SIGNIFICANCE OF A ROTOR BLADE FAILURE FOR FLEET OPERATION, INSPECTION, MAINTENANCE, DESIGN AND CERTIFICATION

by

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CONTENTS

1 INTRODUCTION

Catastrophic fracture of a rotor blade from a large helicopter (Refs. 1 and 2) was analysed, with essentially immediate consequences for continued fleet operation, inspection and maintenance of the same type of rotor blade in the Netherlands.

These consequences were:

- a speed restriction to reduce blade working stresses
- installation of an in-flight blade inspection system
- greater emphasis on repair of debonding (looseness) in the blade spar/pocket joints.

Secondly, it became clear that several aspects of the design of such rotor blades (hollow spars with bonded pockets) should be reconsidered, namely:

- choice of adhesive bonding system for the spar/pocket joint
- choice of pressurizing gas for leak (and hence crack) detection in the hollow spar.

Thirdly, perhaps the most significant conclusion, which is relevant to all rotorcraft manufacturers as well as airworthiness authorities, was that a general re-evaluation of the design philosophy and procedure for critical helicopter items such as rotor blades is needed with regard to:

- the magnitude of the working stresses
- estimation of safe fatigue initiation and crack growth lives from laboratory fatigue tests
- individual operator usage differing significantly from a so-called typical design use.

In this paper the service failure investigation is reviewed briefly to serve as an essential illustration for subsequent discussion of (1) remedial action for similar types of rotor blades, and (2) design aspects, philosophy and procedure.

2 THE SERVICE FAILURE

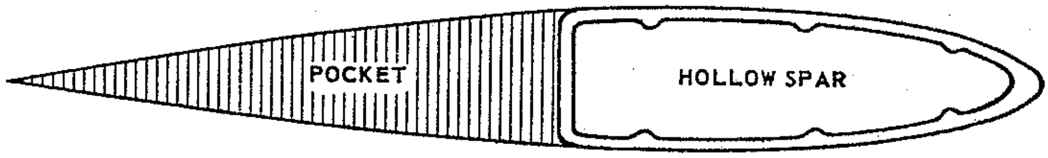
Some years ago a large helicopter crashed into the North Sea. The helicopter was subsequently recovered from 52 metres of water.
All main rotor blades were broken, and outboard sections of the blades could not be recovered. One rotor blade showed little deformation at the fracture location.

This particular blade was submitted to the NLR for fractographic and chemical analysis and mechanical property tests. The chemical composition and mechanical properties of the blade spar were found to be within the manufacturer's specifications for the aluminium alloy extruded spars concerned.

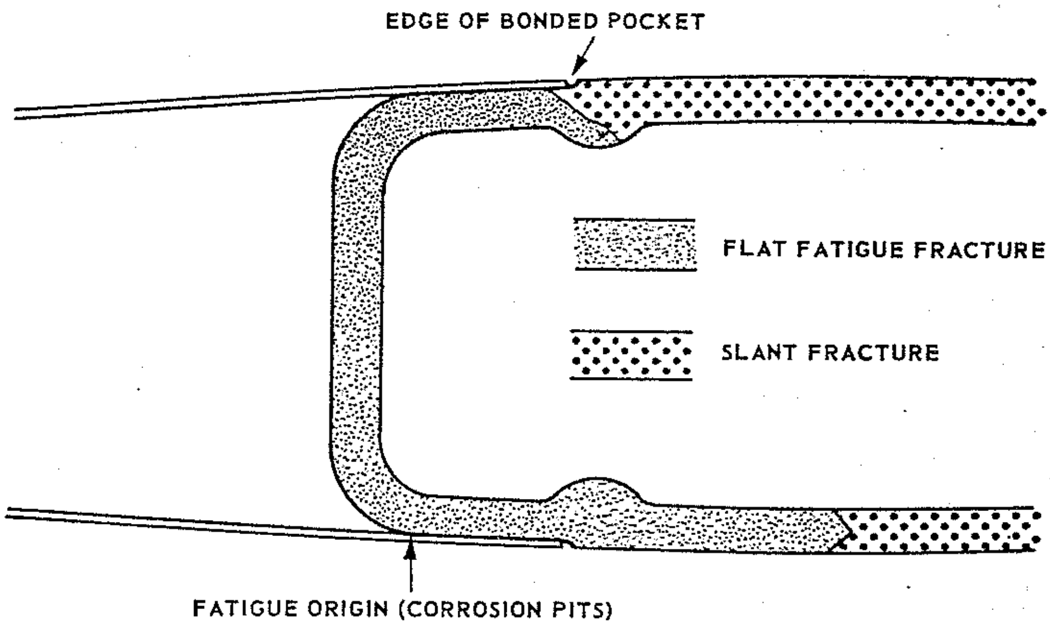
Fractography indicated that the blade spar had failed by fatigue originating from one or two corrosion pits under the spar/pocket joint, as shown schematically in figure 1.

2.1 The blade inspection system

As indicated in figure 1 the rotor blade consists of a hollow spar plus bonded-on pocket. Although the rotor blades are designed and certified (see next section) as safe life components, the helicopter manufacturer had gone one step further and had introduced a blade inspection method. In doing this the philosophy of safety by inspection had become applicable to these originally safe life rotor blades. For this purpose the hollow spars are pressurized and cracking is detectable as a pressure loss. In its original form this system comprises pressure monitoring by indicators installed on each rotor blade. The indicators are checked during pre-flight inspection



(a) ROTOR BLADE CROSS-SECTION



(b) SPAR SERVICE FAILURE

Fig. 1 Schematic of the rotor blade construction and the service failure when the rotor is stationary. For spar cracking to result in a pressure loss a through-thickness crack must develop, and if such a crack lies under the bonded pocket then the adhesive bond must fail, and probably also the pocket skin.

2.2 The spar fatigue life

The manufacturer had performed constant amplitude fatigue initiation life tests on inboard and outboard spar sections. Working curves fitted to the data were used with the well-known Palmgren-Miner cumulative damage hypothesis to calculate fatigue initiation times for variable amplitude loading representing service use. This procedure was allowed by the FAA's Civil Airworthiness Manual no. 6 (CAM 6), Appendix A: "Main Rotor Service Life Determination". It appeared that the outboard spar had a far longer fatigue life than the inboard spar. Based on these calculations and upon application of the usual safety factor, an initial blade retirement life of 7000 hours was recommended. This retirement life was subsequently increased to 9400 hours, and this increase was certified by the FAA.

Figure 2 shows spar fatigue failures from the same type of helicopter up to the time of the crash. All the other failures were detected non-catastrophically by the blade pressurization system. The figure shows the causes of fatigue are various and more or less unpredictable, but it is evident that failure is not a very rare event. Also, the lives are almost uniformly distributed over the range 400-7000 hours. Hence it must be concluded that the FAA-approved fatigue tests, analysis and blade retirement time (which followed CAM 6 procedures, and did not account for possible defects) are not relevant to service operation.

2.3 The spar crack propagation life

Besides fatigue initiation life tests, the manufacturer had also conducted constant amplitude crack propagation life tests on inboard spar sections using combined loading, consisting of a steady centrifugal load of 200 kN (mean stress of 74 MPa) with superimposed vibratory loads. The results, in terms of crack propagation life from a pressure drop indication to complete failure, are given in figure 3, which also shows the working curve used with the Palmgren-Miner hypothesis to calculate the time available from a pressure drop indication to complete failure under service spectrum loading. This calculated time was of the order of several hundreds of flights and it was concluded by the manufacturer that the rotor blade therefore possessed a high degree of crack detectability, since the inboard spar was more susceptible to fatigue initiation than other parts of the blade.

The crack propagation life of the spar service failure, which was in the outboard section, was estimated from extensive fractographic investigation by the NLR (on behalf of the Dutch Aircraft Accident Inquiry Board) and the manufacturer. The total life of the spar can be divided into five stages, as shown in figure 4. These stages are:

- period 1 : time to fatigue initiation
- period 2 : crack propagation to through the thickness
- period 3 : propagation to pocket edge
- period 4 : visible flat fatigue fracture
- period 5 : visible slant fracture, comprising both fatigue and final fracture.

Periods (2) - (5) were estimated to be 4.6 hours, 3.3 hours, 1.5 - 3.7 hours and about 0.6 hours successively, giving a total crack propagation life of 10 - 12.2 hours. The maximum possible time for crack detection by the blade pressurization system, i.e. the sum of periods (3) - (5), is then 5.4 - 7.6 hours. This is very much shorter than the above-mentioned

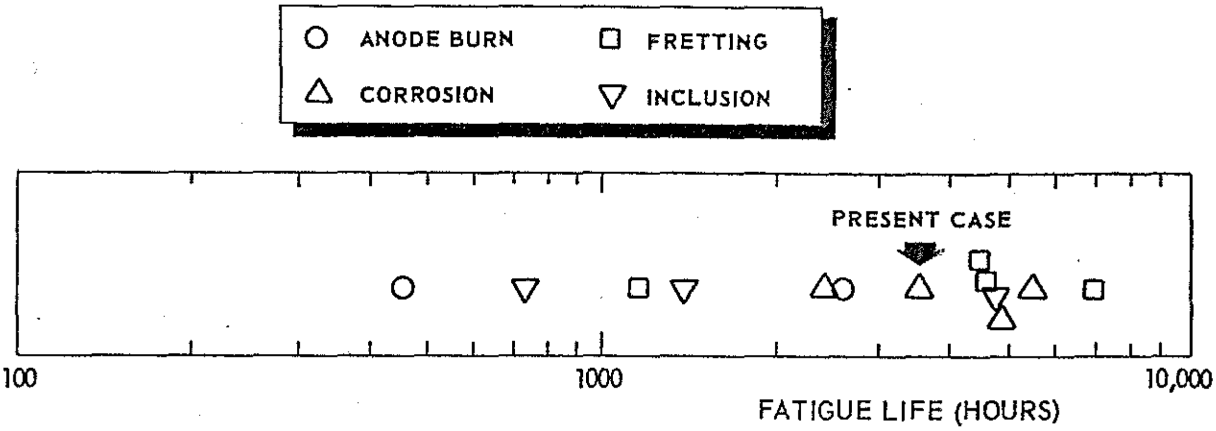


Fig. 2 Spar fatigue failures in service up to the time of the crash

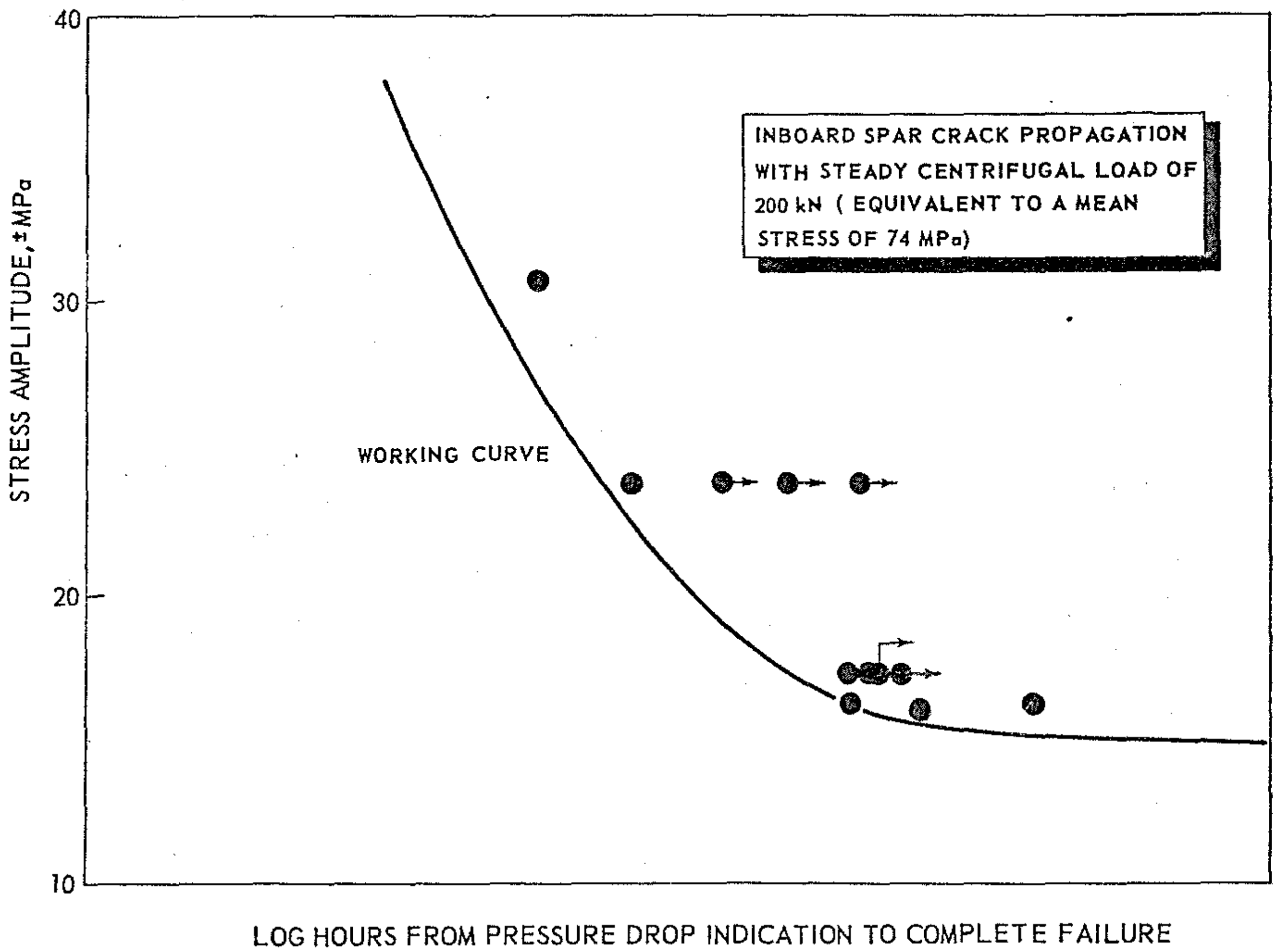


Fig. 3 Fatigue crack propagation tests on inboard spars

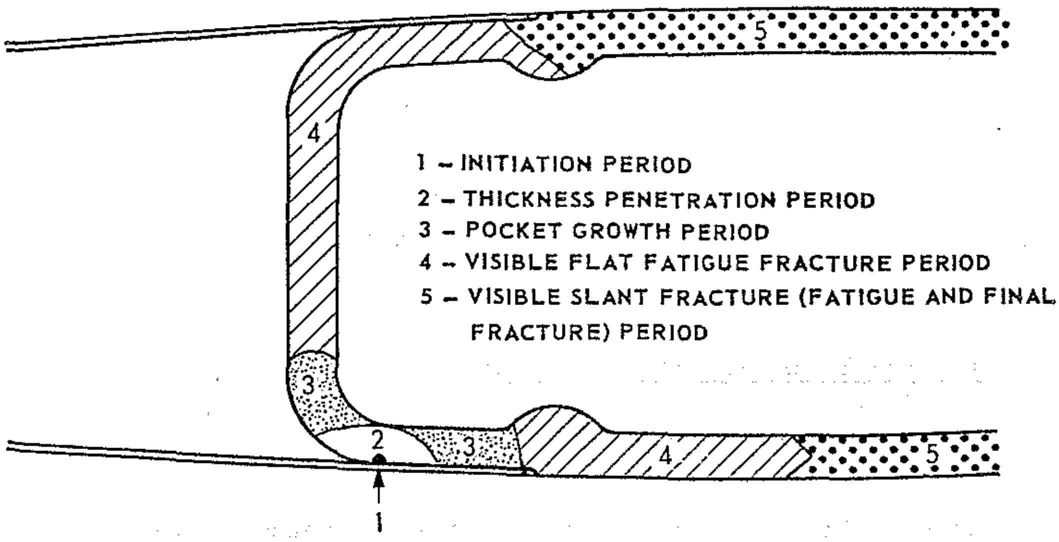


Fig. 4 Phases of the spar fatigue life

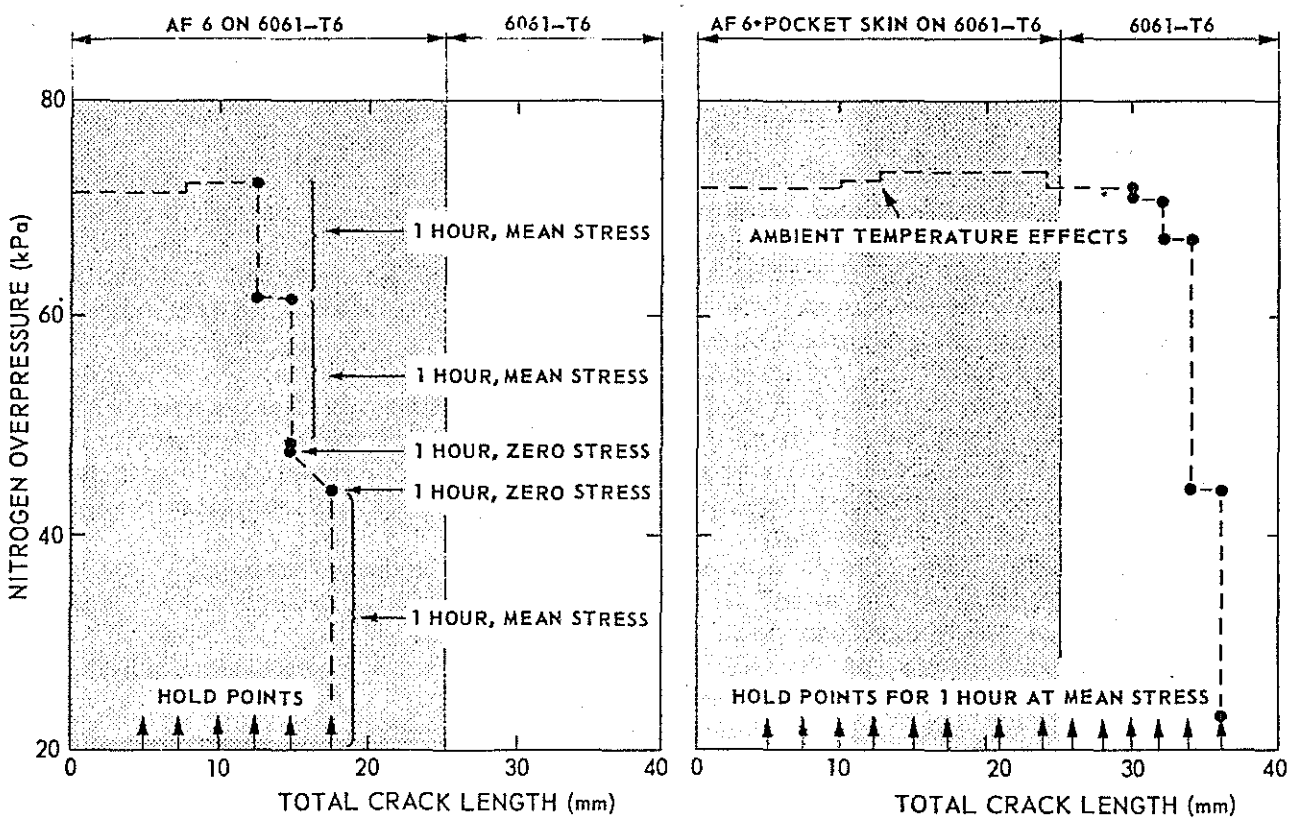


Fig. 5 Fatigue crack propagation + pressurization test results

calculated time of several hundreds of flights, which was for a presumably more critical location.

In view of the short crack propagation life estimated for the service failure, a flight simulation test on a prenotched spar section similar to that of the service failure was done by the manufacturer. The test consisted of applying a steady centrifugal load of 150 kN (mean stress of 55 MPa) with a programmed sequence of three stress amplitude conditions representing three flight conditions, viz. (1) take off, climb and hover (2) cruise at high speed and (3) approach and other transients. It turned out that crack growth in the simulation test was faster than that estimated for the service failure. Assuming the simulation test stress levels to be realistic, then it is unlikely that the fractographic measurements for the service failure overestimated the crack growth rates, i.e. the short crack propagation life for the service failure was confirmed by the flight simulation test.

2.4 Life available for crack detection

As stated in section 2.3, the maximum possible time for crack detection by the blade pressurization system was estimated to be 5.4 - 7.6 hours for the service failure. However, the actual time for crack detection may have been less, for three reasons:

- the adhesive bond and pocket skin at the spar/pocket joint may not have failed with the spar, so that pressure loss would have commenced only when the crack grew beyond the pocket edge, i.e. only during periods (4) and (5)
- the pressure indicator on the blade could only be inspected on the ground, with the blade stopped and effectively unloaded. There is therefore a strong possibility that a fatigue crack propagating under positive stress ratios would be closed and leak-proof in the ground condition, even if the adhesive bond had cracked or if the spar crack had grown beyond the pocket edge. The pressure loss during the previous flight could have been within permissible limits, owing to the basically high crack propagation rates
- NLR tests showed that a low reference pressure in the indicator (e.g. owing to leakage, which can occur in service) could result in the indicator remaining too long in the SAFE position.

To check the first two possibilities the NLR carried out fatigue crack propagation + pressurization tests on specimens simulating the service failure spar/pocket joint. Centre-cracked panels (2 mm pre-crack) were covered with patches of AF 6 (3M) adhesive and AF 6 adhesive + pocket skin dimensioned to correspond to the configuration of the spar service failure. The panels were tested at 69 $\pm$ 34.5 MPa with a nitrogen overpressure on the bare panel side. The cycle frequency was 37 Hz. The results were reported to the Dutch Aircraft Accident Inquiry Board. Examples of results are given in figure 5, which shows that

- relatively long cracks are necessary before leakage occurs
- for the panel with an AF 6+ skin patch leakage commenced only when the crack in the panel was well beyond the patch
- the crack length at which leakage begins depends markedly on whether the hold is at mean or zero stress
- there was negligible leakage during cycling between hold points.

It is thus entirely reasonable to consider that the time available for crack detection might be only periods (4) and (5) (see section 2.3). For the spar service failure the total estimate for these periods is 2.1 - 4.3 hours. This is a dangerously short time, since catastrophic failure could occur during a relatively long flight (maximum about 3 hours) following a nominally satisfactory pre-flight blade pressure indicator inspection.

2.5 Overview

Investigation and analysis of the outboard spar service failure revealed that
- failure was by fatigue fracture originating from corrosion pitting
- the lives of service failures were not predictable from the original fatigue tests and analysis approved by the FAA
- the fatigue crack propagation life was much shorter than the predicted life for the inboard spar, which was presumably a more critical section
- the life available for crack detection might be dangerously short.

3 REMEDIAL ACTION FOR SIMILAR ROTOR BLADES

Following the North Sea crash the Dutch Department of Civil Aviation (RLD) imposed a speed limit of 51 m/s (100 knots) for helicopters of the same type. Using the fractographic estimates of crack propagation rates in the spar service failure, the NLR made a fracture-mechanics based calculation of the increase in crack propagation life for periods (4) and (5), i.e. crack growth beyond the pocket edge, as a function of reduced airspeed. The results were reported to the Dutch Aircraft Accident Inquiry Board and are summarized in figure 6. There are two curves, owing to uncertainty as to whether the actual airspeed during the last flight was 120 or 130 knots (increasing airspeed results in higher blade stresses). Figure 6 shows that a speed limit of 51 m/s (100 knots) gives about a fourfold increase in crack propagation life. This increase brings the minimum estimate of the crack detection life (2.1 hours in section 2.4) well beyond the maximum possible flight time of about 3 hours.

As further actions to improve flight safety, dual pressure indicators were installed on the rotor blades and the RLD requested that cockpit blade pressure indicators, which were already under development by the manufacturer, be introduced as soon as possible. These indicators were installed soon after the accident in all helicopters of the same type flying in the Netherlands, and the speed limit of 51 m/s was removed. Current operation is with no speed restriction until a cockpit indication of blade pressure drop, at which time the speed must be reduced to 46 m/s (90 knots).
It should be noted that this procedure is in addition to pre-flight inspection of the blade pressure indicators installed on the rotor blade roots.
In view of the availability and satisfactory working of cockpit blade pressure indicators, it is in our opinion regrettable that installation of these indicators has not been made mandatory by other airworthiness authorities.

Lastly, the RLD amended the procedure following a spar/pocket joint inspection and the determination of debonding (looseness) in the joint: such debonding could lead to moisture entrapment and corrosion, as was probably the case for the spar service failure. The manufacturer had set permitted tolerances for debonding, and before the North Sea crash the detection of debonding placed no restriction on operation provided that the debonding remained within these tolerances. The post-crash RLD directive requires that the debonding area be sealed with High Speed aluminium foil tape and repaired within two months of detection.

4 DESIGN ASPECTS FOR SIMILAR ROTOR BLADES

In this section the topics of particular relevance to the structural concept of hollow spars with bonded pockets are discussed, namely the choice of adhesive bonding system and pressurizing gas.

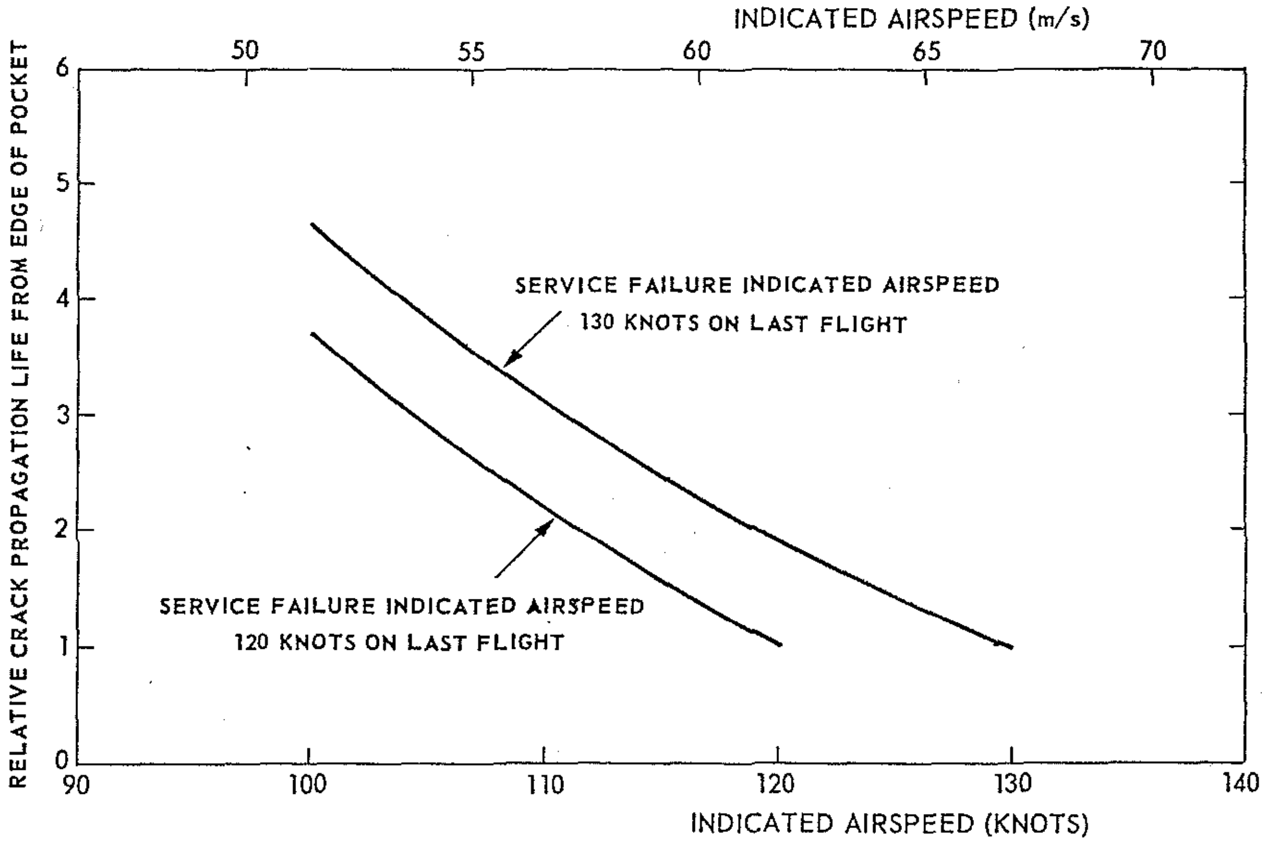


Fig. 6 Relative crack propagation life from the edge of the pocket as a function of airspeed

4.1 <u>The spar/pocket adhesive bonding system</u>

The adhesive bonding system used for the spar/pocket joint in the service failure was EC 1290 primer with AF 6 adhesive. The adhesive was known to be flexible, so that there was the possibility that the bond and pocket skin may not have cracked with the spar, as mentioned in section 2.4, particularly if the bond had loosened.

Investigation of different adhesive systems for the spar/pocket joint was done independently by both the NLR and the manufacturer. As mentioned in section 2.4 the NLR tested centre-cracked specimens covered with patches and subjected to constant amplitude loading, while the manufacturer tested blade sections under flight simulation loading. There were four adhesive systems:

- EC 1290 primer + AF 6
- EC 1290 primer + AF 6 + AF 30 anti-chafing strips
- EC 3917 primer + AF 126 (NLR only)
- EC 3917 primer + AF 126 + AF 30 (manufacturer only).

The tests indicated that the use of AF 126 adhesive instead of AF 6 increased the likelihood of the bond and pocket skin cracking with the spar, and that there was therefore an improvement in crack detectability by pressure monitoring. The increased likelihood of bond and pocket skin cracking when using AF 126 is attributable to the greater rigidity and higher strength as compared to AF 6.

It is to be noted that AF 126 had already supplanted AF 6 in the manufacturing and overhaul procedure as early as 1971. Thus the fatigue tests retrospectively confirmed the usefulness of this change.

4.2 The pressurizing gas

The manufacturer informed the RLD some months after the accident that replacement of nitrogen by helium in the spars was being considered, and that this would result in a much shorter time for a clearly visible pressure drop indication (owing to presumably easier escape of helium through a crack in a spar). Fortunately, as will be shown, this change was not made.

The idea of substituting helium for nitrogen is based on the smaller molecular weight and diameter of helium, which would therefore be expected to leak away much faster than nitrogen. However, this expectation is valid only for the regime of molecular flow and the transition regime between molecular flow and viscous flow. In the regime of viscous flow the viscosity is the controlling parameter.

Because of the possibility that leakage might commence only when a crack grows beyond the pocket edge (see section 2.4) a check as to the appropriate gas flow regime was made. With the parameters listed in table 1 it turned out that except for the very first second of crack growth beyond the pocket edge the leakage occurs in the viscous flow regime. Analytical computation of the pressure drop times with nitrogen or helium as the pressurizing gas where then carried out (Ref.3). Results are shown in figure 7: irrespective of the crack shape factor (i.e. whether the crack may be approximated by a cylindrical hole, a narrow slit,-or some intermediate geometry) it takes longer for the pressure to drop when helium is used, owing to its higher viscosity. Incidentally, considering that the crack shape factor is ~ 0.1 for most of the pressure drop time (Ref.3), then the theoretical values in figure 7 agree very well with the experimental value of 675s found for nitrogen pressurization in the aforementioned NLR tests.

Alternatively, with the improved adhesive system AF 126 the bond and pocket skin may crack with the spar. In this case the relevant distance between the crack faces approximates the total crack opening displacement (COD) rather than the crack tip opening displacement (CTOD), which governs the leakage in the case of a crack growing from under the edge of the nominally intact pocket. Since the COD is much larger than the CTOD, gas leakage is likely to be in the viscous flow regime even at fairly short crack lengths, so that as before there may be no benefit from changing to helium pressurization.

There is in fact, a significant disadvantage in using helium instead of nitrogen. The quality of blade sealing must be much better, since otherwise helium will leak far more readily (molecular flow) through the very small defects in seals, resulting in many more false indications of blade pressure drop and a greater disruption of service operation.

5 GENERAL DESIGN PHILOSOPHY AND CERTIFICATION ASPECTS

In this section there is a general discussion of the magnitude of rotor blade working stresses, the estimation of safe fatigue initiation and crack growth lives, and the consideration of individual operator usage differing significantly from typical design use.

5.1 Rotor blade working stresses

Structural design procedures are basically iterative and one or more stages are frequently omitted owing to experience with previous, similar designs. Thus it is often difficult to determine with certainty whether, for example, the dynamic working stresses mainly define the service life or vice versa.

TABLE 1

Parameters required for spar pressure drop analysis

spar wall thickness and internal volume	= 5 mm and 5 x 10^{-2} m^3, resp.
initial spar pressure (indication SAFE)	= 1.68 x 10^5 Pa
final spar pressure (indication UNSAFE)	= 1.44 x 10^5 Pa
ambient temperature	= 295 K
viscosity of nitrogen at 295 K	= 1.75 x 10^{-5} s.Pa
viscosity of helium at 295 K	= 1.95 x 10^{-5} s.Pa
molecular diameter of nitrogen	= 3.13 x 10^{-10}m
molecular diameter of helium	= 2.66 x 10^{-10}m
molecular weight of nitrogen	= 28 kg/kmol
molecular weight of helium	= 4 kg/kmol
Boltzmann's constant	= 1.38 x 10^{-23} J/K
universal gas constant	= 8.31 kJ/kmol/K
yield stress and elastic modulus of spar	= 260 MPa and 70 GPa, resp.
mean stress in flight	= 55 MPa
initial half crack length (one tip at pocket edge)	= 20 mm
crack growth rate derived from manufactur's data	= 2.84 x 10^{-4} x (half crack length in mm)$^{1.13}$ mm/s
crack tip opening displacement (CTOD)	= 6.65 x 10^{-4} x (half crack length)
1 ≥ crack shape factor (includes tortuosity) ≥ 0	

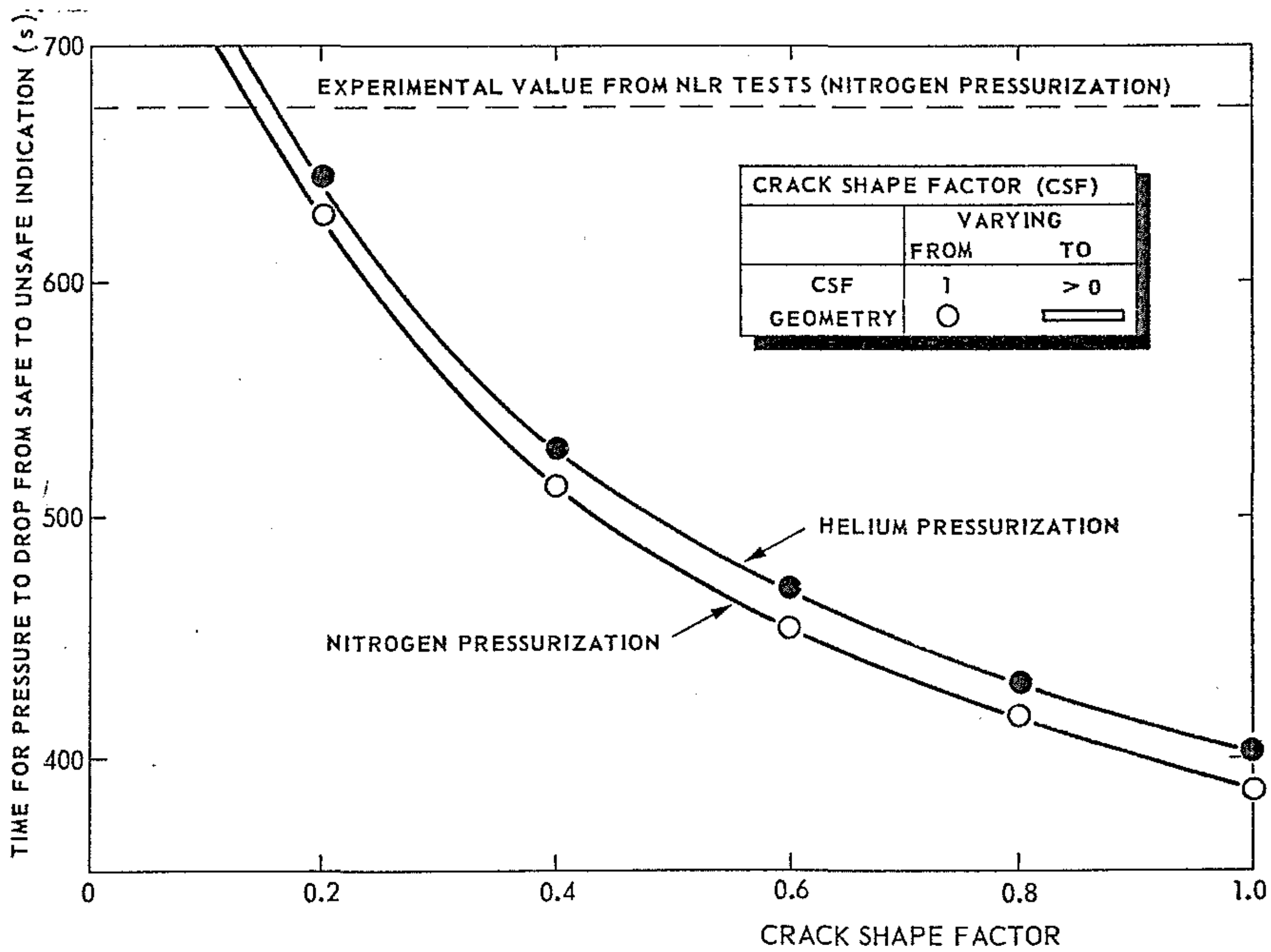


Fig. 7 Comparison of pressure drop times using nitrogen or helium gas to pressurize the rotor blade spar

For the helicopter type discussed earlier in this paper it appears that the rotor blades were sized to meet static strength and stiffness criteria and that the consequent dynamic stresses were used for the fatigue tests, analysis and blade retirement time. It is, however, reasonable to assume that experience had shown a satisfactory defect-free fatigue life should be obtainable with blades of dimensions approximating the size meeting the static criteria. Herein lies the crucial problem with the applied safe life design and certification philosophy, which was once the general trend amongst all aircraft constructors and airworthiness authorities, but is now being gradually replaced by the philosophy of safety by inspection (damage tolerance). In components dimensioned according to the safe life philosophy fatigue crack initiation is emphasized and thus the dynamic working stresses will result in adequate fatigue life only in the absence of defects. But if these components are susceptible to defects no reliance can be made anymore on defect-free fatigue tests and component retirement times. Furthermore, the occurrence of defects need not be unambiguously related to the service life, although it is reasonable to expect that corrosion and fretting damage will occur more often at longer lives. This is the background to the newer safety by inspection approach, which is based on fatigue crack propagation rather than initiation.

Because of the fact that rotor blades of the present configuration have been shown to be susceptible to defects, one can rely only on safety by inspection, as was rightly seen by the manufacturer of the crashed helicopter described previously. The manufacturer added the blade pressure indicators to the otherwise safe life blades. Unfortunately, a design procedure oriented mainly towards achieving the lightest structure meeting static criteria would be expected to result in dynamic stresses of sufficient magnitude to cause high crack propagation rates in metallic spars, leading to necessarily short inspection intervals, as observed (Refs. 4 and 5). In other words, fitting an inspection device to a component with crack propagation characteristics that follow from design based on fatigue crack initiation does not necessarily lead to a truly damage-tolerant component. It is necessary in addition that the magnitude of the working stresses is such that the fatigue crack propagation rates are sufficiently low to ensure safe and practical inspection intervals.

Recognition of the above-mentioned facts has led to a tendency to design rotor blades in composite materials (glass or carbon fibre in epoxy matrices), which have much improved damage tolerance characteristics at the high design stress levels necessary for competitive application with respect to metallic blades (Ref.4).

5.2 Estimation of safe fatigue initiation and crack growth lives

If a type of rotor blade is insensitive to defects (e.g. composite blades, Ref.4) then it is still useful to estimate safe fatigue lives. The safe life is effectively the life to initiation of a propagating crack, since propagation life is comparatively short. Also, it is obviously necessary, as exemplified in section 2.3, to determine the crack propagation lives of every type of rotor blade as accurately as possible.

At present, nearly all helicopter manufacturers conduct fatigue tests with constant amplitude loading only and calculate service lives using the Palmgren-Miner cumulative damage hypothesis. The reasons most often advanced for this procedure are:

- experience has shown it to be satisfactory
- the cumulative damage hypothesis is reasonably valid
- flight simulation tests (the alternative if they were to comprise both crack initiation and propagation) would take too long and are too specific.

The first reason is difficult to refute in a general way, although it is surely significant that many fixed wing aircraft companies carry out tests with realistic load sequences. However, it is certainly possible to dispute the other two reasons. An extreme example of the error that can be made by applying the Palmgren-Miner hypothesis is the calculated crack detection time of the order of several hundreds of flights for the presumably more critical inboard sections of rotor blade spars, while the outboard spar service failure indicated a crack detection life of 2.1 - 7.6 hours (section 2.4). More generally, the cumulative damage hypothesis may be expected to be in valid because:

- it does not account for the fact that under realistic loading conditions there is a significant damage contribution by the large number of stress cycles below the constant amplitude "endurance limit"
- the interaction of stresses of different amplitudes is not accounted for. This interaction has been shown to be important for variable amplitude load sequences, e.g. Ref.6, and it is generally not possible to predict whether the hypothesis will give conservative or unconservative results (Ref.7).

With respect to the suggestion that flight simulation tests would be too time consuming, it is notable that three flight simulation crack propagation tests on outboard spar sections were eventually completed by the manufacturer of the crashed helicopter: total testing time was less than 20 hours and the service problem was confirmed. But earlier constant amplitude tests on the presumably more critical inboard spar sections lasted hundreds of hours, and the results gave no apparent cause for concern when analysed with the Palmgren-Miner hypothesis. Besides improved prediction of service lives, flight simulation tests have the potential advantage of requiring fewer specimens than constant amplitude tests, and this may even result in a net saving in testing time.

The specificity of flight simulation tests, i.e. the assumption that certain types of mission will be flown in service, is hardly a disadvantage compared to the highly artificial nature of constant amplitude loading. It may well turn out that individual operator usage differs significantly from the design assumptions. This is an important problem, which can be dealt with in three ways:

- a new cumulative damage calculation using constant amplitude data is made, with, however, no more guarantee of accuracy than the original calculation
- a cumulative damage calculation for the flight simulation test history is made and a coefficient is introduced so that the calculation agrees with the test results (this is the so-called Relative Miner Rule): then using this correction coefficient the life under differing service condition is calculated
- a new series of flight simulation tests can be done.

The last two methods, both of which involve flight simulation tests, are evidently preferable to the first one.

6 SUMMARY

In this paper we have shown that catastrophic failure of a rotor blade spar resulted in altering the fleet operation and maintenance procedures in the Netherlands. From a list of failures it appears that the spars are susceptible to various types of defects, such that failure can occur well within the factored safe life obtained from fatigue tests of defect-free spars. It was also shown that the predicted crack propagation life available for crack detection, obtained by application of the Palmgren-Miner cumulative damage hypothesis to constant amplitude crack propagation life data, was highly unconservative. The main reason for this is that prediction techniques for damage accumulation are insufficiently developed. Finally, several design and certification aspects, including the estimation of safe fatigue initiation and crack growth lives, have been discussed.

We conclude that there is, in general, a strong case for the introduction of flight simulation fatigue testing of helicopter components, including both crack initiation and growth. Crack propagation data should be required because crack initiation can never be fully excluded: some kind of damage will occur in practice. Then the safety can be judged only from crack growth observed for a realistic test.

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