted into a simple secondary-bending jig which was then loaded by the test machine. The geometric arrangement is shown in Fig. 2. A strain-gauged specimen was used to experimentally determine the jig dimensions to give the same secondary-bending ratio as an F-111 wing at a particular region of lower skin fatigue cracking. The value aimed at was 0.49 when the load was 50% of maximum.

Figure 3 shows the wing secondary bending with respect to load and the simple plate specimen simulation. It is obvious that the secondary bending ratio for the simple specimens was considerably more load dependent than for the wing. The skin of the F-111 wing is much more geometrically constrained than the specimens, and it would be impossible to reproduce the same load dependency of the secondary bending ratio without resorting to a much more complicated specimen and perhaps loading geometry. The specimen test condition chosen for obtaining the same average secondary bending ratio as the wing was a peak tension gross-area stress of 70 MPa which gave lives around 10000 flights for the unenhanced specimens. The same secondary bending geometric conditions were then used for all specimen tests.

4.3 Results and Discussion

The fatigue test lives of the plate specimens for each of the four conditions described above are shown in Fig. 4. Two things are immediately obvious. The life enhancement process of stop drilling a crack, cold expanding the resulting hole and fitting the hole with an interference-fit steel plug, has increased fatigue resistance remarkably. It is usual in fatigue life enhancement studies to quote a *life* improvement ratio, that is, the enhanced life divided by the unenhanced life, both taken at the same test stress, and ratios in the range 1.5 to 8 or 10 are common [5]. With the present results this ratio approaches infinity; in fact to achieve a life of say 10000 flights the stresses in the life-enhanced specimens need to be about 2.6 times larger than those in the unenhanced specimens. The enhancement measures are thus remarkably effective.

The second obvious conclusion is that the secondary bending was remarkably ineffective in reducing fatigue life. In the unenhanced specimens the secondary bending reduced the life by about 25%, and in the enhanced specimens the reduction was about 27%. These reductions are minuscule in comparison with the change in life by the enhancement measures used. The stress increase needed to fail the enhanced specimens under secondary bending at 10000 flights is about 2.7 times that to fail the unenhanced specimens under secondary bending. It is transparently obvious that, in the presence of significant secondary bending (an average ratio of 0.49), stop drilling a crack, cold expanding the hole and fitting an interference plug, have, together, prevented flaw growth and fatigue failure. The scatter in fatigue lives is quite small for the unenhanced specimens and somewhat larger for the enhanced specimens. This difference probably can be attributed to the practical difficulty of manufacturing enhanced specimens with exactly the same degrees of cold expansion and interference of the plug.

No attempt has been made to optimise the enhancement conditions. From all the control measurements taken, however, there appears to be a correlation between the lives of the enhanced specimens and the amounts of plug interference. To indicate the potential for further life enhancement, one specimen was made with plug interferences of 0.67% (about the limit of elastic interference), and tested without secondary bending at a peak stress of 200 MPa. The result can be seen in Fig. 5, and indeed there has been an increase in fatigue life. Taking average lives the increase is nearly a factor of two, though further tests are obviously needed.

Figure 6 shows the fracture surfaces for each of the four test conditions. These are taken from specimens with lives in the region of 5000 flights. There is considerable crack growth in the unenhanced specimens, but much less in the enhanced specimens, no doubt due to the much higher stresses used. It is also apparent from the photographs that the secondary bending has caused some asymmetry of the crack front. What is particularly noticeable is that for the unenhanced specimens the cracks were observed to grow immediately upon spectrum loading, but for the enhanced specimens, cracks were observed only very much later in the spectrum life, as shown in Fig. 7. The process of fracture in the enhanced specimens is also quite interesting. At the commencement of the tests (at 200 MPa) the elongation of the hole under the highest loads in the sequence was observed microscopically to be of such an extent that a gap momentarily appears between the plate and the plug at the top and bottom of the hole. Continued cycling produces fine-scale irreversible plastic deformation in the plate adjacent to the hole in the region of maximum hoop stress. This deformation results in further elongation of the hole; Fig. 8 illustrates this process. Hence, in these regions, the interference of the plug decreases to zero as cycling proceeds but there is still contact between the plug and the plate at the maximum hoop stress region as illustrated in Fig. 8. This contact region then becomes a site of intense fretting and, eventually, of observable crack growth.

5. Conclusions

1. For a cracked plate the simple expedient of stop drilling the crack tips restores most of the tensile strength of the plate. The diameter of the hole and the degree of any cold expansion of the hole bear little on the residual strength.

2. The fatigue life enhancement process of stop drilling a crack, cold expanding the resulting holes and inserting interference-fit steel plugs has dramatically increased the fatigue resistance of a cracked plate. Under spectrum fatigue loading the stresses in the life-enhanced plates were 2.6 times those in the plain cracked plates for the same fatigue life.

3. When secondary bending, to an average ratio of 0.49, is imposed on either unenhanced or enhanced specimens, the reduction in spectrum fatigue life is, surprisingly, only about 25%. It follows that, in the presence of secondary bending, the life enhancement measures are equally effective.

4. In plain cracked plates the cracks continued to grow immediately upon application of the spectrum loading. In the life enhanced plates crack growth was observable only after about 60–90% of the spectrum fatigue life. In the latter specimens the cracks initiated as a result of fretting between the plug and the hole.

6. Acknowledgment

The authors wish to thank Dr M. Heller and Dr G. Jost for their helpfull comments relating to this work.

7. References

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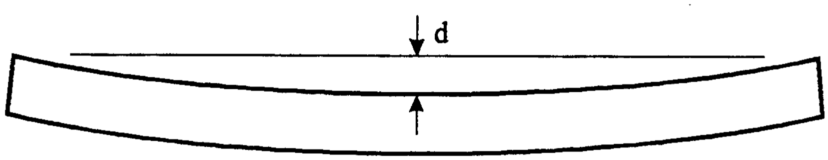
Table 1. Hole conditions

Table 1. Hole conditions	
Reamed hole diameter	Cold expansion
4 mm	none
4 mm	4%
4 mm	10%
10 mm	none
10 mm	4%
10 mm	10%

Table 2. Approximate distortions



Approximate distortions

Hole diameter	Cold expansion	Average displacement d (mm)*	
		across width of specimen	along length of specimen
4 mm	4%	zero	zero
4 mm	10%	0.1	0.1
10 mm	4%	0.2	0.2
10 mm	10%	1.5	3.2

* accuracy roughly 0.2 mm

Table 3. Residual strengths of 2024-T851 aluminium alloy sheet

Specimen type	Specimen number	Width "W" (mm)	Flaw length "2a" (mm)	Thickness "t" (mm)	Failure force (kN)	Gross-area failure stress (MPa)	Net-area failure stress (MPa)
Tensile	FX5E5	59.2		3.66	103.9	479.5	479.5
	FX5E6	59.31		3.66	104.2	480.0	480.0
	FX5E7	59.41		3.66	104.3	479.7	479.7
	FX5E8	58.77		3.66	102.9	478.4	478.4
	FX5E4	59.09		3.66	103.6	479.0	479.0
					Average:	479.3	479.3
Fatigue cracked	FX5B2	124.8	40.1	3.66	85.4	187.0	275.5
	FX5B5	124.81	40.49	3.66	84.9	185.9	275.1
	FX5B6	124.71	40.66	3.66	79.7	174.6	259.1
	FX5B7	124.96	40.67	3.66	81.8	178.9	265.2
					Average:	181.6	268.7
4mm holes, no CX	FX5D9	125.05	39.4	3.66	137.5	300.4	438.6
	FX5E1	125.13	39.41	3.66	134.9	294.6	430.0
	FX5E2	124.87	39.4	3.66	135.8	297.1	434.1
	FX5E3	125.24	39.38	3.66	138.1	301.3	439.5
					Average:	298.4	435.5
4mm holes, 4%CX	FX5D5	125	39.98	3.66	138.3	302.3	444.4
	FX5D6	124.95	39.89	3.66	138.3	302.4	444.2
	FX5D7	125.01	39.34	3.66	139.3	304.5	444.3
	FX5D8	124.9	39.4	3.66	138.3	302.5	442.0
					Average:	302.9	443.7

Specimen type	Specimen number	Width "W" (mm)	Flaw length "2a" (mm)	Thickness "t" (mm)	Failure force (kN)	Gross-area failure stress (MPa)	Net-area failure stress (MPa)
4mm holes, 10%CX	FX5C9	124.95	39.98	3.66	138.1	302.0	444.1
	FX5D2	125.27	40.05	3.66	137.5	299.9	440.8
	FX5D3	125.29	40.03	3.66	139.9	305.1	448.3
	FX5D4	125.3	39.93	3.66	140.5	306.4	449.7
					Average:	303.3	445.7
10mm holes, no CX	FX5C5	124.95	39.99	3.66	140.3	306.8	451.2
	FX5C6	124.99	40	3.66	140.7	307.6	452.3
	FX5C7	124.97	39.9	3.66	140.9	308.1	452.5
	FX5C8	124.97	39.91	3.66	140.2	306.5	450.3
					Average:	307.2	451.6
10mm holes, 4%CX	FX5B1	124.8	39.98	3.66	136.7	299.3	440.3
	FX5B3	124.88	40.01	3.66	141.7	310.0	456.2
	FX5B8	124.96	39.99	3.66	141.5	309.4	455.0
	FX5B9	124.97	39.99	3.66	140.5	307.2	451.7
					Average:	306.5	450.8
10mm holes, 10%CX	FX5C1	124.83	40.02	3.66	139.6	305.6	449.7
	FX5C2	124.82	39.99	3.66	140.5	307.5	452.5
	FX5C3	125.24	39.99	3.66	142.6	311.1	457.0
	FX5C4	124.99	39.98	3.66	142.3	311.1	457.4
					Average:	308.8	454.2

Table 4. Comparative residual strengths of 2024-T851 aluminium alloy sheet

Specimen type	Gross-area		Net-area	
	failure stress (MPa)	% tensile strength	failure stress (MPa)	% tensile strength
Tensile	479.3	100%	479.3	100%
Fatigue cracked	181.6	38%	268.7	56%
4mm holes, no CX	298.4	62%	435.5	91%
4mm holes, 4%CX	302.9	63%	443.7	93%
4mm holes, 10%CX	303.3	63%	445.7	93%
10mm holes, no CX	307.2	64%	451.6	94%
10mm holes, 4%CX	306.5	64%	450.8	94%
10mm holes, 10%CX	308.8	64%	454.2	95%

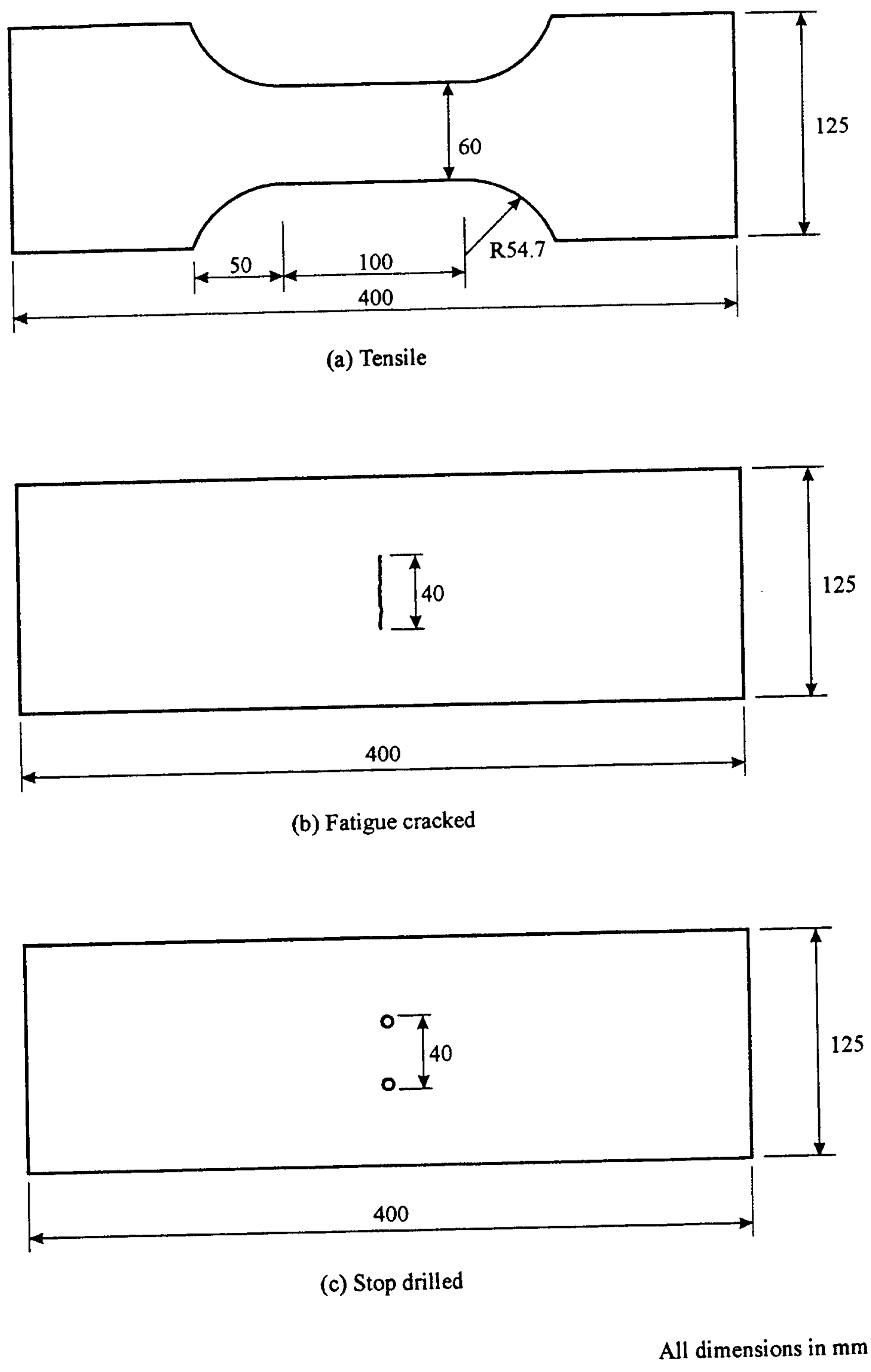


Figure 1. Static fracture test specimens 2024-T851 aluminium alloy, 3.66 mm thick

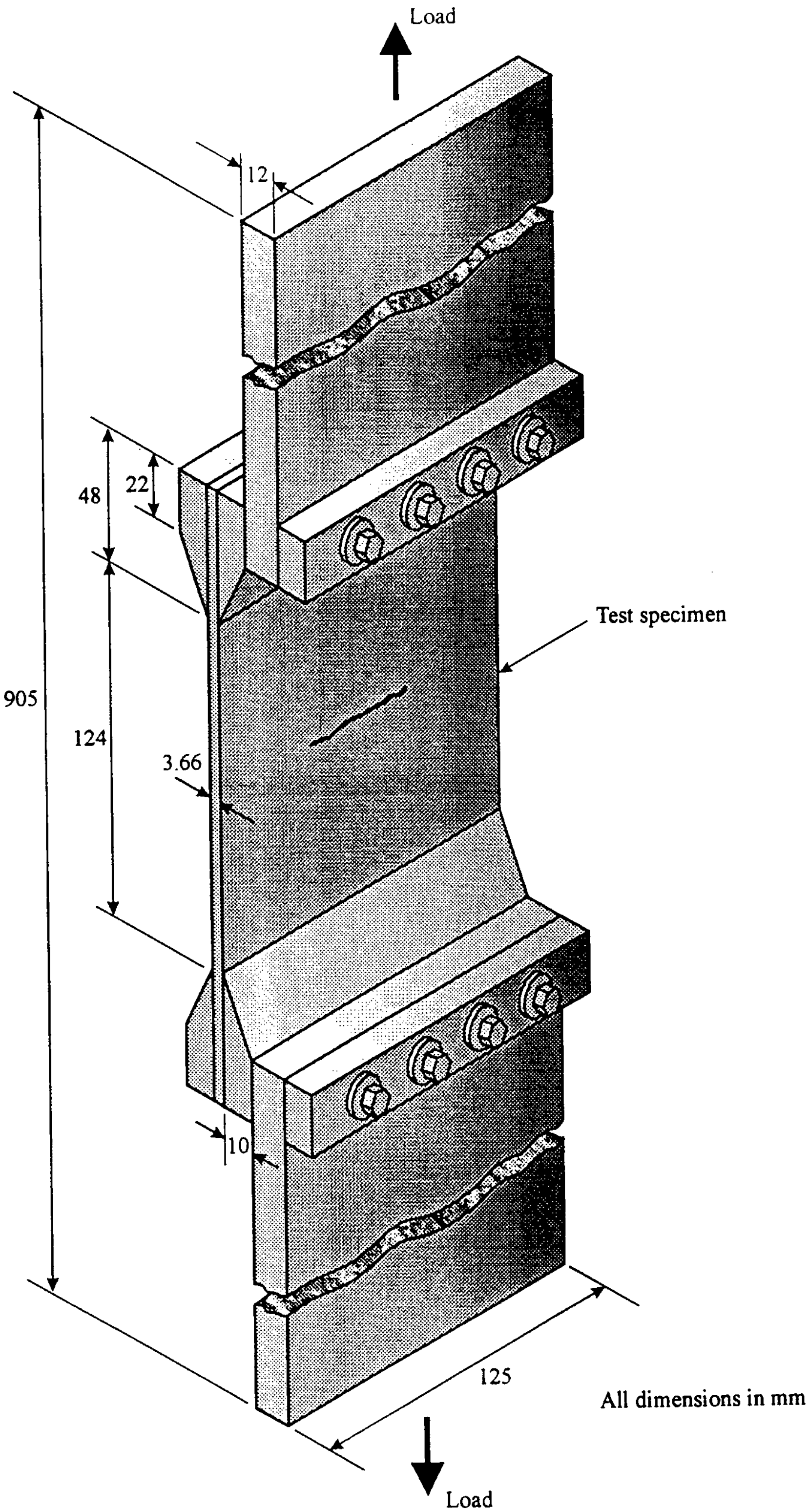


Figure 2. Secondary bending test arrangement

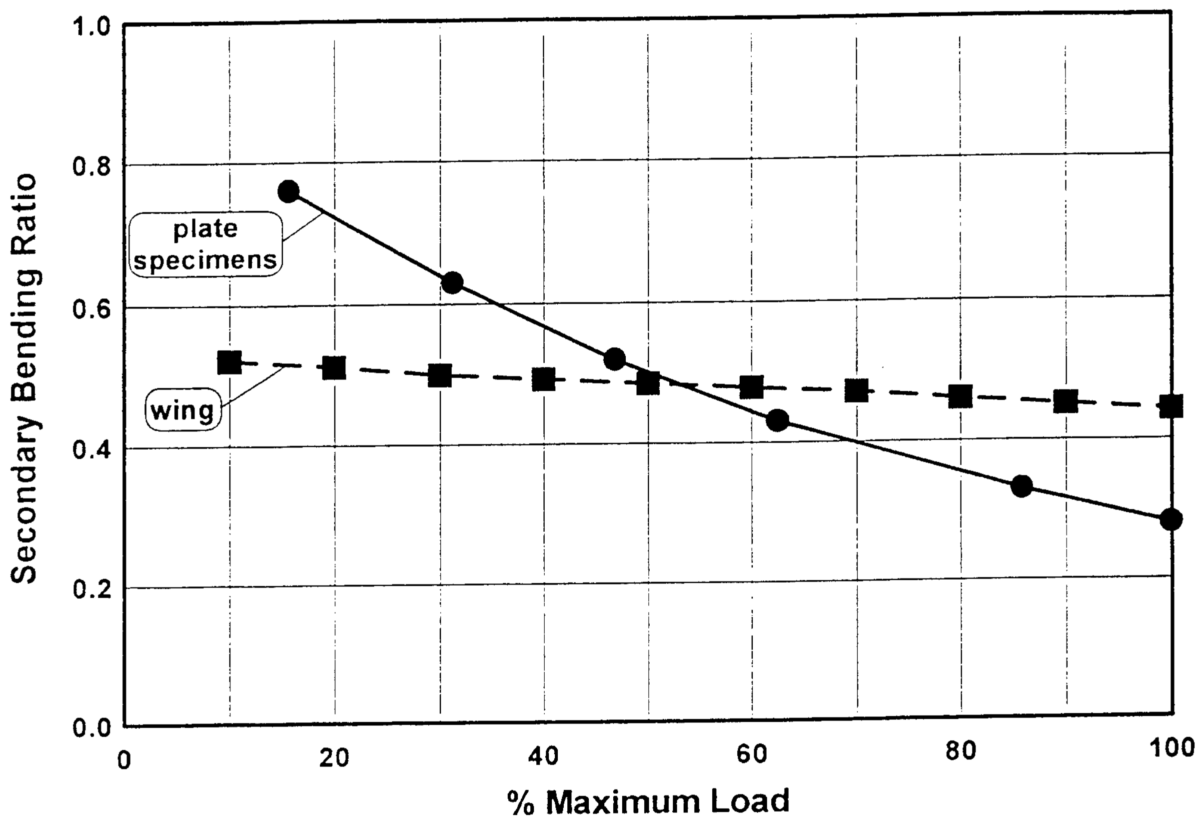


Figure 3. Secondary Bending Ratios

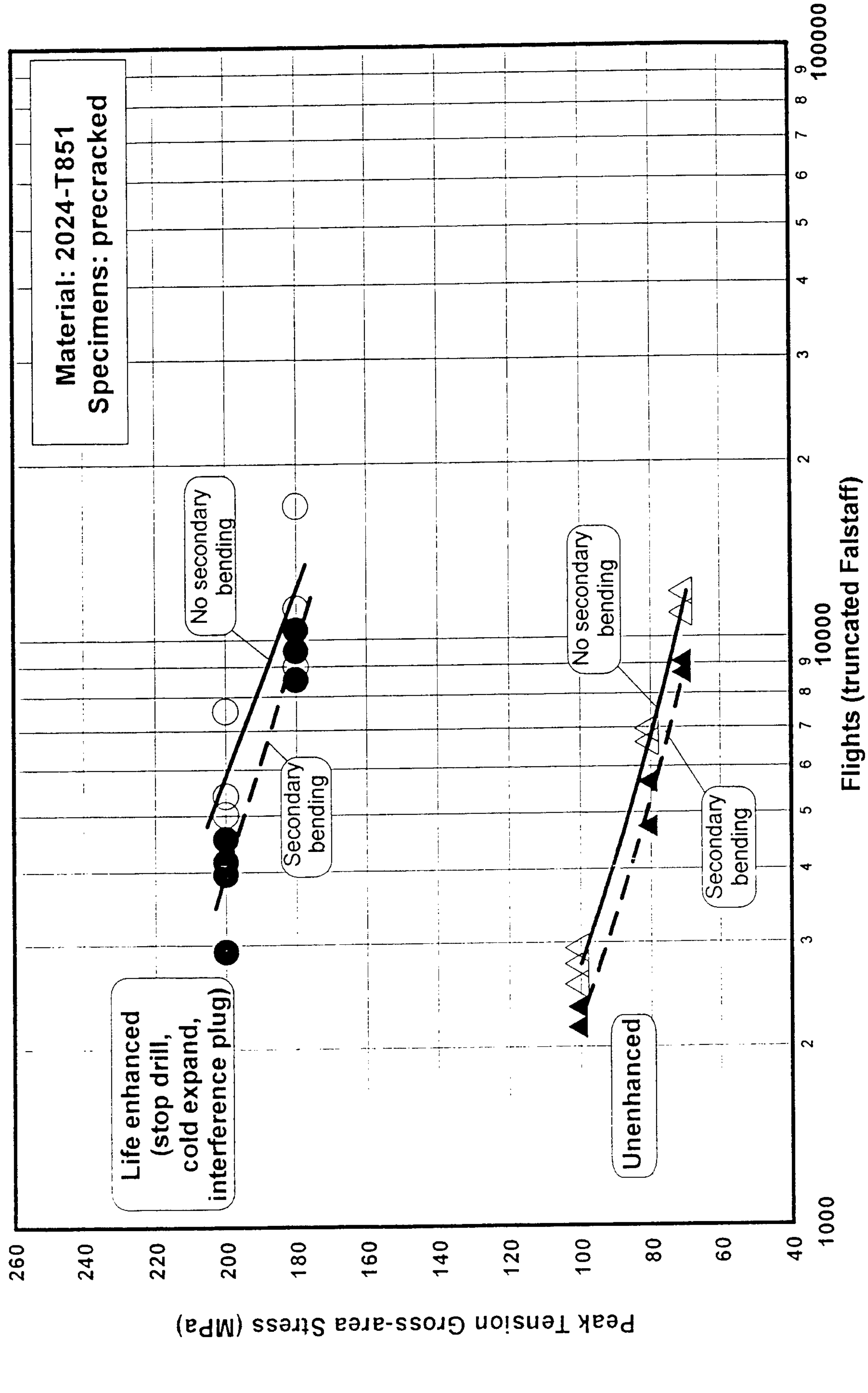


Figure 4. Spectrum fatigue lives of plate specimens

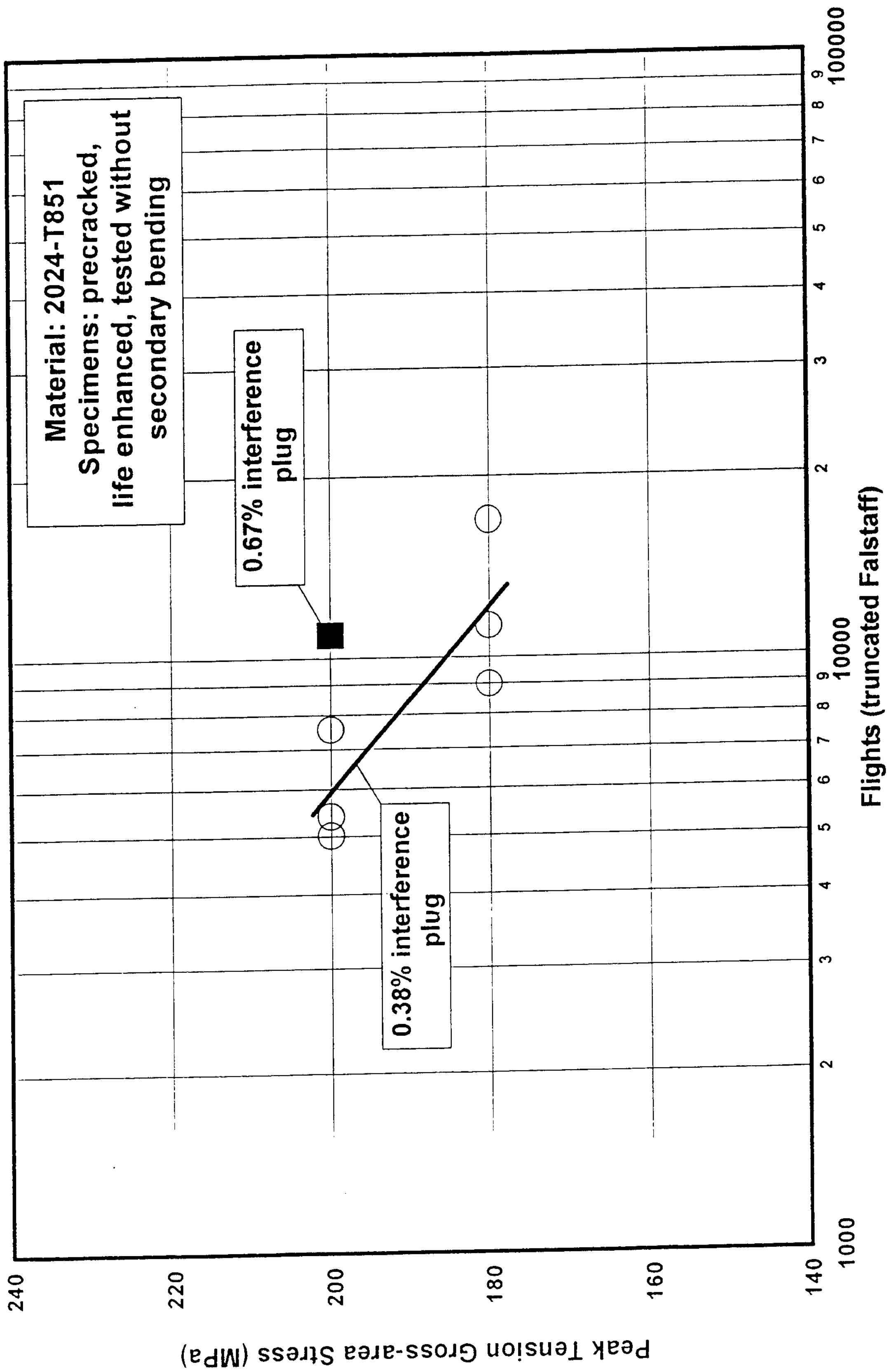
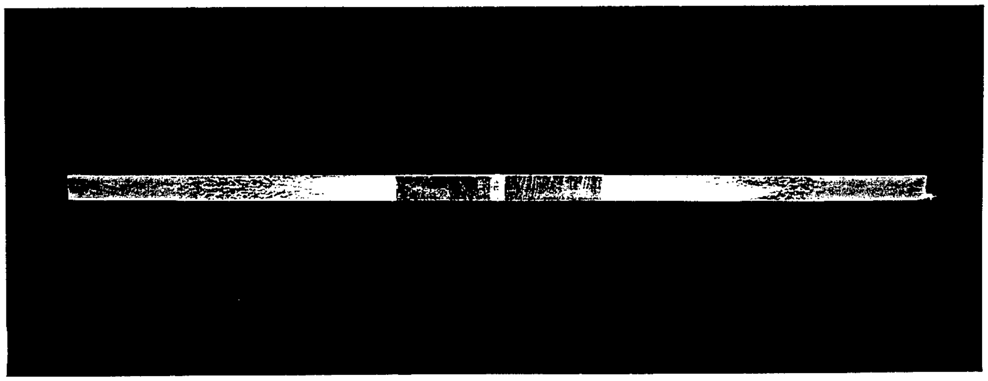
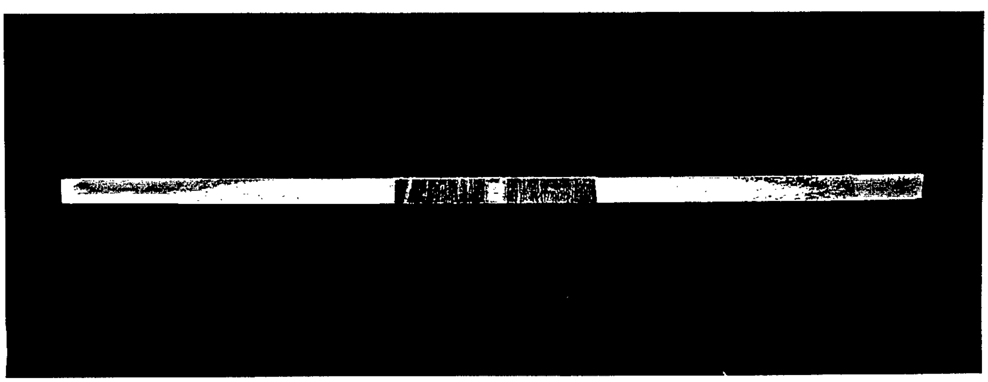


Figure 5. Influence of plug interference on spectrum fatigue life

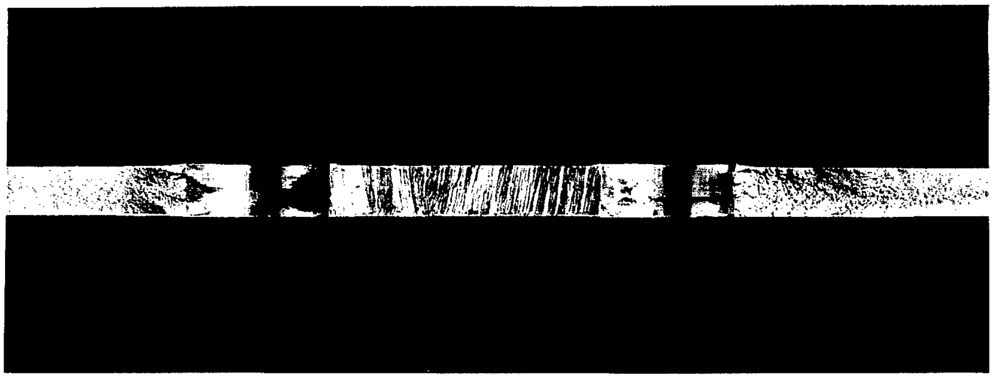


(a) unenhanced, no secondary bending
peak tensile fatigue stress: 80 Mpa
spectrum fatigue life: 6955 flights

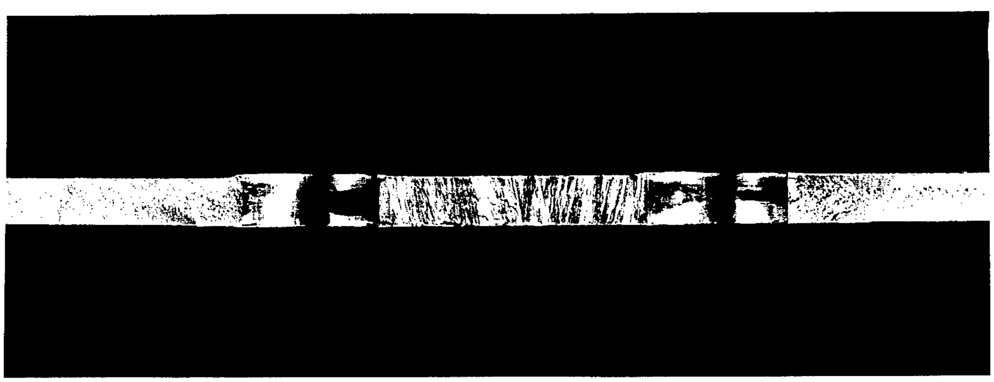


(b) unenhanced, with secondary bending
peak tensile fatigue stress: 80 Mpa
spectrum fatigue life: 5638 flights

Figure 6. Fracture surfaces of fatigue test specimens



(c) enhanced, no secondary bending
peak tensile fatigue stress: 200 Mpa
spectrum fatigue life: 5415 flights



(d) enhanced, with secondary bending
peak tensile fatigue stress: 200 Mpa
spectrum fatigue life: 4564 flights

Figure 6 (cont'd). Fracture surfaces of fatigue test specimens

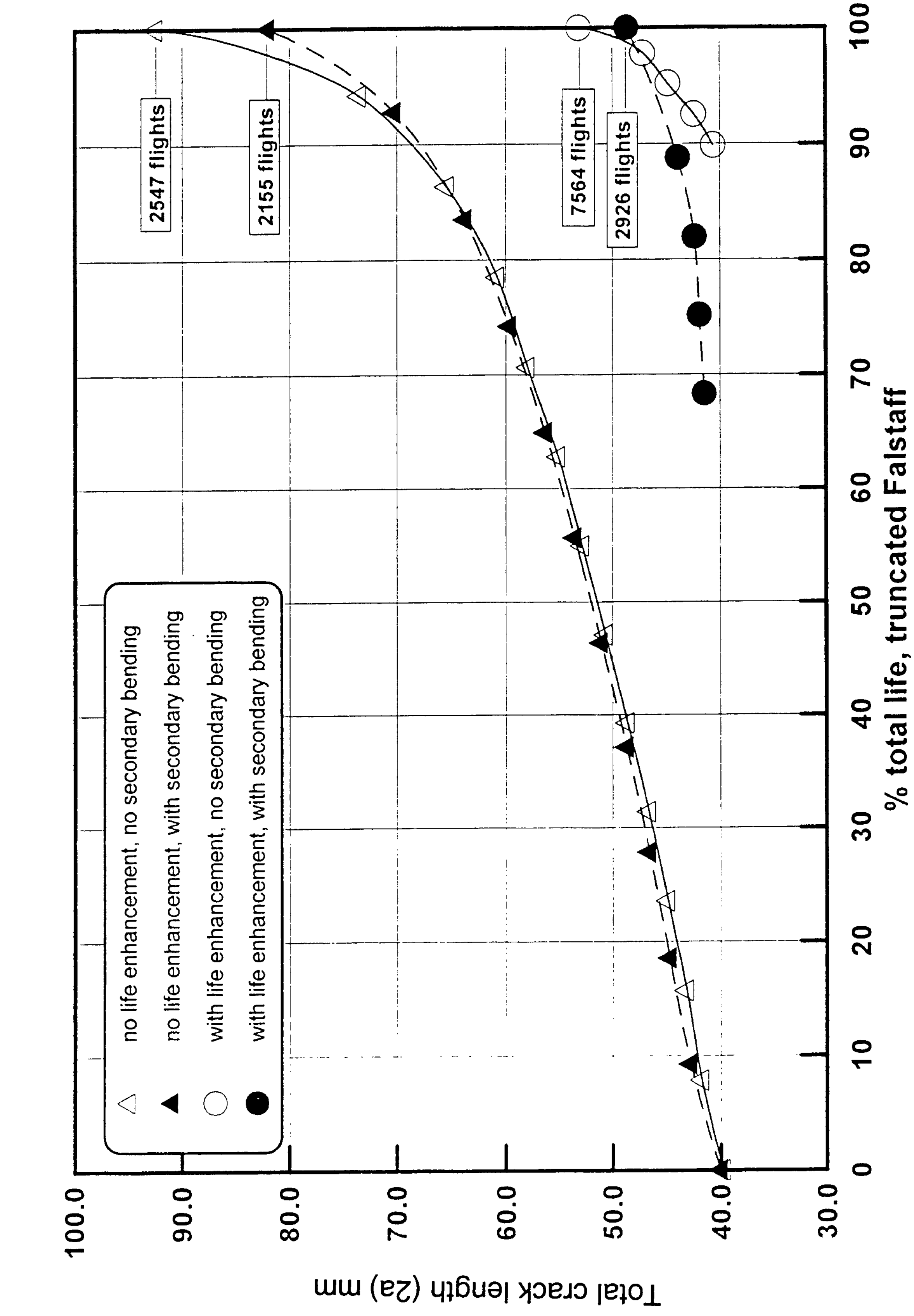


Figure 7. Crack growth curves

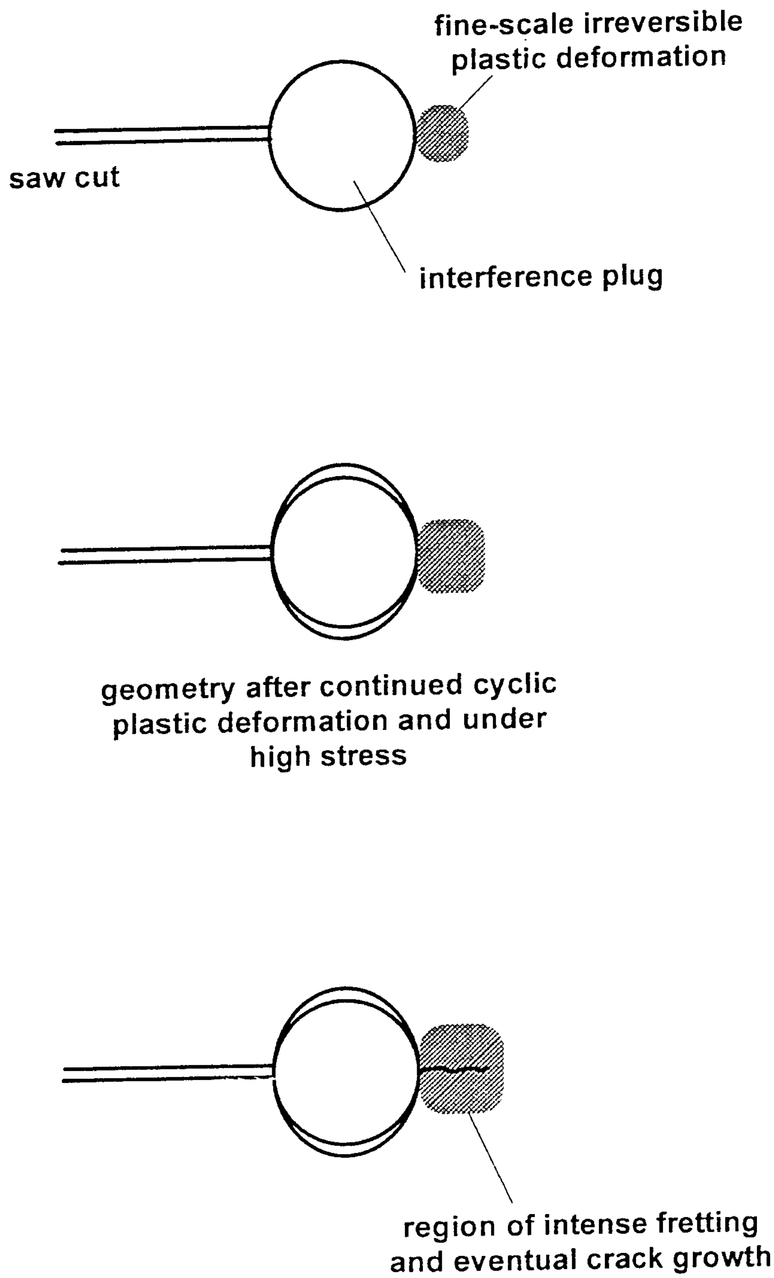


Figure 8. Fracture process in life enhanced specimens.

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J.M. Finney, C. Niessen, N. Absolom, K. Lemm

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19. ABSTRACT
Measures to reclaim both the static strength and the fatigue life of plates with cracks are examined experimentally on 2024 aluminium alloy specimens. The simple measure of stop drilling the crack tips restored most of the tensile strength of the plate. To enhance the fatigue life of a cracked plate the process of stop drilling the crack tips, cold expanding the resulting holes and inserting interference-fit steel plugs has dramatically increased fatigue resistance. For the same spectrum fatigue life the stresses in the life-enhanced plates need to be 2.6 times those in the plain cracked plates. This factor remains practically constant in the presence of secondary bending when the average bending-stress/axial-stress ratio is 0.5. Surprisingly, the secondary bending reduced the fatigue life by only about 25% for both unenhanced and enhanced specimens. These results have obvious application to the lower wing skin cracking in the F-111 aircraft.

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