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Historically, helicopter airframe fatigue evaluation has been based on the Safe Life approach. The EH101 airframe cabin and cockpit structure has completed a full-scale factored load fatigue test incorporating high frequency (rotor) and low frequency (manoeuvre) loads. With production improvements, it has demonstrated Safe Lives in excess of 10000 hours for military and civil variants. Consequently, it is one of the best qualified helicopter airframes in service today. As part of the civil certification activity, a damage tolerance evaluation was also carried out, using the Residual Strength After Flaw Growth (Fail Safe) approach. This evaluation drew on lessons learned during recent collaborative research on helicopter crack growth modelling. The unpalatable results of the evaluation - very short inspection intervals - are in sharp contrast to the good results obtained from the Safe Life approach and to the experience in aeroplanes. This paper discusses the reasons for these differences and draws conclusions on the practicality of a crack-growth-based approach to damage tolerance for helicopters.

1. INTRODUCTION

The EH101 is a modern medium/large three engined helicopter which has just entered service with the Royal Navy and will shortly enter service with the RAF. Civil rear-ramp utility variants are also in production. The EH101 has undergone an extensive development programme which included a full-scale factored load fatigue test. Failure modes observed during this test were eliminated from the production standard by design changes. A stand-alone fatigue test of a production standard lift frame, together with analysis using fine mesh finite element modelling, demonstrated the efficacy of the changes. The Safe Life substantiation process is described in detail in [1]. Safe Lives in excess of 10000 hours have been demonstrated for civil and military variants. Consequently, the EH101 has one of the best qualified helicopter airframes in service today. As part of the civil certification activity of the EH101-510 variant, the Airworthiness Authorities also required damage tolerance evaluations of all fatigue critical components to be carried out. This is in keeping with the current thinking of regulatory bodies, who favour a flaw growth damage tolerance approach over Safe Life or even Flaw Tolerant Safe Life approaches. The FAA led Technical Oversight Group On Ageing Aircraft (TOGAA) have stated [2] that "*the primary design goal should be to require a damage tolerant design unless it entails such complications that an effective damage tolerant structure cannot be achieved within the limitations of geometry, inspectability, or good design practice.*" Furthermore, they believe that "*the Flaw Tolerant Safe Life approach should be expunged from the extant*" regulations. The Residual Strength After Flaw Growth approach was used for the damage tolerance evaluation of the airframe main load path.

2. HELICOPTER LOADS

In conventional helicopters, main rotor loads are transmitted to fuselage frames at the main rotor gearbox attachments. These compact, highly loaded sections have complex, three dimensional stress fields with large numbers of blade-passing frequency cycles super-imposed on low frequency manoeuvre cycles. In the EH101 comprehensive flight test programme, these fatigue loadings were measured at many airframe locations, for the complete suite of conditions throughout the flight envelope. A typical load time history is shown in Figure 1. Helicopter airframe high frequency loads are typically at 15-20 Hz (at 17.85 Hz in the case of the EH101), which equates to around 60000 cycles per hour. This is several orders of magnitude more cycles per hour than in the case of fixed wing aircraft. Furthermore, a large proportion of these cycles are at a high stress ratio, R, (where R= minimum stress/maximum stress) of R=0.7 to 0.9, rather than R=0 to 0.1 as is the case in fixed wing spectra. At a given value of stress intensity range, ΔK, this difference in stress ratio means the crack growth rate is an order of magnitude faster, and the threshold value of ΔK is reduced by 75%. This is shown in the da/dN versus ΔK curves plotted in Figure 2. It is noteworthy that the load spectra contain occasional drops in mean load to levels well below those of the majority of the cycles. These under-loads are analogous to the crack retarding over-loads which occur in fixed wing spectra. These differences in load spectra make a fundamental difference to the practicality of crack growth based approaches to airworthiness.

3. CRACK GROWTH MODELS

A number of analytical models are available with which to calculate crack growth. Most are based on linear elastic fracture mechanics combined with different corrections for the effects of crack tip plasticity and crack closure effects. Some models attempt to allow for load interaction effects which can accelerate or retard crack growth. The significance of these effects on crack growth predictions differ between helicopters and aeroplanes. Another factor is that cracks growing in helicopter structures will spend a higher proportion of their lives at small crack lengths compared to aeroplane structures. Therefore, the accuracy of crack growth models in the near-threshold region are of great importance in helicopter applications. Finally, some of the models are 2-dimensional and applicable to through-thickness cracks only. Three dimensional, part-through-thickness models, are more appropriate for the compact section geometries found in helicopter lift frames and beams, to predict crack growth from the corners of flanges, lightening holes etc.

As part of technology development in helicopters, GKN Westland collaborated in a three year joint research programme with Cranfield University, DERA Farnborough and software suppliers nCode International. Various models were evaluated by comparing their crack growth predictions with test data under the following test conditions [3]:

a) constant amplitude;

b) simple variable amplitude (overload/under-load excursions on constant amplitude), and;

c) complex variable amplitude (spectra derived for the project which are representative of airframe and mechanical component loads).

The following models were evaluated: Wheeler, Willenborg, Loseq, Fastran, Kraken, Esacrack and Stripy. The test specimens were made from aluminium lithium and titanium and ranged from compact C-and corner cracked bars to complex structural elements representative of lift frames and rotating components. Some of the models predicted crack arrest when only a delay in crack growth occurred. Some of the models were empirically adjusted to give a good fit to the constant amplitude and simple variable amplitude test results. Despite this, most of the models, including simple linear summation models, gave non-conservative predictions of crack growth under the representative complex variable amplitude load tests. A comparison of various model predictions with test results for complex variable amplitude loading is shown in Figure 3. This was in contrast with the expected result and to the experience in aeroplanes. Some models were non-conservative by a factor of 10. The only exception was the nSoft computer programme, Kraken, possibly due to conservatism in its data fitting rather than the accuracy of the model. The research project concluded that:

a) none of the currently available models is always reliable;

b) the accuracy of model predictions are highly sensitive to the representation of near threshold, high R ratio, da/dN versus ΔK data, and;

c) crack acceleration effects may be very significant with helicopter spectra and it is possible that improved models will need to be developed for helicopter applications.

Kraken, being the only non-conservative model and the one which generally correlated best with the complex amplitude tests, was used for the damage tolerance evaluations.

4. DAMAGE TOLERANCE EVALUATIONS

4.1 Methods of Analysis

Comprehensive strain gauge data through-out the flight envelope are available for many locations on the airframe main load path. A fine mesh finite element model of the main load path was used to relate these strains to the peak local stresses at adjacent features such as lightening holes, cut-outs and bolt holes. Cracks were assumed to exist at these peak stress locations and compliance functions for crack growth were derived using either standard solutions (corner cracks in bars or from holes) or solutions derived from finite element models or structural element tests. Initial flaw sizes were in accordance with [4] and were 1.3 mm in the primary load path and 0.13 mm in the secondary load path. A load spectrum was derived for each analysis location from the appropriate strain gauge data, which included the peak ground-air-ground cycle and all high frequency loading conditions with loads equal to, or higher than, steady cruise. The load spectrum, the compliance function and the appropriate da/dN versus ΔK data were then input into Kraken, to predict crack growth.

4.2 Crack Growth in the Secondary Load Path

Two different options for Residual Strength After Flaw Growth were considered. The first was to assume complete severance of a primary load path and determine the time for a small (0.13 mm) initial crack in the secondary load path to grow to critical length (unable to support Limit Load). This option is attractive because a visual inspection, for the complete separation of the primary load path, would suffice. However, when complete failures of primary load path items were analysed statically on the fine mesh finite element model, it was shown that they produced load increases of 2 to 2.5 times in the secondary load paths. This renders the secondary load path crack growth approach unviable since, with a limit load residual strength requirement and an ultimate reserve factor of 1.0, the maximum permissible load increase would be 1.5 times in the secondary load path. This finding is likely to apply to the majority of helicopters, which have similar structural configurations. Furthermore, a primary load path failure would certainly change the dynamic characteristics of the airframe and, consequently, the high frequency loads in the secondary load path. The changes would be very difficult to analyse and could not be readily substantiated by flight measurements.

4.3 Crack Growth in the Primary Load Path

The second option relies upon detecting cracks in the primary load path before the residual strength of the structure falls below limit load. An initial corner crack of 1.3 mm long was assumed and the crack growth was predicted using the Kraken software as described above. Figure 4 shows a typical crack growth history prediction and Figure 5 shows the airframe structure, which has four main gearbox attachments and machined frames, and the locations which were analysed. However, it was found that the high frequency, high stress ratio, loads exceeded the threshold stress intensity at small crack lengths - despite the EH101's very effective active vibration reduction system, which er. ures low airframe vibration levels and stresses below the safe endurance fatigue limit. This is illustrated in Figure 6 which is a typical plot of the mean and vibratory strains at a location on a main lift frame, for both low frequency (GAG and manoeuvre-to-manouevre range pairs) and high frequency loads. Superimposed on this plot are the working Goodman Diagram endurance lines together with lines showing the crack growth thresholds at the initial (1.3 mm) and inspectable (2.0 mm, by eddy current) crack lengths. The threshold lines have been scaled down by the ratio of the peak stress at the feature analysed and the stress at the strain gauge location. Note how the threshold drops as the R ratio increases. It can be seen that a large proportion of the conditions in the spectrum give high frequency loads which are above the crack growth threshold at the initial crack length. As the crack grows, more conditions exceed the threshold. By the time the crack has reached 2 mm, which is the lowest limit of detectability by eddy current, a very large proportion of the spectrum is above the threshold. It can also be seen, from the location of the Goodman Diagram lines, why good Safe Lives are achieved compared with very poor crack growth inspection intervals. Applying the usual safety factor of 3 on the crack growth period from detectable to critical length, yielded NDT inspection intervals of less than 100 hours at most of the locations analysed. Some were as low as 10 hours.

4.4 Discussion

The crack growth analysis described above for the primary load path was conservative in that maximum reversal loads were used rather than average or cycle-counted loads. Also, no account was taken of load redistribution due to crack growth and load path stiffness reduction. However, these effects are unlikely to yield much benefit since some steady state conditions are above the threshold (and therefore cycle-counting is of little benefit) and little load redistribution will occur at the relatively short crack lengths which become critical at limit load. The conclusion must be that the Residual Strength After Flaw Growth approach to Damage Tolerance is not viable if any part of the load spectrum is above the crack growth threshold. Even if only the low frequency conditions are above the threshold, the crack will grow until the high frequency loads start to produce flaw growth. This will occur at a very modest stress level. There are, however, a small number of situations in which a flaw growth approach may prove viable. Examples include lugs which are designed to relatively low stress levels due to the presence of fretting. These conclusions are not limited to the EH101: recent experience on life extension programmes undertaken for the Westland Lynx and the S61-based SeaKing have also demonstrated inadequate post-failure residual strength and very short crack growth inspection periods.

5. FULL-SCALE AIRFRAME TEST VALIDATION

A full-scale airframe crack growth test was performed to validate the results of the analysis. A production standard roof frame was retrofitted into the development fatigue test article. The test set-up is illustrated in Figure 7. A 4 mm flaw was introduced into the highly loaded upper flange close to the gearbox attachments. Increased amplitude loads were necessary to initiate a crack, but, once started, it grew under representative high and low frequency loads. After 2 million cycles (equivalent to 28 flight hours) the crack had grown to 10.5 mm and fast fracture occurred. The crack arrested at a lightening hole but re-initiated from the other side of the hole after the equivalent of less than one flight hour. This is illustrated in Figure 8.

The test confirmed the following:
a) cracks are very difficult to initiate even from large and severe flaws;
b) the analytical predictions of crack growth were accurate - growth is indeed very rapid under helicopter spectrum loads and inspection intervals of less than 10 hours would be required;
c) complete separation of the primary load path does occur as a result of crack growth ie the crack is not completely arrested or diverted. The transverse main lift frames and the longitudinal beams of the EH101 are of double-I-beam construction and it was thought possible that the centre flange might act as a crack stopper. This would have prevented complete separation of the primary load path and the resulting overloading of the secondary load path leading to viable inspection intervals under a secondary load path approach. Since complete separation occurred this is not viable.

6. FLAW TOLERANT SAFE LIFE

GKN Westland Helicopters are now under-taking a programme of coupon testing to determine the effects of accidental damage and corrosion on the S-N curves of the main load path forging material. The programme comprises three severities of both types of damage, with ten specimens of each used to generate comparative S-N data. This will allow a Flaw Tolerant (Enhanced) Safe Life evaluation to be carried out. The size of the damage will be representative of what maybe found in service and will be at least that which is detectable by close visual inspection. Coupon damage levels will be consistent with the corrosion damage that could occur during the inspection period defined in the maintenance manual. GKN Westland Helicopters believe that this approach will prove to be a viable option for airframe "damage tolerance".

7. LESSONS FROM SERVICE EXPERIENCE

The crack growth results presented above might suggest that airframe cracks will grow rapidly and result in catastrophic failure. This is not the case in service, due to a number of factors. Firstly, the manufacture and corrosion prevention measures of helicopter components are of a high standard and helicopter manufacturers establish inspection schedules based on experience and good engineering practice. In this situation, the Safe Life approach has historically proved to be adequate, particularly in view of the fact that most helicopter manufacturers adopt a conservative approach to Safe Life calculation. Secondly, most accidental damage is not characterised by a crack; at worst it is a sharp notch and a crack must still initiate. In fact, on EH101 structural element and full-scale component crack growth tests, it proved difficult to initiate cracks from 3 mm deep saw cuts, which had been sharpened with a scalpel, and then etched. Finally, and perhaps most importantly, limit load conditions are rare in helicopter usage; For example, one of the worst limit load conditions for the EH101 lift frames, starts with a dive at a speed 20% above maximum cruise speed (Vne), followed by a rolling pull-out where 2.4 g is achieved when the speed is still 1.1 Vne. Normal operating loads are lower than this and can be withstood with a cracked lift frame. Furthermore, accidental damage rarely occurs at the point of highest stress but this is the assumption which is made for damage tolerance evaluation.

8. THE EFFECT OF MANDATORY FAIL-SAFETY ON HELICOPTER DESIGN

It is worth considering the effect on helicopter airframe design, that would result if the Fail Safe (Residual Strength After Flaw Growth) approach were to be mandatory. Metallic structures would need to be made more redundant with higher part counts and increased weight, designed to limit-load-with-an-element-failed cases as well as to ultimate load with an intact structure. This might result in more complex aircraft lay-outs, for example six gearbox feet, rather than the normal four. Composite airframes might become a more attractive option because they are inherently more damage tolerant, although there are still issues regarding residual strength demonstration to be resolved. Since some older aircraft types lack even a thorough airframe Safe Life evaluation, these design solutions represent a considerable penalty to new aircraft competing with them.

9. CONCLUSIONS

The conclusion to be drawn is that the Residual Strength After Flaw Growth approach to Damage Tolerance is not viable for helicopter airframes. The key cause of this result, which is quite different from aeroplanes, is the presence of high numbers of stress cycles at blade passing frequencies, at high stress ratios and above the threshold stress intensity for flaw growth. This means that relatively small cracks grow very rapidly and impracticably short inspection intervals would be required. The high frequency stresses are, however, below the working fatigue endurance level and therefore produce no damage in the Safe Life approach. Helicopter service experience indicates that a more onerous inspection regime is not warranted. This paper has not addressed dynamic components, but it is likely that the conclusions drawn for the airframe will apply equally to them.

The Flaw Tolerant ("Enhanced") Safe Life approach may well be the only viable approach available to helicopter manufacturers to demonstrate Damage Tolerance, within the limitations of geometry and good design practice, and so this should be retained in any future regulations.

10. REFERENCES

1. Overd, M. L., "EH101 Airframe - The Fatigue Testing And Resulting Timely Modifications For The Production Aircraft", Proceedings Of The 19th Symposium Of The International Committee On Aeronautical Fatigue, 1997, Volume 1, pp. 346-364.

2. TOGAA., "Memo To The Rotorcraft Working Group Subject: The White Paper", April 13 1998.

3. Cook, R., Graaskov, D., and Chilton, A., "Robust Crack Growth Models For Rotorcraft Metallic Structures - Final Report - Crack Growth Models" DERA Reference DERA/MSS2/WP980180/1.0.

4. Mil Spec MIL-A-87221, "USAF General Specification For Aircraft Structures", 28.02.85, Table XXVI.

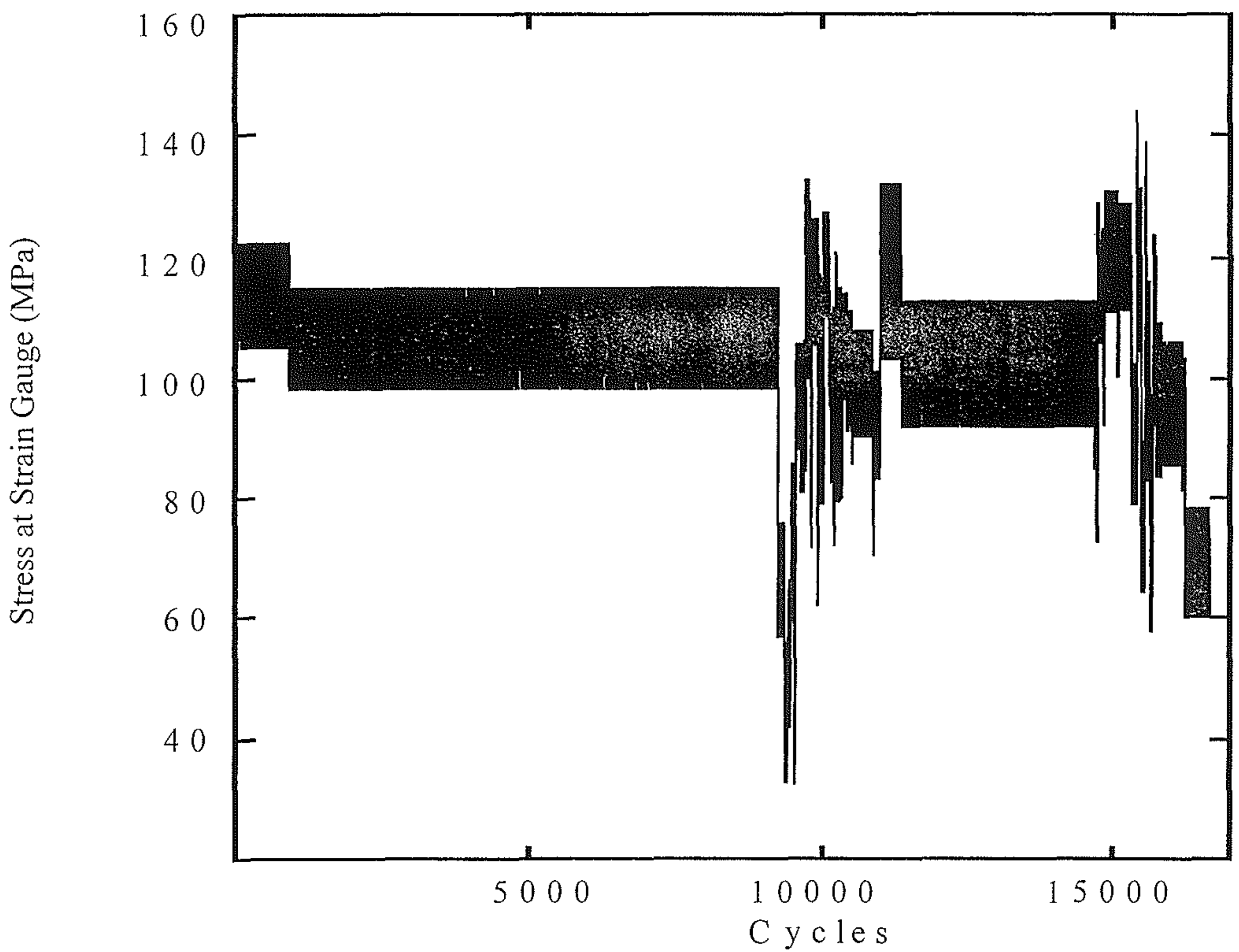


FIGURE 1 TYPICAL HELICOPTER LOAD SPECTRUM

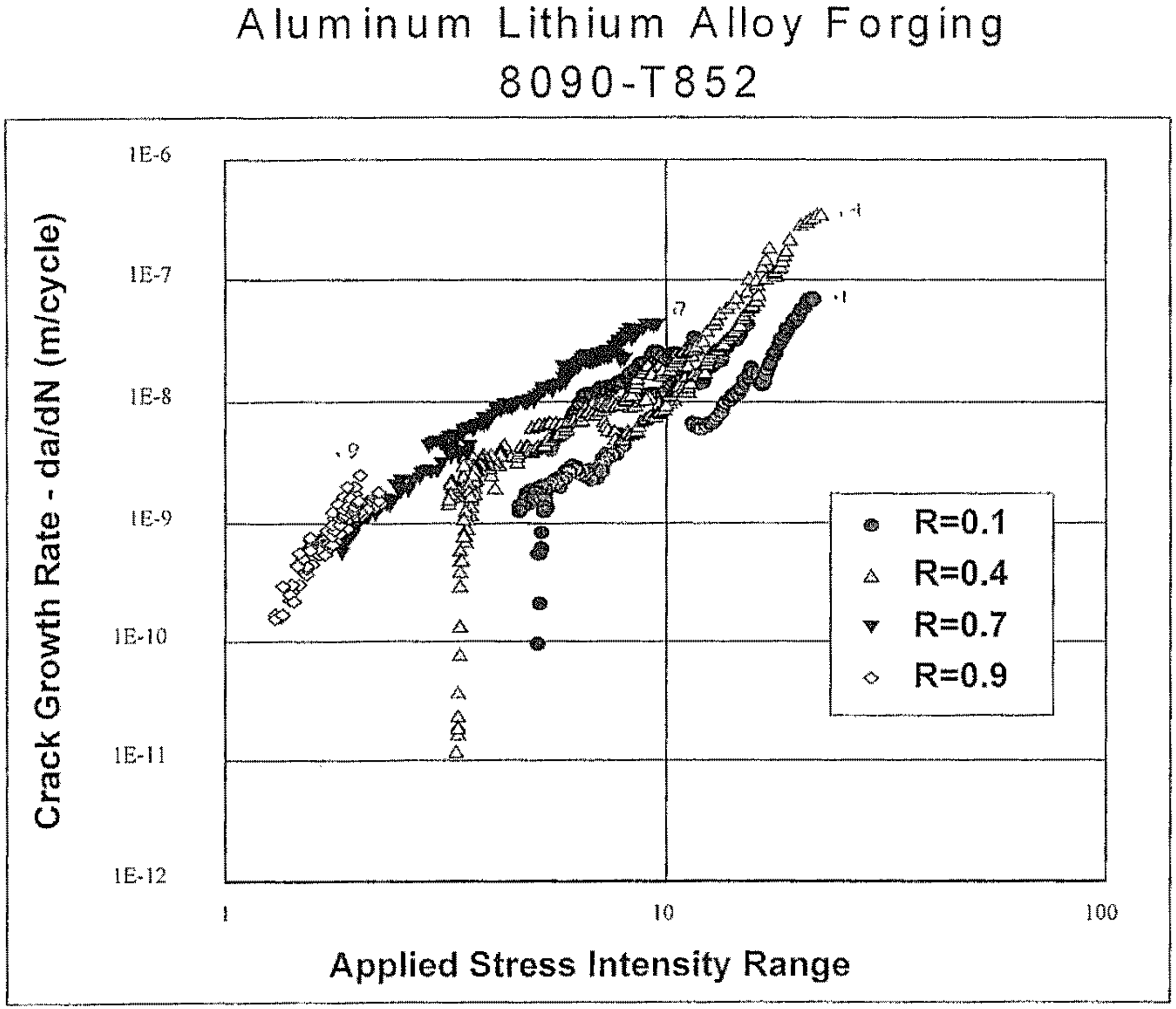


FIGURE 2 ALUMINIUM LITHIUM CRACK GROWTH DATA

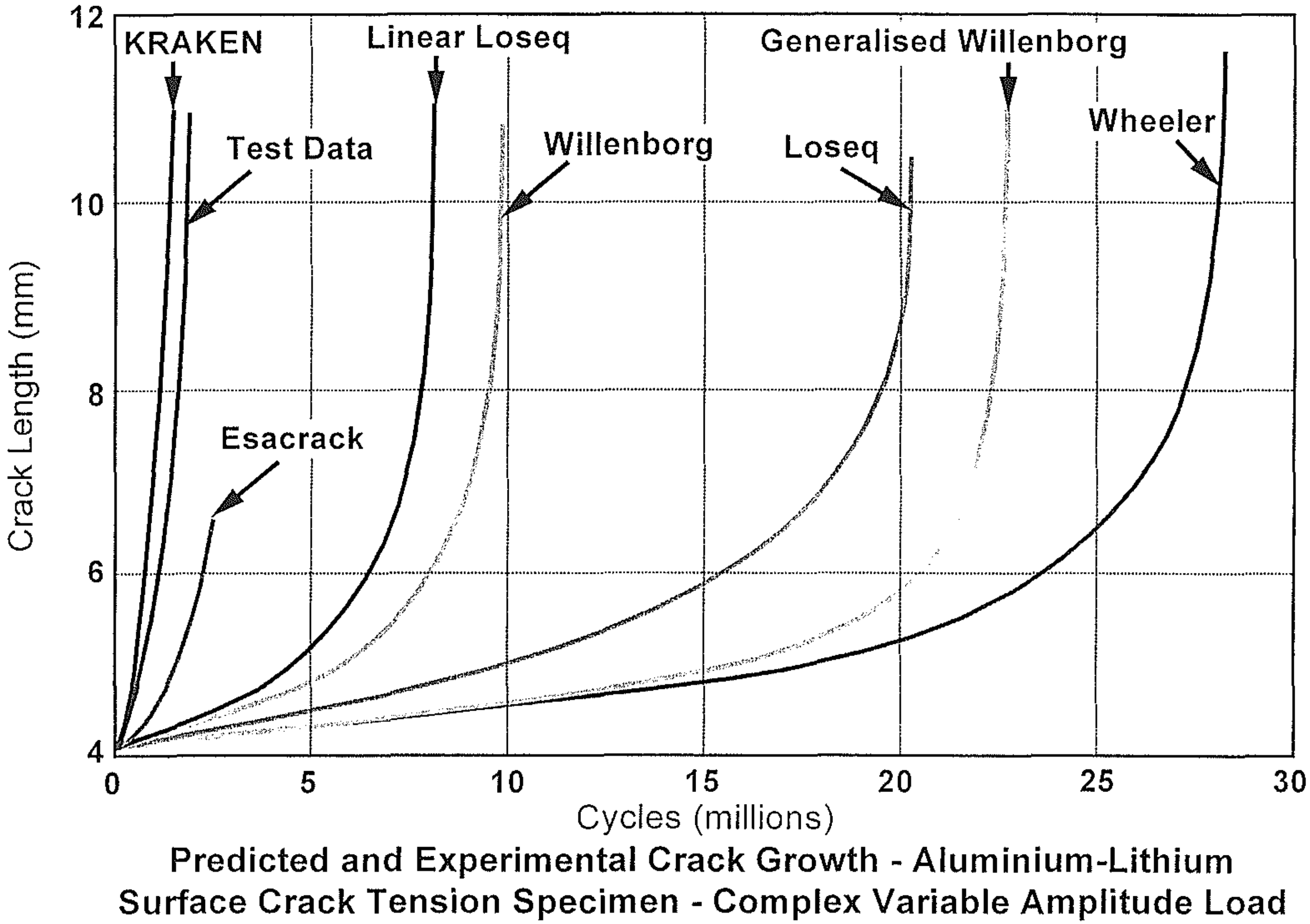


FIGURE 3 COMPARISON OF CRACK GROWTH MODELS AND TEST DATA

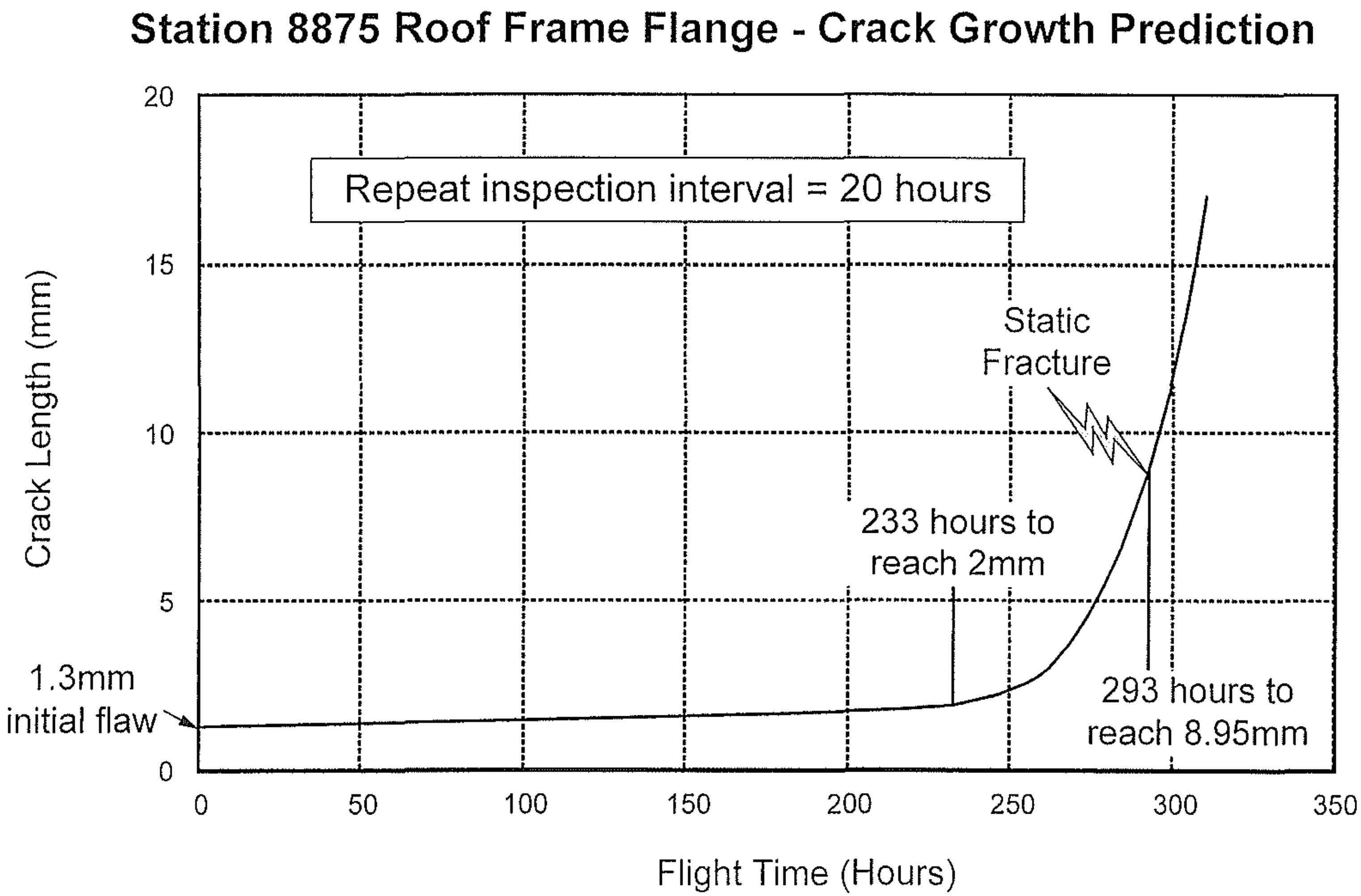
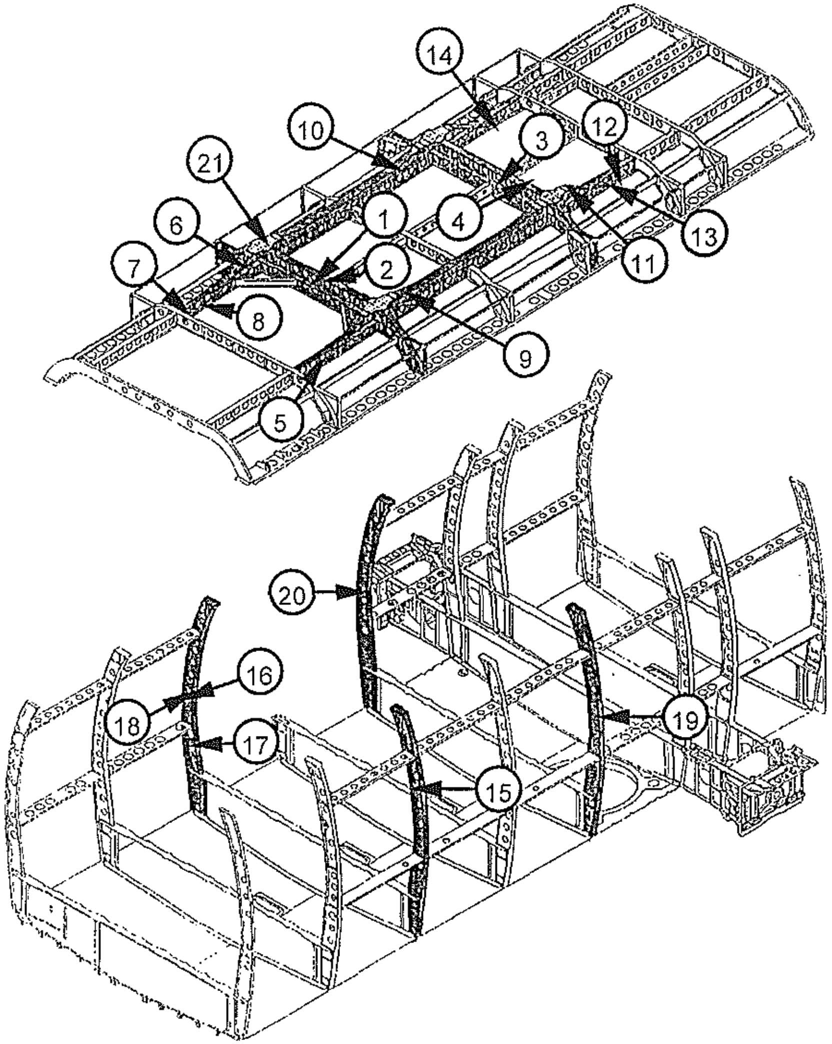


FIGURE 4 TYPICAL CRACK GROWTH HISTORY PREDICTION



Location	Inspection Interval		Life to
	Threshold	Repeat	Failure
1	9	20	89
2	8	11	59
3	459	534	2980
4	77	20	293
5	Greater than starboard		
6	16	4	65
7	181	341	1569
8	62	17	242
9	26	13	244
	Greater than starboard		
	Greater than port		
10	6	8	45
11	1.8	2	13
12	5	10	49
13	27	4	98
14	Greater than port		
15	Greater than Sta 6875 stbd. side frame		
16	427	597	3073
17	14	6	61
18	3	5	27
19	Greater than Sta 6875 stbd. side frame		
20	Greater than Sta 6875 stbd. side frame		
21	130	965	3286

FIGURE 5 AIRFRAME LOCATIONS ANALYSED AND RESULTS

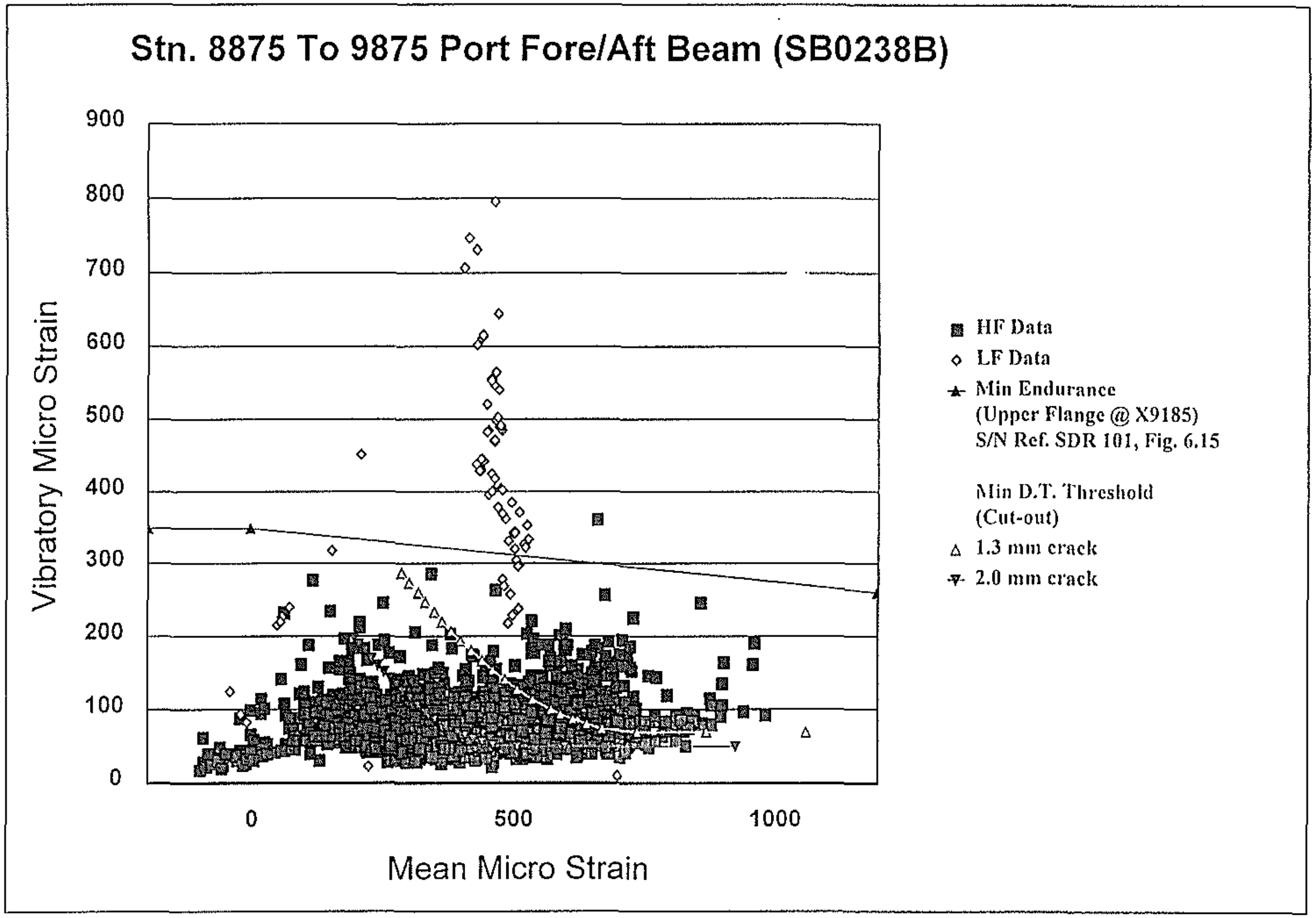


FIGURE 6
MEAN AND VIBRATORY STRAINS COMPARED WITH
THRESHOLD AND GOODMAN LINES

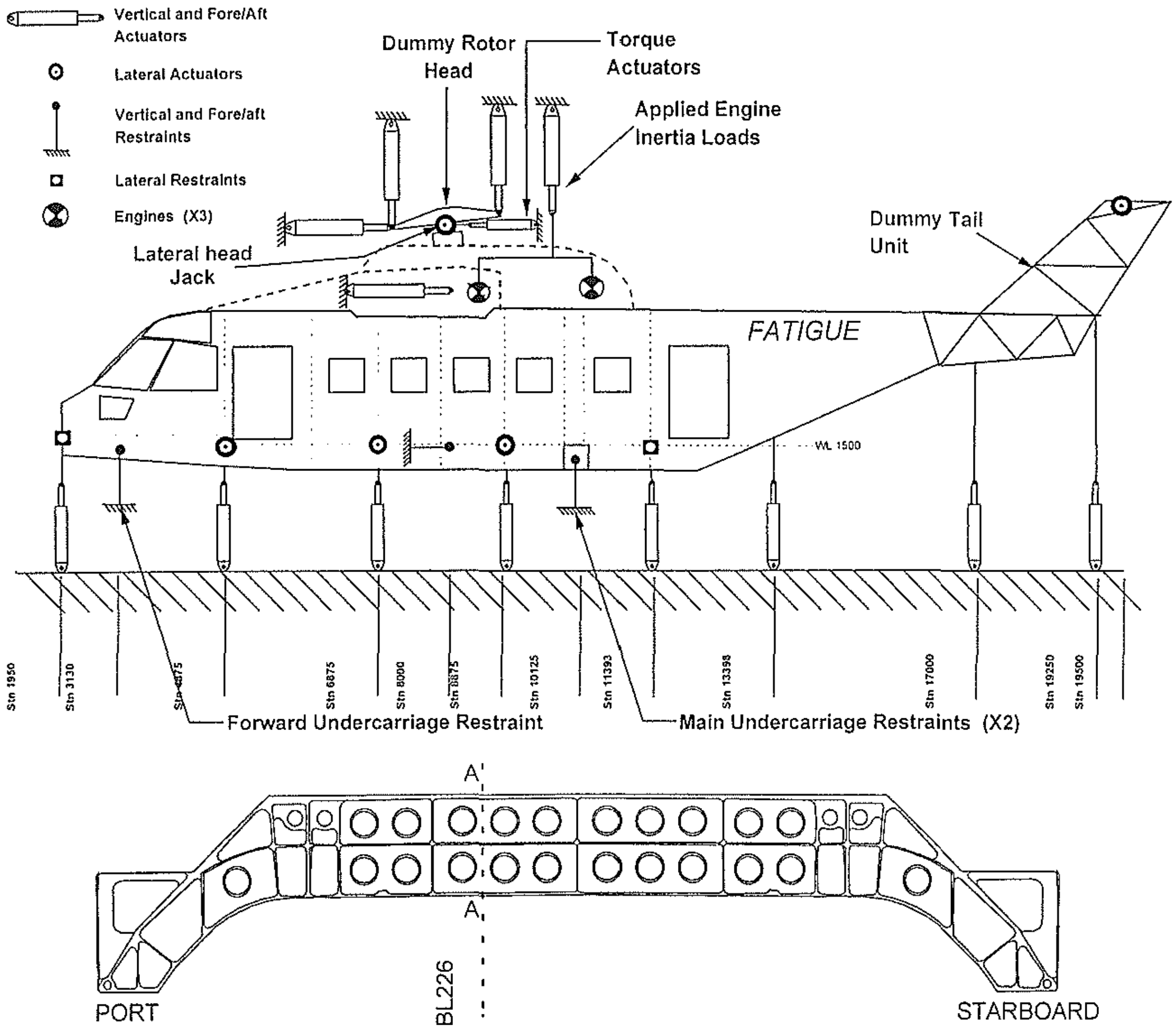


FIGURE 7 THE FULL-SCALE AIRFRAME DAMAGE TOLERANCE TEST

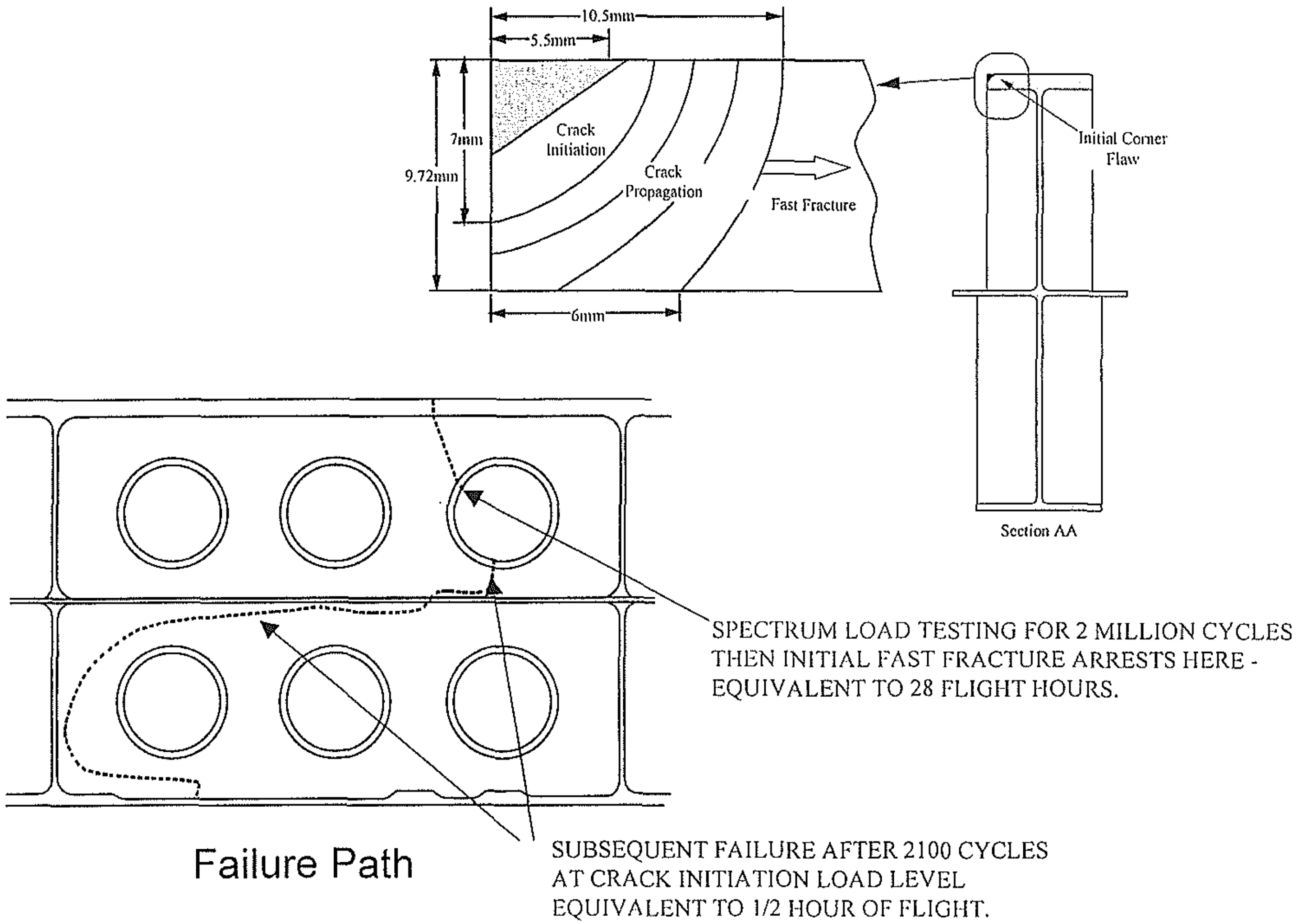


FIGURE 8 AIRFRAME TEST CRACK BEHAVIOUR