2 Analysis

2.1 Catastrophic failure

The accident occurred when the outer portion of the yellow colour-coded main rotor blade separated as the helicopter was approaching to land on the Santa Fe Monarch. A fatigue crack, originating from the combination of a manufacturing anomaly and the effects of a lightning strike, had progressed, very slowly at first, in a chordwise direction, both forwards and rearwards from an initiation point on the upper surface of the blade's titanium spar.

A few hours before the catastrophic failure, the crack propagation rate accelerated markedly and the crack grew to about half the blade spar's cross-section before the various forces acting on the spar were sufficient to overcome its residual strength and the outer two thirds of the blade separated. The main rotor blades rotate at about 304 RPM and the centrifugal force imbalance created by the missing blade section was about 27,000 lbf. This imbalance exceeded the design requirements for the main rotor gearbox attachment fittings by a factor of more than four. Consequently, the gearbox, together with the three intact blades and the stub of the broken blade, tore away from the helicopter's fuselage. This probably happened a fraction of a second after the blade failed. Even if the gearbox had been retained, the helicopter would have been uncontrollable and the environment for its occupants would probably have been unsurvivable. Therefore, separation of the main rotor gearbox had no appreciable effect on the outcome.

The gearbox separated when the helicopter was travelling at about 94 kt airspeed and at a height of 323 feet. After the main rotor gearbox separated, drive to the tail rotor would have been lost and so there would have been no tendency for the helicopter's fuselage to rotate due to tail rotor yaw force. With an initial airspeed of 94 kt and with the fin in place and largely undamaged, the fuselage probably fell nose-first in a ballistic curve, accelerating until it struck the sea surface. The combination of forward and vertical speeds was responsible for the severe disintegration of the fuselage and the immediate fatal injuries suffered by everyone on board.

The gearbox, with three intact rotor blades plus the inboard stub of the yellow coded blade fell to the sea, probably in a manner somewhat resembling a sycamore seed as reported by one of the few eye witnesses. As it fell, the main rotor assembly, which was also no longer supporting the weight of the fuselage, would have lost most of its forward speed and much of its rotational speed. These factors explain the relatively superficial damage to the assembly compared to the remainder of the helicopter and why it was found some distance from the main debris field.

Search and Rescue assets at sea and ashore were deployed without delay but there was nothing that could be done to save any of the 11 persons on board.

2.2 Scope of analysis

This was the first known in-flight fatigue failure of a titanium sparred main rotor blade manufactured by Sikorsky and so, from the outset, it seemed likely that exceptional circumstances were involved in this failure. The origin of the crack in the yellow colour-coded blade was beneath the tang of one section of the two-piece titanium erosion cover. This tang had been inadvertently distorted during blade manufacture in 1981, resulting in a small malformation of the scarf joint between the two pieces of the erosion cover. Although the malformation went unnoticed for 18 years, it seems to have had no material effect on the integrity of the blade until it was exploited by an electrical pulse that can be attributed to a low-magnitude lightning strike (less than 10,000 amps) in 1999. The electrical discharge passed between the two sections of erosion strip via the tang, which at the fold line, was either in contact with the spar or very nearly so, and the spar itself. This momentarily created sufficient heat to change the material properties of the titanium spar in a small elliptical region probably less than 2 mm wide. The reasons why the manufacturing anomaly was not noticed and rectified during blade manufacture, and had apparently not been noticed during subsequent inspections and blade repairs, deserve an explanation.

An electrically conductive path (bonding[1]) giving some protection against lighting strikes had been built into the blades. However, there was no deliberate electrical bonding path between the upper and lower conductive aluminium wire meshes and the titanium erosion cover although the gap between these components appeared to be very small (significantly less than 1 mm). Some burning of this aluminium wire mesh was visible but not in the area of the scarf joint. In this region an electrical discharge passed between the malformed tang and the spar. Had the spar damage been detected (or as will be explained later, detectable) when, following the lightning strike, the blade was returned to its manufacturer for assessment, this accident might have been avoided. However, it is appropriate to review the assessment in the context of that time, when the potential for microstructural damage through this mechanism was not appreciated, perhaps principally because it had not previously occurred.

[1] Joining together all major metal parts of an aircraft, especially an aircraft not of all–metal construction, to ensure low-resistance electrical continuity throughout. Bonding is necessary for Earth–return systems and to dissipate lightning strikes and other electrical charges safely with no tendency to arcing

After refurbishment the blade was returned to service in 2001 having accumulated 8,261 flying hours but it did not fail until some 1,400 flying hours later. Current estimates of the crack propagation rate suggest that a through-crack[1] did not form until between 21.3 and 29.8 flying hours before complete failure. This analysis will examine the symptoms of imminent blade failure during this latter period and what, if anything, could have been done to detect an imminent failure in time to prevent the accident.

There were three basic means by which the crack might have been detected; by visual inspection; by non-destructive testing; and by trend analysis using data recorded on board the helicopter. The potential for each of these basic means will be examined. Finally, the analysis concludes with a review of the safety actions taken, assessing whether these are sufficient to prevent a similar accident, and whether the current blade inspection, testing and crack detection processes might be adapted to better detect fatigue cracks in spars.

2.3 Interim Safety Action

Representatives from the helicopter manufacturer and the Federal Aviation Administration of the USA were incorporated into the investigation team as advisers to the Accredited Representative from the USA. As soon as the investigation team was certain that the fatigue crack emanated from the combination of a folded tang and thermal damage by a lightning strike, the aircraft manufacturer issued an Alert Service Bulletin (ASB). This ASB, issued nine days after the accident, required the identification and removal from service of all S-76 main rotor blades that had suffered a lightning strike and any which in future might be similarly affected. The next day, 26 July 2002, the AAIB sent an interim safety recommendation identified as 2002-25 to the FAA recommending that:

> *The Federal Aviation Administration mandates appropriate action to ensure the continued airworthiness of Sikorsky S-76 main rotor blades which have either:*
>
> *a. A two-piece leading edge titanium sheath (erosion strip).*
> *or*
> *b. Have suffered a lightning strike.*
>
> (Safety Recommendation 2002-025)

That same day the FAA acted by giving mandatory status to the manufacturer's ASB through an FAA Airworthiness Directive. Subsequently, on 9 August 2002 the manufacturer issued another ASB requiring a one-time

[1] The point at which the embryonic crack in the spar's outer surface reached the spar's inner surface.

inspection of the scarf joint on all S-76 main rotor blades fitted with a two-piece leading edge erosion cover. This inspection required removal of any patches, paint, primer and adhesive covering the scarf joint and inspection for a folded tang. No more folded tangs were found.

2.4 The manufacturing anomaly

2.4.1 Blade manufacture

The techniques and materials used for manufacturing the S-76 main rotor blades were very similar to those used for manufacturing blades for the larger H-60 and H-53 helicopters, the main differences being those of scale. The blades all have titanium spars and all are made at the same facility using much the same processes carried out by the same skilled workforce.

The plasma arc butt welding process used to join the rolled titanium plate is partially automated and carefully controlled making stray thermal damage unlikely. The weld is then subjected to rigorous inspection, both internally and externally, throughout its length before the tube is flattened leaving the weld at the centre of curvature facing the trailing edge of the blade. Consequently, any stray thermal damage during the welding process should be detected but if not, it would be near the trailing edge side of the spar and well away from the area beneath the scarf joint. Plasma arc butt welding was the only high temperature process during blade manufacture so the heat damage near the spar's leading edge could not have been inflicted during the blade's manufacture.

The blade components were assembled by hand and most were joined using structural adhesive bonding. The tooling used to hold the blade during assembly generally supported it inverted at a height above the ground that was a few inches above the average man's waist height. The convention of keeping the blade inverted arose because the shape of the upper surface of the blade is aerodynamically more critical than the underside. By assembling the blade inverted, the upper surface of the blade lay on the rigid metal former and the slightly flexible caul was placed on top of the assembly which by then was the underside of the blade. The assembled blade was transferred from the compaction fixture to the bonding fixture by an overhead crane that needed to raise the blade by only a few inches to move it. Consequently, the upper surface of the blade was always on the underside throughout the assembly and curing processes. In that orientation, the scarf joint anomaly was unlikely to be observed since it was facing the ground and well below eye level.

The uncured epoxy structural adhesive used to bond the erosion strip to the blade was very 'sticky' to touch. It seems likely that the unsupported rear edge of the thin metal tang on the outboard section of the erosion cover was gripped by the adhesive before the assembly was pressed fully home in the compaction fixture. As the assembly was pressed home, the gripped tang folded backwards into alignment with the direction of compaction. Being underneath the blade and a little above waist height, it was not obvious to the technicians that this had occurred. The remaining components of the blade were added and the whole assembly was then transferred to the bonding fixture before curing in an autoclave. Sufficient adhesive was used to be sure of high quality bonding and the filling of any minor voids before being exuded from the interior. This excess adhesive was sufficient to fill a small depression left in the surface of the blade by the folded tang and its opacity was sufficient to hide the anomaly.

Although the quality control process during blade assembly was extensive, it did not include a specific check of the scarf joint prior to autoclaving and hence the anomaly was not noticed during the manufacturing process. There was however, one final chance to detect the anomaly before the blade was released to service. Each new blade is examined across its entire width and length by fluoroscopy (described in paragraph 1.18.3.3.2). This examination was capable of revealing the folded tang had the operative been searching for such a flaw but the search objectives were the detection of foreign objects and correct assembly of the internal metallic parts, particularly at the root and tip ends of the blade.

All S-76 main rotor blades manufactured since 1989 have a one-piece erosion cover that is far less susceptible to assembly distortion. Moreover, the manufacturer now uses improved tooling and a greater degree of automation to attach the erosion cover assembly to the leading edge.

2.4.2 Other folded tangs

Some months after the yellow blade was constructed, a similar deformation of one tang was discovered during manufacture of an S-76 blade (see paragraph 1.18.9). Although the defect was discovered, the mechanism by which it arose was unknown. A Material Review Board assessed the defect as 'not significant' and this blade was re-worked, principally by cutting out the folded tang and filling the void created. Had the same anomaly been discovered during manufacture of the yellow blade, it seems probable that a similar decision would have been taken.

After this accident, a check of all S-76 main rotor blades bearing a two-piece erosion strip was initiated by the manufacturer. The check required a one-time inspection of the scarf joint for a deformed tang. Of the 1,600 to 1,675 blades affected only 310 reports were received by the manufacturer, each of which reported no anomaly. It is likely that many more blades were examined but, having found no anomalies, the examiners declined to submit reports to the manufacturer. Since no other instances of folded tangs were reported it appears likely that the yellow colour-coded blade fitted to G-BJVX was the only one blade to enter service with a folded tang.

2.4.3 Airworthiness of the yellow blade

There was no evidence on the recovered section of the blade of mechanical interference between the deformed erosion cover tang and the spar. Nevertheless, even if the tang had been rubbing the spar's surface, given the calculations conducted by the manufacturer, there is no reason to suppose that the anomaly would have had any impact on the airworthiness or service life of the blade, unless it was exploited by another damage mechanism. This investigation determined that it was exploited by a lightning strike.

2.5 Thermal damage to the spar

2.5.1 Lightning strike examinations

Although two of the blades damaged by the lightning strike suffered by G-BHBF were photographed before removal, no photographs of the blade that features in this investigation could be found. The only records of the damage suffered were the written records retained by the UK repair organization, the USA Repair Station and the helicopter manufacturer.

It seems clear from all three sets of records that the blade exhibited conspicuous external signs of lightning damage. The signs included burning of the aluminium wire mesh embedded in the upper and lower surface skins at blade station 44 which is inboard of the scarf joint and in a similar location to the strikes on the blades photographed after the strike to G-BHBF. There were also arc marks and abnormal features described as 'blow out' marks on the leading edge near the erosion strip. Nevertheless, although there were obvious lightning strike marks on the blade, there were none in the region of the scarf joint. The areas with conspicuous damage were repairable in accordance with procedures in place at the time.

The two repair agencies that examined the blade before the manufacturer declined to repair it but found no signs of damage to the blade spar, although one organisation recorded visible marks inside the hollow spar. After the accident only two of those marks could be assessed since the remainder were inside the outboard portion of the blade that was not recovered. These marks were difficult to identify using a boroscope but after blade sectioning and closer inspection, they were thought to be marks left by solidified contaminants such as oil, grease, paint and possibly dirt. These contaminants were almost certainly harmless and were probably introduced when the blade was in service, making their way along the inside of the spar under the influence of centrifugal forces. Certainly, there were no signs of lightning induced or any other mechanical damage inside the recovered section of spar.

The authority for the final decision as to whether the blade was repairable was vested in the manufacturer's engineering staff. There were no prescribed limits for the size or extent of any lightning damage. As the Engineering Instruction (Appendix C) makes clear, the assessment ('disposition') had to be made by experienced engineers using their engineering judgement. They had examined other blades struck by lightning and had amassed relevant experience. Some of these blades were scrapped but some were repaired and returned to service. Damage due to wear and tear on the blade (eg water ingress) was detected during the blade receipt fluoroscopy inspection (which was not part of the EI lightning strike inspection but a more general examination of the internal condition of the blade). However, the presence of the folded tang was not detected. The engineering assessment was that the yellow blade was repairable provided that it passed all the detailed inspections stipulated in the Engineering Instruction. Other minor defects that had arisen during normal in-service operation were also identified as requiring repair.

2.5.2 Detection of thermal damage

Inspection of the blade by its manufacturer was unlikely to have revealed the folded tang unless it was noticed during either the receipt or release fluoroscope inspections. These radiographic inspections were not intended to detect such flaws; their purpose was to detect foreign objects, internal metallic damage or incorrect assembly of metallic parts, and to inspect the interior for water ingress. These aims were achieved; water ingress was identified during the incoming inspection but not metallic damage or incorrect assembly. Moreover, throughout the inspection and refurbishment processes, the damage to the spar was thermal and not mechanical. The size, shape and thickness of the spar metal in the area beneath the scarf joint would have been unaltered by the thermal damage. There was no crack in the spar and so there was no prospect of detecting the small area of thermal damage beneath the folded tang using fluoroscopy.

As the QinetiQ report at Appendix L makes clear, the fatigue crack initiated from a localised change in the microstructure of the titanium metal. After cleaning the fracture area, the change was visible on the spar's surface as a small region of discolouration but to identify the extent of the damage, QinetiQ had to use a scanning electron microscope. The region of changed microstructure within the recovered half of the fracture face was a semi-elliptical shape 1.23 mm wide and 0.23 mm deep.

There is no known non-destructive method for detecting such small and localised changes in the microstructure of a large metallic component that is encased within layers of fibreglass, sheet metal or metallic mesh, and epoxy adhesive. In areas where there was suspicion that the spar might have been damaged, it would have been possible to remove the damaged skins, expose the spar and chemically clean the epoxy adhesive and primer paint layer from the spar surface. However, the etching agent used to remove the adhesive and primer residues would also have removed the tinted layer on the titanium surface. Since there was no physical deformation of the metal, the small area of microstructural damage would then have been invisible.

The spar of the blade was the primary load bearing structure and therefore critical to its continued airworthiness. The assessors made a judgement that being encased in secondary structure, the spar was unlikely to be damaged in areas where that secondary structure was undamaged. There was no visible damage in the scarf joint area (blade stations 75.25 to 82.75). The nearest visibly damaged area was a crack in the leading edge erosion cover at blade station 112 which was 29.25 inches from the scarf joint. That defect was not attributable to lightning damage. The nearest burn damage to the blade's skin was 36 inches from the scarf joint at station 118.75. Consequently, there was no realistic prospect of detecting the thermal damage in the spar before the blade was repaired.

2.5.3 Blade refurbishment

The actions taken to refurbish the blade were extensive and detailed but the focus of the repairs was remote from the scarf joint area. The manufacturer had earlier used a splicing process to repair cracks in the leading edge erosion covers but this practice was no longer invariably used when the blade was repaired. Unless an area of disbonding wider than four inches from the crack was found, approved 'repairs' to the erosion cover were achieved by covering them with plastic tape in accordance with a procedure listed in the manufacturer's Composite Materials Manual.

Had cracked sections of the two-piece leading edge erosion strip been removed and replaced, the folded tang might have been discovered since the epoxy adhesive filled void might have been exposed. The void would have been exposed if the entire erosion cover had been replaced by a new one-piece unit. The manufacturer determined that because any disbonding was within limits, the cracks could be covered with tape in accordance with the Composite

Materials Manual. The blade manufacturer did not carry out these patch 'repairs'. They were deferred for corrective action by the customer (the USA Repair Station) as part of the blade finishing process that was carried out before the repaired blade was returned to the operator.

All the cracks in the erosion cover were outboard of the scarf joint and the nearest was 29 inches away. Consequently, it seems unlikely that any splice repair which physically replaced sections of cracked metal would have involved disturbing the scarf joint, so the change in repair method probably had no effect on the prospects of discovering the folded tang. Nevertheless, had the scarf joint been disturbed during refurbishment, the epoxy filled void and possibly the discoloration beneath the folded tang might have been noticed by a technician and reported to engineering for 'disposition'. This was the only method by which the thermal damage might have been detected and assessed during blade refurbishment.

2.6 Fatigue crack formation

2.6.1 Crack initiation

The failed blade had been subjected to flight loads for 1,403.6 hours since refurbishment. At some indeterminate stage during that period, the crack formed. Cyclic loading of the blade was responsible for the crack. This cyclic loading arises mainly from changes in blade lift, blade flapping, blade bending and blade lead/lag motions. Centrifugal on-off cycling is also a significant fatigue inducing load.

Many materials under the influence of cyclic loading are susceptible to fatigue. In this case, the spar surfaces were shot-peened to produce compressive residual stresses in the spar surface and so reduce the likelihood of fatigue initiation within the design life of the blade. The thermal damage caused by the lightning strike removed the beneficial effects of the shot peening and changed the microstructure of the metal to a structure more susceptible to fatigue initiation. After the thermal damage had occurred, cyclic loading in the blade initiated a fatigue crack at this point within the design life of the blade.

The crack took several hundred hours to materialise after the lightning strike. This is a typical feature of fatigue cracks. The nucleation of a fatigue crack absorbs the majority of the fatigue life of a component in that initiation can take significantly longer than propagation. Initially the crack direction was from the outside of the spar towards the inside until a through-crack formed. This initial through-crack growth was likely to be different from the crack growth in the two opposite propagating crack fronts.

2.6.2 Spar crack striation counts

The AAIB employed the QinetiQ materials laboratory to analyse the fracture face. QinetiQ's assessment of the crack comprised three sectors:

1. From the origin to a length of about 0.6 mm in each direction (a total crack length of 1.2 mm). This sector was associated with through-crack growth and the elapsed time for this sector was indeterminate.

2. From about 0.6 mm to about 14 mm from the origin in both directions (a total crack length of 28.3 mm). The elapsed time for this sector was calculated by extrapolation of the striation count in the third sector. This technique produced an estimate of around 1.7 million striations in these sectors (1,689,429 for the trailing edge and 1,691,669 for the leading edge).

3. From about 14 mm either side of the origin to a total crack length of 143 mm. The striation densities in this sector were measured and represent a good estimate of the crack growth. Within this sector there were 158,824 striations towards the trailing edge and 137,402 striations towards the leading edge and around the nose curvature of the spar.

This early work was later reviewed in the light of Sikorsky's striation count. Their assessment estimated the number of striations inside 14 mm from the origin by measuring striation density and not by graphic extrapolation. The striation density measurements were combined with QinetiQ's power curve mathematical method to estimate the number of striations in this region. The results were a very significant reduction from about 1.7 million to about 244,000 on either side of the fatigue origin.

Interpolation of the graph at Figure 4 of Annex P shows that in the regions from about 14 mm either side of the origin to the ends of the fatigue region, the Sikorsky striation density measurements also produced increases in the striation counts in the order of 25%. However, there were surface features that were counted as valid fatigue striations by Sikorsky's assessments of crack face replicas but not by QinetiQ who assessed the original crack face. Given that these differences cannot be resolved without a great deal of further work, the investigation team considered it reasonable to combine Sikorsky's striation density measurements inside 14 mm from the origin, where the TEM method was superior, with QinetiQ's striation densities beyond 14 mm, where the SEM method was more than adequate and the crack itself was assessed. The results of this combined data assessment are shown in column 3 of Table 5.

Table 5 – Total Striation Count Estimates

	QINETIQ DATA	SIKORSKY DATA	COMBINED DATA
Leading Edge of Crack Origin to end	1,829,071	542,704	**388,025**
Trailing Edge of Crack Origin to end	1,848,253	537,008	**444,712**

2.6.3 Fracture mechanics assumptions

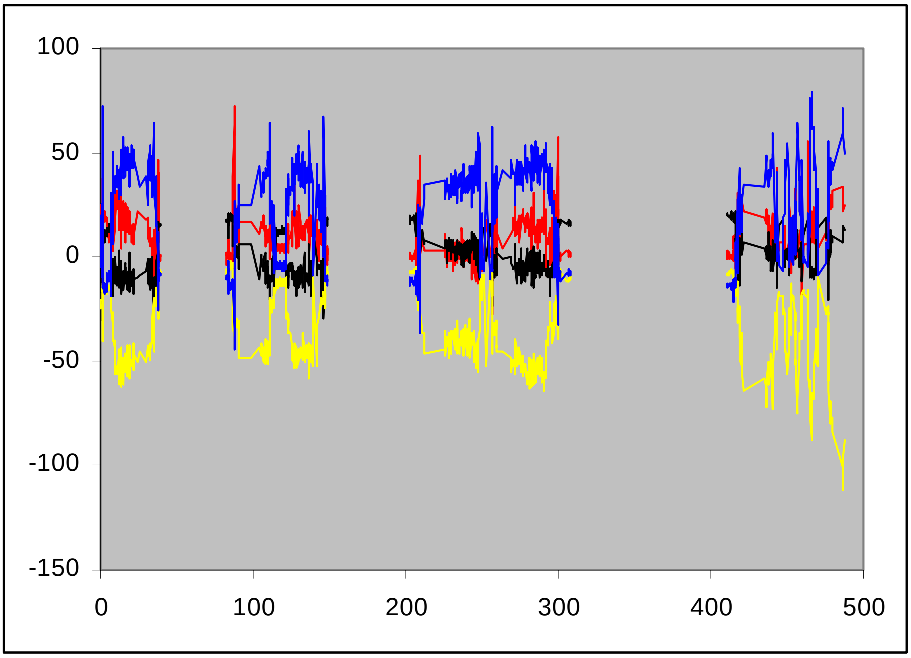
QinetiQ's first striation report at Annex M states*:*

> *'The use of striation counting to estimate crack growth must be used as a guide only. Many assumptions are made about the loading cycle. In this case it has been assumed that each blade revolution produces one striation and that each blade revolution occurs at a speed of 304 RPM. It is possible that some striations are produced by cyclic loads not associated with the blade rotation such as vibration and wind gusts, although in this case that problem was addressed by measuring the distance between the prominent striations and ignoring the 2 to 3 smaller striations that were observed in-between. However, this again is an assumption.'*

It is not possible to calculate the flight hours during which the crack progressed without making some assumptions and QinetiQ were fully aware of the fragility of their initial assumptions. The helicopter manufacturer also cast doubt upon the direct relationship between striations and blade rotations. The company thought the ground-air-ground cycle, during which a blade transitions from producing little or no lift to lift for flight, followed by a reduction to near zero lift after landing, would also have a major effect on crack progression. Additionally, it is possible that the blade accelerate-decelerate cycle, which coincided with engine starts and stops and normally occurred just once per multi-sector flight, might also have had an appreciable effect on fatigue progression. This cycle induces significant changes in centrifugal and lead/lag forces within the blades. Furthermore, there was a possibility of crack progression whilst the rotors were running on the ground or on a helideck; these periods represented 'rotors-running' operations for which flight time was not logged. Unfortunately the study into these potential contributory factors was not completed.

2.6.4 Spar crack progression during final flight

Figure 4 of Annex K (reproduced below) shows little difference in the track of the yellow colour-coded blade during the first three flights on the graph (the second, third and fourth flights of the day) but a change is unmistakeable during the fifth and final multi-sector flight. These changes must be related to spar crack size and so the crack was growing rapidly during this flight.



Annex K Figure 4 - Track Data (mm) from FDR vs Pseudo Time (min)

The rapidity of the crack growth during the final five sectors of the last flight is also illustrated by Figure 7 on page 15 of Annex K which shows the yellow blade track changing during each sector departure. This graph also suggests that something significant happened to the blade between the third and fourth sectors; this could, however, have simply been the change in payload from zero to eight and then nine passengers and their baggage for the last two sectors (see Table 2).

All these discernable changes were taking place during a period of just 72 minutes (of which 55 minutes were flight time) which encompassed 5 takeoffs, 4 landings and 1 blade acceleration. Consequently, it seems likely that the crack growth rate during the last few flight hours was driven predominantly if not exclusively by blade rotation. If the ground-air-ground and blade accelerate-decelerate cycles did have an appreciable effect on crack propagation, then it seems likely that their effects were more pronounced during the early stages of crack formation and progression. Therefore, the analysis proceeds on the basis that during the final stages of crack growth, one striation occurred with each blade rotation.

2.6.5 Total time for spar crack propagation

The total time for spar crack propagation can only be estimated by dividing the total number of striations by the in-flight blade rotation speed of 304 RPM (18,240 cycles per hour). The results are shown in Table 6 below. It must be emphasised that of these figures, the combined data represents the minimum period.

Table 6 – Spar crack propagation time estimates

	QINETIQ DATA	SIKORSKY DATA	COMBINED DATA
Leading Edge of Crack Origin to end	100.3 hours	29.4 hours	24.4.hours
Trailing Edge of Crack Origin to end	101.3 hours	29.8 hours	21.3 hours

The only other clue to the total period is the significance of the 63 'macro bands' in one direction and 61 bands in the other direction observed by QinetiQ which are indicative of a change in circumstances during crack propagation. The record of flight times and sectors flown at Annex F shows that the last 63 sectors were flown over 19.6 flight hours and that the last 63 engine starts took place over 70.4 flight hours. If the macro bands represent engine starts and stops, then the total crack propagation period is likely to be nearer 70 hours than 20 hours. If they represent ground-air or air-ground changes in vertical blade loading, then they correspond to events during the last 19.6 flying hours and so the total period is likely to be nearer 20 hours than 70 hours. Neither correlation seems a good match for any of the data sets in Table 6 although the closest match is the combined data and the ground-air-ground cycles. Consequently, the period of spar crack propagation remains indeterminate but allowing for the different striation counts, it seems very likely that significant spar crack growth began within the final 100 flight hours.

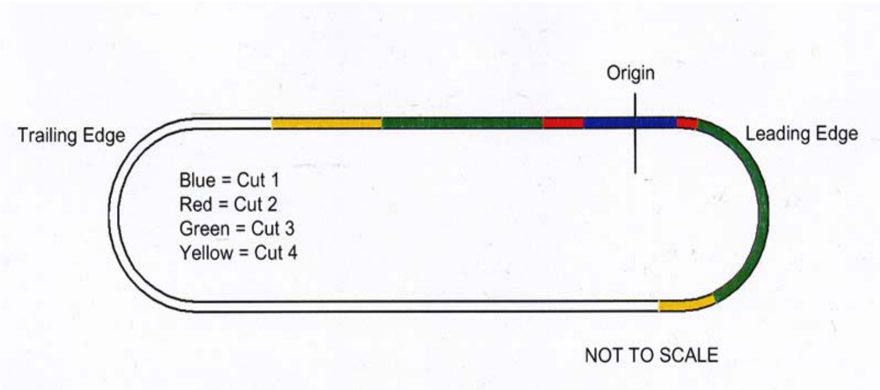
2.6.6 Crack growth rate shortly before the accident

Defining a specific phase of spar crack progression is somewhat subjective but before the external crack formed, there was no visible evidence suggesting internal damage to the spar. Therefore, the start point of the most important phase is taken to be when the sympathetic crack first formed in the blade's exterior close to the crack origin.

During the manufacturer's coupon tests (see paragraph 1.16.1) a sympathetic crack in the fibreglass skin and paint layers first appeared, when the crack in the adjoining titanium spar material was 2 inches (nominal 50 mm) long. Therefore, a crack length of 50 mm appears to be an appropriate start point for analysing the subsequent, rapid growth phase. It also permits a meaningful comparison of QinetiQ's striation count with Sikorsky's count and permits scoping of the likely flight hours during which an exterior surface crack was present.

QinetiQ's graph on page 3 of Annex N suggests that the spar crack length grew from 50 mm to 143 mm (failure) in a little less than 5 flight hours. Interpretation of the combined data, as shown at Figure 4 of Annex P, suggests that it grew the same amount in 6.3 flight hours (21 mm leading edge and 29 mm trailing edge). Consequently, despite the differences in striation counts, both elapsed time assessments were broadly similar for the phase of crack growth from 50 mm to failure.

The major part of the crack was in the upper surface and leading edge of the spar. The trial report at Annex R contains a colour-coded diagram of saw cuts used to simulate the crack size at various stages; this diagram is repeated in Figure 8.



Coloured band(s)	Crack length (mm)	Flight time before failure
Blue	30	9 hrs 30 mins
Blue and red	41	7 hours 35 mins
Blue red and green	102	2 hours
Blue, red, green and yellow	142	0 mins

Figure 8 – Crack growth simulation

The amalgamation of the coloured bands in Figure 8 illustrates the approximate size of the crack at failure. The flight times in the third column have been amended (relative to those in Appendix Q) to conform to the combined data curves in Appendix O.

The data show the spar crack did not reach the underside of the spar until the day of the accident. Had the crack initially formed in the underside of the spar, flight loads would have tended to open the crack and the spar would probably have failed much earlier due to bending loads in the upper surface.

2.6.7 The period between formation of a sympathetic crack and blade failure

A few hours before the accident the spar crack beneath the inboard section of the erosion cover spawned a sympathetic crack in the thin, titanium metal. It was estimated that the flight time required for erosion cover crack growth from approximately 18.5 mm from its origin to final failure was 2.4 hours. Extrapolation of the striation densities using a best-fit curve was plotted to estimate the total number of load cycles for crack growth. The total number of cycles was 133,444, which equates to a time required for crack growth of 7.3 hours. Because of the limitations of striation counting, this should be considered a minimum time and the crack may have formed earlier.

2.6.8 External crack growth

Interpolation of the striation count for the fatigued erosion cover suggests that an external crack was present in the blade's leading edge structure on the morning of 16 July. When applied to the graphs at Appendix O, the result of the crack relationship testing (see paragraph 1.16.1) suggests that an external crack would have formed in the blade's upper skin when the spar crack beneath it extended 50 mm aft of the erosion cover. However, the skin crack might also have formed earlier, particularly if the blade acceleration cycle was tending to open the spar and erosion cover cracks during engine starts.

The spar fatigue origin was 6.7 mm forward of the rear edge of the erosion cover so a skin crack would probably have formed by the time the spar crack front moving towards the blade's trailing edge was 56.7 mm from the origin. Figure 4 of Annex P suggests that this happened about 2.4 flight hours before the accident. Therefore, a crack in the blade's upper skin would most probably have been present on the day of the accident. If a skin crack formed, which seems very likely, the moment that it became visible was when it appeared aft of the polyurethane protective patch.

2.7 Crack detection

2.7.1 Detection opportunities

There was no practical method which the operator could have used to detect the spar crack by visual inspection until a sympathetic crack formed on the blade's exterior. Therefore, exactly how long the spar crack had existed is technically interesting but of limited significance to its discovery. The practicality of timely detection of any exterior crack in the blade's surface is of much greater significance.

Unfortunately, when the sympathetic crack in the erosion cover first appeared, it would have been hidden underneath the black, opaque, polyurethane patch that had been fitted to prevent water ingress into the scarf joint. That patch always hid the crack front that was moving towards the blade's leading edge until the accident. However, the crack front moving towards the trailing edge might have been visible before the blade failed.

2.7.2 Crack detection by routine visual inspection

The last detailed visual inspection of the main rotor blade surfaces was performed during the 50 hour Inspection that was carried out 37 flight hours prior to the accident but there would have been no external symptoms of a crack then. A Daily Inspection was carried out on 11 July on completion of that day's flying and a Pre-flight/turnround Inspection was carried out before each of the five flights on the day of the accident. All six of these inspections required a visual check of the blades' surfaces for obvious signs of damage or disbonding and they were carried out by an authorised engineer (this was the operator's initiative and not an aircraft manufacturer's requirement). The underside of the blades' surfaces would have been inspected whilst standing on the ground whereas their upper surfaces would have been viewed from a position standing on the left side engine decking. Before each flight a pilot may also have inspected the blades but possibly only from a ground viewpoint.

When a main rotor assembly stops rotating after flight, the position of each rotor blade in the horizontal plane will be randomly different. Also, because the main rotor mast is tilted 5° forward from the vertical axis of the helicopter, each rotor blade will adopt a randomly different angle away from the eye of the engineer standing on the left side engine decking. When standing on the ground, the blades can be rotated to improve the view of the underside of each blade but this is not practical when standing on the engine decking. These conditions would make the visual examination and detection of a short, chordwise, crack on a rotor blade's upper surface in the area of the erosion cover's scarf joint, which is approximately 6.61 feet (2.01 metres) from the blade root, extremely difficult, particularly for those blades that came to rest on the starboard side of the machine.

2.7.3 Crack detection through blade droop or discontinuity

The S-76 main rotor blades are flexible and droop when stationary. The experiment report at Annex R shows that in static laboratory conditions a test blade with an undamaged spar drooped 542 mm at the tip. There was no inspection procedure in the Maintenance Manual (nor is there now) that refers to blade droop. The objective of the experiment was to determine whether there would have been appreciable extra droop or a visible discontinuity in the cracked blade's shape when viewed from one end.

The results showed that there was no additional droop when the spar crack was 41 mm long; the crack was about this length when the blade was inspected after flying twice on 10 July. When the blade was inspected during the last Check A on 11 July after the crack had grown significantly, the additional droop was probably about 30 to 40 mm. This much extra droop over the 6,089 mm length of the blade was indiscernible in the laboratory when viewed from either end with the blade stationary. On the airport apron in the open air, the additional droop would have been impossible to see because the blades tend to float up and down in any breeze and the surface wind on 16 July was 6 kt.

2.7.4 The visibility of an external skin crack

The protective patch applied to the area of the scarf joint was not a manufacturer's requirement nor did it conform to any of the transparent tapes specified in the Composite Materials Manual for a repair patch. Nevertheless the fitting of such patches over scarf joints was common practice to prevent the ingress of moisture into the joint.

The patch extended about 35 mm rearwards from the trailing edge of the upper surface of the erosion cover and 42 mm rearwards from the lower trailing edge of the erosion cover. An illustration of the approximate extent of the patch relative to the titanium leading edge erosion cover and its interfaces with the upper and lower fibreglass skins is shown below at Figure 9.

The spar fatigue origin was 6.7 mm forward of the rear edge of the erosion cover so, on the upper surface of the blade, the patch extended at least 41 mm rearwards from the spar crack origin. The graph on page 6 of Annex P shows that at the time of the Daily Inspection, 6.25 flight hours prior to the final failure, the spar crack had progressed approximately 30 mm rearwards from its origin. The crack relationship test showed than any associated skin crack was unlikely to be longer than the spar crack. Consequently, if a skin crack existed at that time, it was invisible.

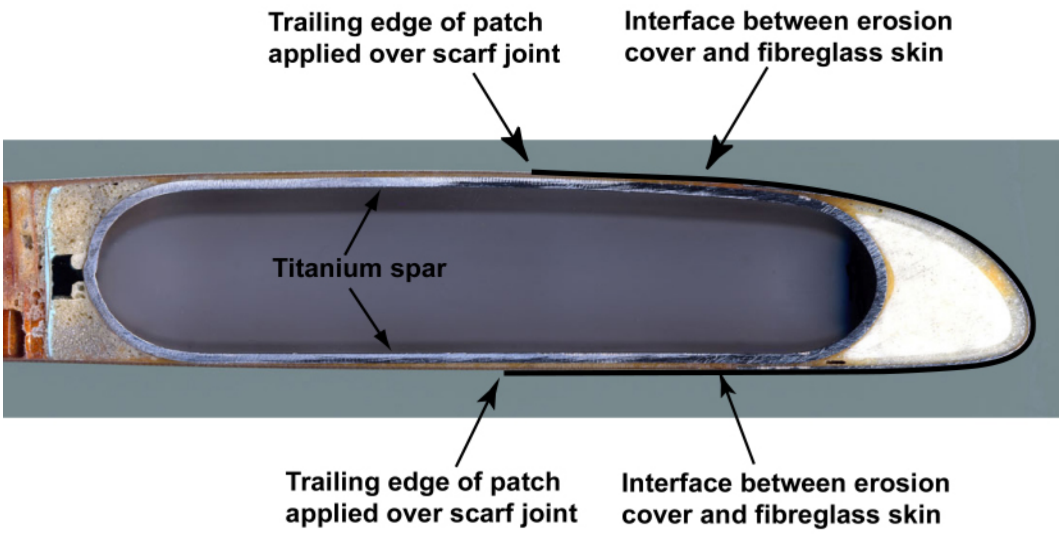


Figure 9 – Approximate extent of polyurethane patch over scarf joint

The graph suggests that the rearward extent of any skin crack did not progress aft of the patch until about 4.3 flight hours before the accident, which encompasses the last three pre-flight checks. Given the inspection difficulties and the few opportunities to see an external crack aft of the patch, if it existed it is not surprising that its existence went unnoticed.

2.7.5 Crack detection by NDT

Since the thermal damage to the blade spar was unlikely to be discovered immediately before or during blade refurbishment, and the operator's opportunities to detect a crack during routine visual inspections were extremely limited, the catastrophic accident could only have been averted if the crack had been detected by NDT before it progressed to failure.

At the time of the accident there had been no known cracking within the body of an S-76 titanium spar and none predicted by the manufacturer. Consequently, there was no requirement for crack detection by NDT. However, in the context of this accident, it is relevant to consider whether there was a realistic opportunity to detect the crack if NDT techniques had been employed during a routine inspection.

The last opportunity when an NDT inspection could have been performed as a routine operation was at the final 50-hour maintenance inspection, which was carried out 37 flight hours prior to the final failure. The graphs at Annex P suggest that the spar crack did not exist when this inspection took place, but if it did, it would have been less than 5 mm long.

The three potential NDT techniques were: eddy current, ultrasonic and radiographic. All three techniques would be very time consuming, they would require removal of the blade from the helicopter, and they would be very much dependent on the expertise of the equipment operator.

It is not easy to determine the size of crack that could be detected without conducting trials using various techniques on a blade with known crack lengths introduced. Both the eddy current and ultrasonic techniques could have detected a 5 mm crack but to enable the techniques to be effective, both would have required removal of the titanium leading edge erosion cover and the wire mesh layer embedded into the surface skins. These operations would be impractical using an operator's expertise and facilities. Every blade would have to be returned to a Repair Station or to the manufacturer for this process. Moreover, to be certain that such inspections were effective, an inspection would have to be carried out at intervals not greater than 20 hours which would be totally impractical and commercially prohibitive.

Since the crack was 'internal', an X-ray technique would be required to detect it in its early stages of propagation without removing the leading edge erosion cover and the embedded wire mesh. In theory, this technique could detect cracks of a few millimetres in length but a general X-ray inspection probably would not reveal a 5 mm crack. The chances of success would be improved if the inspection were focussed on a specific area of the spar using high-resolution equipment. However, the crack may not have grown as far as the through-crack stage at that time and there was no reason to suspect a crack in the scarf joint area. Therefore, X-ray examination was unlikely to have averted the accident.

2.8 Protective patches

Had an opaque patch not covered the scarf joint, the detection of a small crack in the erosion cover at the scarf joint would not necessarily have prompted further investigative action apart from a tap test to check for disbonding on either side of the crack. Assuming that any spanwise disbonding extended less than 4 inches from the crack, and that seems likely, shortly after its discovery the crack would probably have been covered by a patch to prevent water ingress. Consequently, had the scarf joint not been covered by an opaque protective patch before it was returned to service, it is possible that it would have acquired an opaque patch before the accident. Therefore, the fitting of an opaque patch over the scarf joint after the blade was repaired may have had little if any effect on the eventual outcome.

2.8.1 The significance of a crack in an erosion cover

Although the appearance of a crack in an erosion cover is very likely to be benign, it might be the one visible symptom of an underlying crack in the spar or composite structure, especially if the crack spreads into the adjacent composite skin. This feature could have been identified as a symptom of a serious defect prompting blade removal. In this accident, the presence of the patch prevented prompt discovery of the sympathetic crack in the erosion cover. Therefore, in the light of this accident, the addition of patches to the leading edges of main rotor blades should be more rigorously controlled.

Moreover, the crack inspection and monitoring process ought to be more rigorous than a periodic tap test for disbonding on either side of the crack. Once a new crack in the erosion cover is identified, given the knowledge acquired during this investigation, it would seem sensible to monitor the crack's growth frequently, for example before or after every flight, and for a period to be sure that the external crack is only a crack in the erosion cover and not a symptom of a far more serious defect in the spar. It would seem sensible that any spread of a crack into the upper or lower composite skins should prompt immediate withdrawal from service of the affected blade for further investigation.

2.8.2 Patch material

The manufacturer's CMM (Composite Materials Manual) specified the use of transparent patch material in its descriptive list of consumable materials. However, within Repair Procedure No 6, the materials were specified by part number and only one of three part numbers quoted, tape 8663, was identified as transparent.

The USA Repair Station used tape 8663 because it considered it to be more durable due to its greater thickness (0.018 inches versus 0.014 inches for tapes 8761 and 8681). Although both the CMM references to tape 8663 specified the material as transparent, it was available in black and black was used.

Opaque patches were sometimes used because the other approved materials were considered to be less durable. Once a patch is fitted, the recurrent inspection period of an erosion cover crack prompting removal of the patch is never less than 50 hours. If a growing crack is to be visually detected in time to avert an accident, this event demonstrates that a 50 hour interval is too long.

Had there been a requirement to monitor crack growth, an opaque patch would have prevented any further monitoring of the crack size until the patch was removed whereas a transparent patch would have permitted closer monitoring of the crack size during daily and pre-flight inspections. It is accepted that patches placed over the erosion strip are themselves likely to suffer erosion which may render them progressively less transparent. However, a transparent patch, replaced when and if necessary, would better facilitate crack growth assessment during pre-flight inspections until the crack has been confirmed as just a crack in the erosion cover. Thereafter, the crack could safely be covered with a durable polyurethane patch and monitored every 50 or 300 hours as required in the Maintenance Manual. Therefore, it was recommended that:

> The Sikorsky Aircraft Corporation should, within Repair Procedure No 6, clearly specify a durable transparent patch material for covering cracks in the leading edge erosion covers of S-76 main rotor blades. (Safety Recommendation 2004-037).

> The Sikorsky Aircraft Corporation should ensure that new cracks in the leading edge erosion covers of S-76 main rotor blades are frequently monitored for growth by an appropriately qualified person and for a suitable period to ensure that the crack is not symptomatic of a deeper flaw within the blade. (Safety Recommendation 2004-038).

2.9 Pre-flight inspections

It is unlikely that an engineer or a pilot would notice a small defect on the upper surface of the blade, protruding from the rear of a patch several feet distant, particularly since there was no requirement specified in the inspection procedures to look specifically for cracks. The only requirement was to inspect for skin disbonding.

Whilst a responsible engineer would investigate or at least draw his superior's attention to a crack in the skin of a main or tail rotor blade, there was no specific written requirement for that to be done. Although such a defect could rightly be considered an aspect of 'general condition', there was no definition of this term. In the context of the blades' external surfaces, the procedure required the checker to look for 'raised skin indicative of a disbond' and there was no mention of cracking.

Maintenance checks of the main rotor blades are required every 50 flight hours but this accident has shown that a sympathetic crack in the blade surface may not appear until a few hours before catastrophic spar failure (in the order of 5 to 10 flight hours). It would not be practicable to increase the frequency of the maintenance inspection to an interval approaching 10 hours, nor is it considered necessary based on this blade failure which was caused by a combination of two controllable factors: the malformed tang and the lightning strike. Nevertheless, an awareness of surface cracks and their significance should be a requirement for both a pilot's pre-flight check and an engineer's 50-hour inspection.

Whilst it is not appropriate that pilots should become responsible for rotor blade integrity, pilots are required to carry out a pre-flight inspection of the main rotor blades on both the upper and lower surfaces (see Annex B page 1). Despite the difficulties for pilots of inspecting all the blades' surfaces (as described in paragraph 2.7.2) they are, nevertheless, potentially able to notice crack growth from the leading edge erosion strip into the composite skin. Such a surface crack may be indicative of spar cracking and it should prompt immediate investigation by an engineer before undertaking the flight. Therefore, the written instructions for the 'Pilot's Pre-flight Check' (see Appendix B page1) and the engineer's '50-Hour Inspection Checklist' (see Appendix B pages 2 and 3) could usefully remind pilots and engineers respectively to inspect for cracks in the blade skin, particularly around the edges of any patches placed on the erosion strip to cover cracks in the strip. Consequently it was recommended that:

> The Sikorsky Aircraft Corporation should amend the S-76 Pre-flight Check and 50-Hour Inspection procedures to include a search for cracks in the upper and lower skins of main rotor blades. The procedures should prompt investigation of the underlying reason(s) for such cracks before the next flight. (Safety Recommendation 2004-039).

2.10 Crack detection by onboard technology

Since the crack was unlikely to be detected visually or by NDT, the sole remaining possibility was to detect it using on-board technology. The helicopter was fitted with flight data recording systems and it might have been possible to predict the development of a crack by regular data processing and analysis. However, the only approved onboard system for detecting cracks is the Blade Inspection Method (BIM). Each of these possibilities is considered.

2.10.1 Flight data recording

2.10.1.1 Conventional flight data

The conventional flight data parameters recorded by the IHUMS were numerous but there were none that related to the main rotor apart from rotation speed and a discrete parameter (on or off) for the rotor brake; both these parameters were in the normal range. The time history of conventional flight parameters at Annex I showed a normal flight until the blade fractured whereupon the recorder stopped. Consequently, these data would be of no use in predicting a blade failure.

2.10.1.2 Rotor Track and Balance Data limitations

The function of the onboard Rotor Track and Balance (RTB) installation is to capture data for analysis by an IHUMS ground station. Consequently, the RTB data gathered by G-BJVX could not be processed in flight.

As Annex K shows, analysis of rotor track and balance data is a complicated and specialist task. Data from the operator's S-76 aircraft was processed at Norwich and the RTB results were also sent to Aberdeen for assessment by the operator's specialist. To avoid acting upon spurious samples, it was normal practice for engineers to wait for consistent RTB data and consult the specialist before making any adjustments. Sometimes data from previous flights was not processed until the next day. Consequently, to be effective, the data would have to show significant signs of abnormal blade behaviour several flying hours before blade failure. The requirement to provide several hours warning is further complicated by the fact that the rotor track sensor only works in daylight. This is seldom a problem in summer but in winter, when daylight represents about one third of the day, this is a significant limitation because flight operations have to take place in darkness. A third limiting factor is false data. Bright sunlight and reflections can corrupt blade track data although corrupted data can sometimes be identified by visual inspection of the conditions.

2.10.1.3 Rotor tracking vibration

All helicopters vibrate in flight and the RTB system is used to tune the main rotor to minimise vibration from this source. Once the blades have been mass balanced, any increase in vibration would probably be caused by a tendency for one or more blades to rotate with a different tip-plane path to the majority of the blades. When this happens the aerodynamic disturbance caused by the 'out of plane' blade or blades creates vibration. Figure 3 on page 11 of Annex K shows that on the day of the accident, the red and black blades were rotating in much the same tip path. However, the blue blade was flying high and the yellow blade was flying low; these blades were adjacent and so immune to cross-coupling. The blue blade was intact but the yellow blade was close to failing so the tendency to fly high or low, in itself, is not a good indication of blade damage.

2.10.1.4 General vibration data

The RTB data from previous recordings was reviewed and compared with that recorded on other S-76 aircraft. This review concluded that, apart from the exceedence described below, all the monitored vibration parameters on G-BJXV were within established limits and its vibration levels were lower than the operator's S-76 fleet average.

2.10.1.5 Exceedence warnings

An exceedence[1] warning was generated by the IHUMS ground station on the day of the accident. The warning related to data downloaded earlier that day and processed during the afternoon. The exceedence automatically generated a diagnostic printout which is reproduced on page 18 of Annex K.

The warning related to rotor track and balance. It was triggered by high vertical vibration whilst the aircraft was on the ground at 1353 hrs. A vertical value of 0.481 at MPOG (Minimum [Blade] Pitch on Ground) exceeded the threshold of 0.400. As explained in the Annex, '*this would not have caused any alarm because occasional spurious outputs are a characteristic of the system*.'

The underlying reason for the vertical vibration would not have been apparent unless the accompanying warning:

> *'MAIN ROTOR: Tracker data suspect at CRUISE. Maximum velocity difference of 39.00 exceeds threshold of 38.00'*

had prompted the IHUMS ground station operator at Aberdeen to produce and study a detailed diagnostic printout for the cruise portion of the flight.

Due to the signal variability and noise in the IHUMS RTB data it was entirely reasonable under the circumstances to wait for more data to see if there was a repeat warning. However, even if the IHUMS technician had considered the warning as genuine and had produced the detailed printout immediately, at that time there was no reason for anyone to presume that a blade was approaching structural failure. Moreover, even if the technician had recognised its significance and decided to contact the maintenance base at Norwich to advise them of the blade's abnormal behaviour, he was unlikely to have completed this process before 1700 hrs. This was the scheduled time for departure for the final flight so it is unlikely that action would have been taken in time to stop the helicopter taking off.

2.10.1.6 Track, lag and velocity trends

Figure 7 of Annex K plots the track of the yellow blade during the final five accelerations in speed after takeoff from Norwich and each of the four offshore

[1] A single event, recordable on all HUM systems, in which an engine or other device suffers an excursion in operating regime that is beyond limits. In this case the limits were set by the aircraft operator.

installations. The yellow blade flies significantly lower after each acceleration. This graph shows more vividly than any other, the rapidity at which the defect developed during the final sectors. Such was the rapid propagation of the fatigue crack during the final multi-sector flight, that analysis of the RTB data could not have provided more than about 30 minutes warning of anomalous blade behaviour.

It is important to remember that the practical value of the RTB system is to simplify and accelerate main rotor tuning when components are replaced. It was not designed to predict blade failure. Analysis of the data gathered was incapable of predicting blade failure within a practical timescale of several hours flying. Therefore, there seems to be little if any value in trying to develop the system as a method of warning of impending blade failure.

2.10.2 Crack detection using the BIM system

The BIM system involves pressurising the internal cavity of the spar with a gas and monitoring any decrease in the gas pressure. Any crack that propagated through the spar material from the inner to the outer surface would produce a path for the gas to leak causing a pressure drop which in turn would give a warning to the cockpit crew and/or the maintenance personnel. The rate of pressure drop would vary depending on the length and tightness of the crack and the gas path to atmosphere through the secondary structure surrounding the spar.

If a BIM system had been fitted to the main rotor blade, then a warning that a pressure loss had occurred within the spar would have been given to the cockpit crew and/or the maintenance personnel at least 7.3 flight hours, if not more, prior to the final failure. This could have been sufficient time to investigate the pressure loss, discover the crack and prevent the accident.

2.11 Noises and vibrations in flight

In view of the fact that the RTB system detected a vibration exceedance at the start of the penultimate flight, the reports of unusual noises described by passengers on those flights are not easily dismissed. However, by the nature of their employment, the flight crew were more familiar with the S-76 than any of the passengers and if they heard the same noises, they did not appear to be troubled by them. Most of the noises heard in a helicopter cabin with the engines and gearboxes mounted above, emanate from those components. Apart from the damage to the main rotor gearbox caused by overload failure of its attachment fittings, no pre-impact damage was found within the rotating assemblies inside the engines or gearboxes. Also, there were no indications of abnormal engine or gearbox characteristics within the IHUMS data, so the unusual noises heard during rotor start were probably unrelated to the subsequent blade failure. Equally, the grass lifted by the rotor downwash would have been inconsequential and the laboured hover was probably due to the relatively heavy take-off weight on a warm summer evening. Moreover, everyone interviewed who had travelled on the offshore sectors prior to the helicopter's final departure from the Clipper helideck described the aircraft's operation as completely normal.

The passenger sat immediately behind the pilots during the penultimate flight from the Barque PL to the Clipper described normal operations and pilot behaviour. This impression was supported by the audio recording from the CVFDR. In their dialogue both pilots were behaving responsibly and professionally but in a relaxed manner which suggested there was good rapport and co-operation between them and confidence in each other. The co-pilot was handling the controls for the final sector and the only indication of anything unusual was the shared perception of increased vibration seven minutes before the blade failed. They diagnosed the problem by looking at the main rotor tip path plane and noted a 'ragged' track. They then used the IHUMS cockpit interface to trigger an in-flight recording of the RTB which, when downloaded, would advise the engineer on the action required to restore the blade tracking to symmetry. The pilots then continued the flight without giving any audible perception of further concern in their dialogue, nor was there another mention of the vibration. It is likely that the commander intended to report the vibration due to a blade 'out of track' on their return to Norwich.

2.12 Flight crew's response to increased vibration

All helicopters vibrate in flight and the pilots' shared lack of concern was probably due to having flown other helicopters that vibrated with blades out of track. Blade track can change slowly with time and the fact that a blade is 'out of track' is not necessarily an indication of a serviceability problem. Moreover, as page 25 of Annex K shows, G-BJVX had been operated before *'with an overall track split in excess of 200mm – greater than that immediately preceding the blade failure.'* The reporter also concluded that *'G-BJVX had been operated with higher main rotor vibration'*. Consequently, neither pilot had any particular reason to conclude that the vibration was imminently dangerous.

If they had been concerned and wished to land as soon as practicable, since it takes time to prepare a helideck for an unscheduled landing, they could still have been airborne seven minutes later, waiting for the nearest helideck, probably on the Vulcan 2, to be cleared and ready to accept them. In reality, the

Santa Fe Monarch, which they had every right to expect to be ready for them, was probably the earliest suitable place for a planned, precautionary landing.

There was no checklist, advice or previous experience that the pilots could have drawn upon to suspect that a blade might be out of track because of a cracked spar. However, even if they had suspicions and decided to land immediately on the water, it is quite possible that the change in airspeed and blade loading could have precipitated failure before they could accomplish such a landing. The blade failed when they were decelerating for the intended landing on the Monarch's helideck.

2.13 The final events

The flight deck audio recording ended suddenly and without warning about one tenth of one second after the second of two loud 'crack' sounds. These two sounds were separated in time by less than one tenth of one second. Three events could have caused these two sounds; the shock of the rotor blade breaking, the separated section of blade striking the rear fuselage and the gearbox separating from its mountings. It is neither practical nor worthwhile to determine which two of these three events were responsible. All that need be concluded is that, given the industry knowledge at the time, there was no recognisable warning of impending blade failure and nothing the pilots should have done to avert the accident.

2.14 Rotor tracking vibration

With the knowledge gained during this investigation, it is now possible to recognise an increase in vibration coupled with a blade 'out of track' observation as symptoms of impending blade failure. Moreover, although the tracking characteristics of the yellow colour-coded blade were deteriorating relatively quickly during the final flight when compared to the previous flight, the change was still more gradual than sudden. Helicopter vibration can be minimised but not entirely eliminated. It can also have multiple sources and numerous influential factors. Consequently, it is impractical to issue instructions to pilots regarding a threshold level of vibration that is unacceptable and which deserves immediate action; this has to be a subjective assessment. Therefore, no safety recommendation on pilot reaction to vibration is made within this report.

2.15 Modifications to the blade design

The investigation carefully considered whether corrective action, including modification to the S-76 main rotor blade design, should be recommended to prevent a recurrence of crack propagation to in-flight failure.

The spar within the S-76 blade is primary structure with no alternative load path. If the spar fails in flight the result is bound to be catastrophic. Moreover, the external surface of the spar is always submerged in secondary structure which is bonded to it and although hollow, the greater part of the spar's internal surface cannot be seen during periodic inspections because of its length. Consequently, maintainers have to judge the condition of the spar by the condition of the secondary structure which, as this investigation has demonstrated, is an imperfect method.

Although the BIM seems likely to be the best physical method of detecting a crack in time to avert an accident, the disadvantages of retro-fitting the system should be weighed against the probability of a recurrence.

2.15.1 Retrospective fitting of the BIM system

The S-76 main rotor blade was not designed to have a gas-tight spar. Modifications to the root and tip of the blade would be required to make the BIM system work. There would probably be spurious warnings due to gas leaks from imperfect seals and spurious warnings tend to be ignored after a while. Also, the spar itself would have to be drilled to install the BIM detector. Unfortunately, a drilled hole would introduce a stress concentration near the blade root that would enhance the probability of crack development. The end result could be a BIM system that was unreliable but which detected genuine cracks induced by retrospective fitting of a BIM system.

2.15.2 Future blade designs

The consequence of an undetected crack in a main rotor blade can be catastrophic, particularly as in this case when the crack propagates very rapidly in a region where there is no redundant load path. In essence, the design was not damage tolerant as currently specified in Certification Specification 29 applicable to large rotorcraft. However, the development emphasis within the helicopter industry is shifting away from metal sparred blades to composite blades. Such blades are not prone to fatigue cracking. Therefore, it seems unlikely that future helicopter rotor blade designs will have titanium spars but if they do, it would be wise for such blades to be equipped with an automatic onboard crack detection system from the outset. Therefore:

> It was recommended to the European Aviation Safety Agency and to the US Federal Aviation Administration that their Airworthiness Requirements for helicopters should ensure that any future design of main rotor blade that incorporates a hollow metal spar should be designed from the outset to incorporate an automatic onboard crack detection system

covering spar areas which cannot readily be inspected and are not damage tolerant. (Safety Recommendation 2004-040).

2.16 Probability of a recurrence

This accident was caused by a spar fatigue crack that was positively attributed to a combination of two phenomena: a lightning strike that exploited the area of reduced insulation created by a manufacturing anomaly. At the time the blade was repaired, the potential for this damage mechanism was not known by the blade manufacturer but it is now better understood.

It is difficult to calculate the statistical probability of another S-76 blade failing in flight through spar fatigue due to the small statistical sample (one) and combination of factors. Empirically, the probability would seem to be dependent on four factors: the quality of blade design and manufacture, the likelihood of another blade having a similar or comparable anomaly, the likelihood of a lightning strike and the potential for thermal damage from that strike.

Beginning with blade design and quality, S-76 main rotor blades have accumulated some 12.8 million blade flying hours. Taking the wider view, blades of comparable design and construction fitted to the manufacturer's other H-60 and H-53 helicopter models, including some struck by lightning and repaired, have amassed another 28 million flying hours.

Apart from the yellow blade on G-BJVX, within the 40.8 million flying hours accrued by blades constructed with titanium spars by the same manufacturer, none has failed in flight through spar fatigue. Only two blades have exhibited spar cracking in regions which do not have a redundant load path. The first was the crack in the UH-60 blade attributable to battle damage; the other was the subject of this investigation. Consequently, the quality of spar design and manufacture is satisfactory.

Of the 2,800 S-76 rotor blades manufactured, not more than 1,675 of these have two-piece erosion covers and every blade constructed since 1989 has a one-piece 'jointless' erosion cover. The inspection programme instigated after this accident was likely to be assiduously fulfilled by S-76 operators and did not disclose any more blades with folded tangs. Therefore, the probability of another in-service blade having a similar folded tang is also very low.

Lightning strikes affecting rotor blades are infrequent but more common than manufacturing anomalies. After this accident, 26 S-76 blades were removed from service. Of these, 17 had been struck by lighting and the other 9 had incomplete log card histories. Any indication of burns or pits in a blade requires its removal from service but it is conceivable that there are a small number of blades still in service which have been subjected to an electrical discharge which did not produce burns or pits. A lightning discharge so weak that it did not damage the blade's aluminium wire mesh is unlikely to have been sufficiently powerful to inflict thermal damage to the spar. Moreover, the probability of one of these blades also having a folded tang seems extremely low so the risk inherent in these blades remaining in service is negligible.

2.17 Safety actions taken

On balance, the interim safety recommendation made by the AAIB 10 days after the accident coupled with the two ASB safety actions instigated by the manufacturer have ensured that the circumstances of the failure which occurred in this accident have not been repeated. Moreover, the endorsement by the FAA of the decision to remove lightning damaged blades from service, effectively giving the manufacturer's ASB the force of regulation, should ensure that it will never be repeated on a civilian registered S-76 helicopter.

Although the chance of recurrence appears to be extremely low, it is conceivable that thermal damage inflicted by a lighting strike, perhaps associated with a different manufacturing anomaly such as an area of conductive wire mesh touching the spar, could cause similar damage to that seen on the failed blade. Therefore, it would be unwise to return to service any titanium sparred blades affected by lightning fitted to any type of helicopter unless the entire outer surface of the spar can be inspected for thermal damage. Given the method of blade construction and the potential for microstructural damage, this proposition seems impractical. In the case of the S-76, it is immaterial since an Airworthiness Directive was issued requiring all lightning struck blades to be permanently withdrawn from service.

The only alternative, practical and currently available method of monitoring a blade's structural integrity after a lightning strike would be the retrospective fitting of a BIM system. On balance, this proposition is not warranted provided that all S-76 main rotor blades struck by lightning remain permanently withdrawn from service.

Sikorsky is the only known volume producer of hollow titanium sparred main rotor blades for helicopters. The H-60 and H-53 variants are all in military service leaving the S-76 as the only model in civilian service. Consequently, there was no need to recommend the withdrawal of titanium sparred blades from other public transport helicopter models that had suffered a lightning strike.

2.18 Search and Rescue

Two minor issues arose during the course of the investigation concerning the search and rescue phase immediately after the accident. They were the failure of the ADELT to deploy and indicate the surface position of the crash, and the fact that the crews of the Putford Achilles and the Global Santa Fe Monarch had not anticipated the arrival of G-BJVX.

2.18.1 ADELT

The ADELT beacon, a variant of the more generic ELT (Emergency Locater Transmitter) did not perform its intended functions of automatically marking the crash position of the helicopter and transmitting on international distress frequencies. When found, it was on the sea bed still loosely within its carrier. The launcher squib had fired but the spring ejection mechanism was completely overpowered by the speed and force of the impact. Calculations indicated that the speed of water impact was in the order of 140 kt with the fuselage in a 37° dive.

The beacon itself and its ejector mechanism were probably serviceable before water impact but the equipment specification was probably exceeded by the unforeseen brutality of the water impact.

2.18.2 Unexpected arrival of G-BJVX

Nobody on the Santa Fe Monarch was expecting a helicopter movement because the person who requested the movement left the installation before the helicopter arrived and without informing the radio operator. Although the arrival of G-BJVX came as a surprise, as soon as the helicopter came on frequency the helideck was manned and then prepared for a landing. Had there not been an accident, the effect of this lack of awareness would have been little more than a minor nuisance since the helicopter might have had to 'hold off' for a minute or two. The breakdown in the booking procedure on board the Santa Fe Monarch was a minor mistake which was rapidly addressed soon after the accident by the client company.

In the circumstances, the principal effect of the breakdown in procedure was the status of the standby vessel. She was out of position by a mile or so and facing in the wrong direction for the landing. However, when notified of the accident, her crew lost no time in responding and immediately launched two fast rescue craft. They arrived on scene some seven minutes after the accident but it was not survivable. Had anyone survived and been able to operate their lifejacket, the fast rescue craft would have been able to recover them within minutes and transfer them to the Achilles long before the arrival of any other Search and Rescue assets. Consequently, the fact that the Putford Achilles was out of position for the landing had no effect on the survivability of this accident.

2.18.3 Flight-following

The hypothetical question arises that if nobody on board the Santa Fe Monarch had seen the accident, and it had occurred before contact was established with her radio operator, how much time might have elapsed before anyone appreciated that the helicopter was overdue.

On this flight the pilots were always in radio contact with either a land-based agency or an offshore installation, and sometimes in contact with both simultaneously. Had the helicopter crashed out of sight of the Santa Fe Monarch, one of these agencies might have taken follow-up action if the helicopter failed to notify them of a landing or a frequency change. On the other hand, at long range pilots might lose radio contact with an agency before they have attempted to change frequency and so they are forced to change without prior notification. Taking overdue action each time this happened would be inappropriate.

Flight following could be made more efficient if the departure installation positively contacts the destination installation once the aircraft takes off to ensure that the aircraft is expected and at what time. The destination installation could then relay the same notification to the standby vessel and obtain an acknowledgement. If subsequently the aircraft fails to arrive soon after its ETA, appropriate action could be initiated without undue delay.

These communication procedures are not governed directly by the regulatory body for aviation. However, the UK Offshore Operators Association (UKOOA) is the representative organisation for the UK offshore oil and gas industry. Therefore, it was recommended that:

> The UK Offshore Operators Association should amend its guidelines to include a responsibility on offshore installation operators to ensure that, for all flights between manned offshore installations, radio operators of such installations establish positive contact with the destination installation immediately after the departure of a helicopter and convey the relevant flight details such as persons on board and estimated time of arrival. (Safety Recommendation 2004-041).