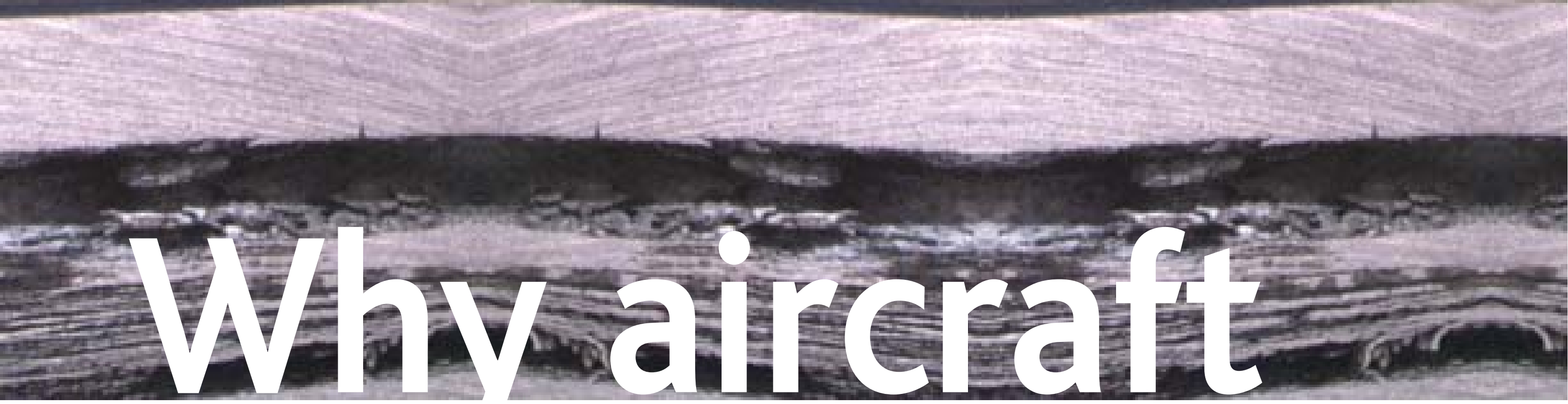

Why aircraft fail

by S. J. Findlay and N. D. Harrison

Failure of an aircraft structural component can have catastrophic consequences, with resultant loss of life and of the aircraft. The investigation of defects and failures in aircraft structures is, thus, of vital importance in preventing further incidents. This review discusses the common failure modes observed in aircraft structures, with examples drawn from case histories. The review will also outline the investigative procedures employed in the examination of failed components.

QinetiQ Ltd,
Cody Technology Park,
Ively Road,
Farnborough,
Hampshire GU14 0LX, UK
Email: sjfindlay@qinetiq.com

QinetiQ has long been associated with the analysis of structural failures and has a long history of involvement with aircraft failure investigations since the days of the Royal Aircraft Establishment. With respect to metallurgical failure investigations and advice, an unbroken sequence of records exists from the Second World War and contains approximately 6000 case histories, of which approximately half relate to structural failure on aircraft. The work undertaken is primarily for the UK Ministry of Defence and the Air Accident Investigations Branch (AAIB) of the UK Government's Department for Transport. Increasingly over the last few years, investigations have also been undertaken for insurance loss adjusters and as expert witnesses in litigation cases.

While assisting in the investigation of accidents to both civil and military aircraft is an important aspect of the work, there are many examples of component failures and defects that were detected before an accident could occur, i.e. during routine maintenance and inspection operations. Many in-service aircraft are now required to operate beyond their original design life, partly as a result of the accelerating costs of replacement and also the ability to upgrade systems in old airframes. As part of the life extension program and aging aircraft audit, in recent years QinetiQ has conducted the teardown inspection of several military airframes. This involves the dismantling of a representative example of older airframes of a particular aircraft type and making a thorough inspection of each component to assess its condition. This

ISSN:1369 7021 © Elsevier Science Ltd 2002

teardown of aircraft has enabled the assessment of components that would not normally be addressed during routine maintenance because of their inaccessibility.

Historically, the majority of the structural failures examined have been in metallic materials, reflecting the predominance of metallic structures in aircraft. However, since the mid-1980s an increasing number of aircraft manufacturers have been making use of fiber-reinforced polymer composites for structural components, which has led to the formation of a specialist team of failure investigators within QinetiQ for this category of material. This review will, however, concentrate on metallic failures.

What causes failure?

In general, failures occur when a component or structure is no longer able to withstand the stresses imposed on it during operation. Commonly, failures are associated with stress concentrations, which can occur for several reasons including:

- Design errors, e.g. the presence of holes, notches, and tight fillet radii;
- The microstructure of the material may contain voids, inclusions etc.;
- Corrosive attack of the material, e.g. pitting, can also generate a local stress concentration.

From our records and case histories data, an assessment can be made of the frequency of failure modes (Table 1). This reveals that the incidence of fatigue failure dominates the distribution in aircraft. This would suggest, therefore, that fatigue is the predominant failure mode in service. The detection and rectification of corrosion damage on in-service aircraft, however, consumes more effort than the repair of fatigue cracking. The high occurrence of fatigue failure observed probably reflects the destructive nature of this failure mode, while corrosive attack is generally slower than fatigue, and usually more easily spotted and rectified during routine maintenance.

Common failure modes

Fatigue is a process whereby cracking occurs under the influence of repeated or cyclic stresses, which are normally substantially below the nominal yield strength of the material. Components that fail by fatigue usually undergo three separate stages of crack growth, which are described as follows:

- Initiation of a fatigue crack. This can be influenced by stress concentrations such as material defects or design.
- Propagation of the fatigue crack. This is progressive cyclic growth of the crack.
- Final sudden failure. Eventually, the propagating crack reaches a critical size at which the remaining material cannot support the applied loads and sudden rupture occurs.

Fatigue failures generally leave characteristic markings on the fracture surface of cracks from which the failure investigator can deduce a great deal of information. The most obvious are the classic 'beach marks', which are commonly observed macroscopically. Beach marks indicate successive positions of the advancing crack front and are usually the first telltale signs that the mode of crack growth is fatigue. Fatigue fractures tend to be relatively smooth near the origin and show slight roughening of the surface as the crack progresses. There tends to be little or no macroscopic ductility associated with fatigue cracking.

Detailed examination of the fracture surface in a scanning electron microscope (SEM) usually shows evidence of fatigue striations (dependant on the material), which represent one

Table 1 *Frequency of failure mechanisms.*

	Percentage of Failures	
	Engineering Components	Aircraft Components
Corrosion	29	16
Fatigue	25	55
Brittle fracture	16	-
Overload	11	14
High temperature corrosion	7	2
SCC/Corrosion fatigue/HE	6	7
Creep	3	-
Wear/abrasion/erosion	3	6

cycle of load and crack propagation. If the magnitude of load cycle remains constant, the striations normally appear closer near the origin, gradually increasing in spacing as the crack front progresses due to the increasing stress at the crack tip. By taking measurements of striation spacing at various distances from the origin to the end of the crack, it is possible to estimate the total number of load cycles to cause failure. If the cause of the loading can be determined, the number of cycles to failure can then be used to estimate the time required for crack growth.

Fatigue cracking is the most common cause of structural failure in aircraft, even though the laboratory fatigue behavior of most metals and alloys is well understood. Materials and their design can be taken into consideration so that the probability of fatigue cracks occurring can be reduced, but it is often the case that the possibility cannot be removed completely. Therefore many aircraft structural components are designed with a safe or inspection-free life, below which fatigue cracking should not be a cause for concern. The fact that fatigue failures still occur, however, indicates the complex nature of this problem. There are many variables that influence fatigue, some of which are the mean stress, peak stress, frequency of loading, temperature, environment, material microstructure, surface finish, and residual stresses. Many of these factors are taken into account when determining the safe life of a component and, therefore, the majority of fatigue failures in aircraft causing catastrophic failure tend to be those that initiate as the result of unforeseen circumstances.

Material surface defects such as forging laps or surface cracking can increase the local stress, producing a concentration at these points that could initiate fatigue much quicker than would be expected. However, many aircraft components are thoroughly inspected by non-destructive techniques after manufacturing and these types of defects are usually detected and rectified. Stress concentrations caused by surface defects such as scratches and wear tend to be more common as these may not be present at build, but can be introduced during service. Another common cause of stress concentration is corrosion, which can lead to fatigue crack initiation.

Table 2 shows a summary of the common fatigue crack initiation sites observed in aircraft[1,2] that have led to accidents.

Ductile or ***overload*** failure occurs when a material has been exposed to an applied load at a relatively slow rate to the breaking point of the material. This results in a ductile fracture of the material, with the fracture surface exhibiting tearing of the metal and plastic deformation.

On rapid application of a load, fast fracture or ***brittle*** failure can occur. Microscopic examination of brittle fractures reveals intergranular or transgranular facets on the fracture surface.

Corrosion is the chemical degradation of metals as a result of a reaction with the environment. It usually results in failure of components when the metal wastes to such an extent that the remaining material cannot support the applied loads or the corrosion renders the component

Table 2 *Summary of fatigue initiation sites observed in aircraft.*

Initiation Site	Number of Accidents	
	Fixed Wing	Rotary Wing
Bolt, stud or screw	108	32
Fastener hole or other hole	72	12
Fillet, radius or sharp notch	57	22
Weld	53	3
Corrosion	43	19
Thread (other than bolt or stud)	32	4
Manufacturing defect or tool mark	27	9
Scratch, nick or dent	26	2
Fretting	13	10
Surface or subsurface flaw	6	3
Improper heat treatment	4	2
Maintenance-induced crack	4	
Work-hardened area	2	
Wear	2	7

susceptible to failure by some other mode (e.g. fatigue). Extensive work has been carried out on the rates and types of corrosion observed in different materials so that selecting a suitable material in terms of corrosion resistance for a known environment is relatively straightforward. In aircraft structures, however, the strength to weight ratio can be a more desirable property than corrosion resistance and in these circumstances the most suitable material cannot always be used. In cases like this, measures must be taken to limit corrosion, which most commonly involve the use of a coating, such as a paint system, to act as a barrier to the environment. There are various forms of corrosion that exist, each of which poses different problems to aircraft structures. The most common types of corrosion observed are discussed below:

- ***Uniform corrosion***, as its name suggests, is corrosion that occurs without appreciable localized attack, resulting in uniform thinning.
- ***Pitting corrosion*** is a localized form of attack, in which pits develop in a material causing localized perforation of the material. Pitting corrosion occurs when one area of a metal surface becomes anodic with respect to the rest of the surface of the material. The pits formed by this type of attack are generally very small and, therefore, difficult to detect during routine inspection. Pitting attack can cause failure by perforation with very little weight loss to the material.
- ***Crevice corrosion*** occurs when localized changes in the corrosive environment exist and lead to accelerated localized attack. These changes in the localized corrosive environment are generated by the existence of narrow crevices that contain a stagnant environment, which results in a difference in concentration of the cathode reactant between the crevice region and the external surface of the material. Crevices can be formed at joints between two materials, e.g. riveted, threaded, or welded structures, contact of a metal with a non-metallic material, or a deposit of debris on the metal surface.
- ***Galvanic corrosion*** occurs when dissimilar metals are in direct electrical contact in a corrosive environment. This results in enhanced and aggressive corrosion of the less noble metal and protection of the more noble metal of the bimetallic couple. This type of corrosion can be recognized by severe corrosion near to the junction of the two dissimilar metals, while the remaining surfaces are relatively corrosion-product free. Galvanic corrosion is generally a result of poor design and materials selection.
- ***Stress corrosion*** cracking is a mechanical-environmental failure process in which tensile stress and environmental attack combine to initiate and propagate a fracture. Failure by stress corrosion cracking is frequently caused by simultaneous exposure to an apparently mild chemical environment and to a tensile stress well below the yield strength of the material. The stress required for failure can originate from in-service conditions or from residual stress during component manufacturing.

Hydrogen embrittlement is a failure process that results from the retention or absorption of hydrogen in metals, usually in combination with applied tensile or residual stresses. It most frequently occurs in high-strength steels (>1100 MPa). For aircraft components, the common source of hydrogen embrittlement is hydrogen absorption during manufacturing processes such as pickling and electroplating.

Investigation procedure

Every investigation has its own unique features and, therefore, it is difficult to describe a set of procedures and techniques that must be employed for all eventualities. However, the principle stages of an investigation will follow along these general lines:

- Recovery and identification of the failed components is generally undertaken by accident inspectors, typically the AAIB in the UK, who also undertake the initial inspection of the components. If metallurgical failure is suspected, then the components are sent for specialist examination. It is also important at this stage of the investigation to gather as much information as possible on the flight and maintenance history of the aircraft, for example when trying to match the evidence of fatigue to structural loading.
- On receipt of the components, each item is methodically recorded and photographed.
- Macro-optical examination is used to identify the failure sites, sometimes supported by non-destructive evaluation techniques such as dye-penetrant inspection, X-radiography, eddy current inspection, and ultrasonic inspection.

- Typically, microscopic examination of the failed components is then undertaken using SEM. If required, qualitative elemental analysis of features can be undertaken during SEM examination by using Energy Dispersive X-ray analysis (EDX).
- Samples are generally taken for metallographic examination using both optical microscopy and SEM techniques.
- Supplementary techniques that are commonly employed for more detailed examination of material microstructures, deposits, and corrosion products can include:
  - Quantitative elemental analysis by Electron Probe Microanalysis (EPMA),
  - Orientation imaging microscopy using Electron Backscattered Diffraction (EBSD),
  - Phase identification and residual stress analysis by X-ray Diffraction (XRD),
  - Surface analysis of deposits using X-ray Photoelectron Spectroscopy (XPS) and Auger Electron Spectroscopy (AES).
- Checks are commonly made on the material composition using analytical chemistry techniques, and the metallurgical condition of the material is assessed using both microscopy and mechanical testing.
- The final stage of the investigation is the generation of a written report of the findings, which will include analysis of all the evidence, formulation of a conclusion, and recommendations for corrective action to avoid similar incidences of failure.

During the investigation, photographic evidence and thorough record keeping of analytical data and procedures is vitally important, as the data may be required for a court case or litigation many years hence.

Case study: Fatigue

A nose undercarriage turning tube was delivered for metallurgical examination after failing catastrophically on landing. It was ascertained that the undercarriage turning tube had undergone 1300 flight cycles during its life, which is well below the expected service life. Therefore, the turning tube was subjected to a detailed examination to identify the cause of the premature failure.

Initially the fracture surface was examined using SEM to determine the fracture mode. This revealed that the majority of the fracture surface showed a ductile appearance consistent with a static ductile overload. However, towards the origin of the overload region, fatigue striations were observed (Fig. 1). This indicated that the fracture mode was initially fatigue, turning to catastrophic fast fracture (overload) when a critical crack length was reached.

The striations that were identified on the fracture surface consisted of distinct bands of repeating units of striation spacing (Fig. 2). The individual striation spacings and band spacings were measured at various points along

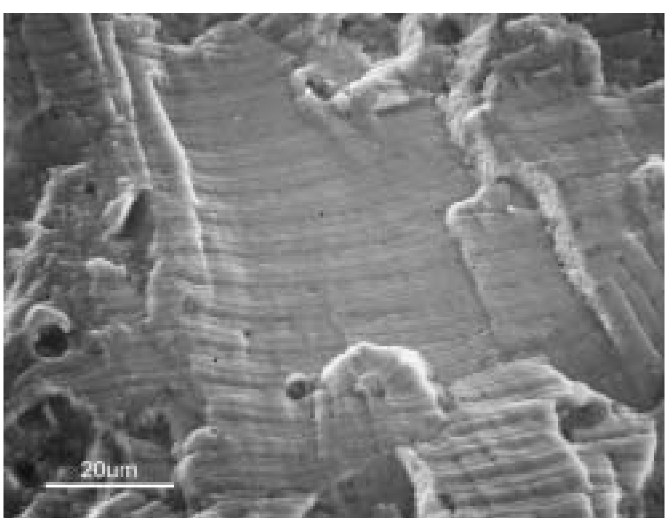

Fig. 1 Fatigue striations observed on the fracture surface.

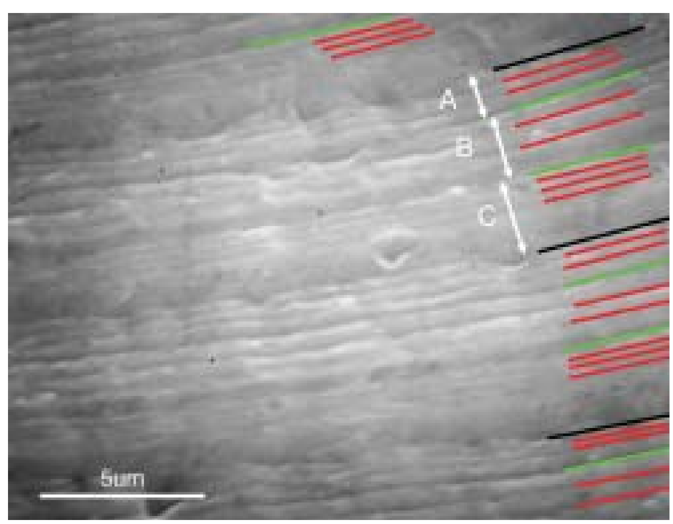

Fig. 2 Distinct bands of repeating units of striation spacing.

Fig. 3 Outer surface of wing panel and door after paint stripping and removal of catches.

the crack length from the origin to the end. These were then plotted to determine the crack growth rate. The striations and bands relate to load cycles and, by comparing them with the anticipated load spectrum, it was possible to determine how many load cycles were required to propagate the fatigue crack to the point of failure. It was found that the number of cycles that caused failure was in the region of the total number of cycles the component was subjected to during its life. This indicated that the fatigue crack had initiated very close to the beginning of its service life.

The material specification was checked and found to be satisfactory, which indicated that the premature failure was not due to a material fault. It was observed, however, that the origin of the fatigue crack occurred at a notch in the surface of the tube. The notch would have produced a stress concentration in the surface of the tube, thus reducing the time required for fatigue cracks to initiate. This notch in the surface, which most likely occurred during manufacture, was attributed as the cause of the premature fatigue failure.

Case study: Corrosion

An upper surface wing panel containing an access door was subjected to a detailed metallurgical examination after corrosion was found during a scheduled service inspection.

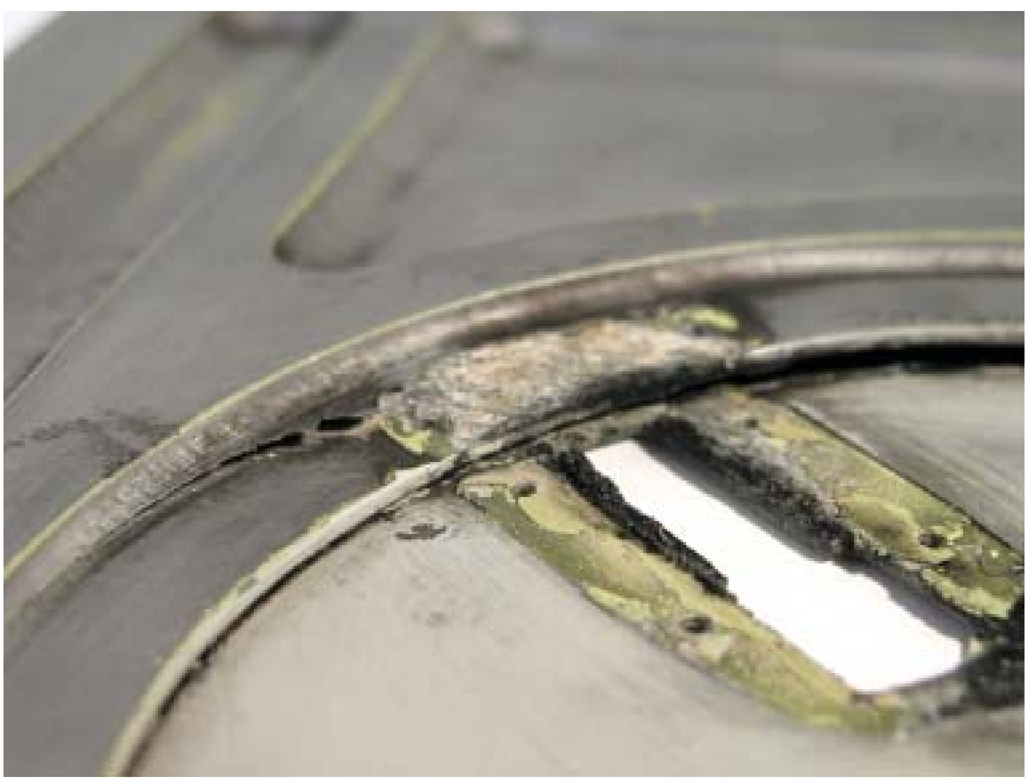

Fig. 4 Exfoliation corrosion on the inner surface of the panel and door around a catch location.

The panel and door were manufactured from aluminum alloy plate to which aluminum catches for securing the door in the closed position were attached. It was found that stainless steel shims had been fitted between the catches and the aluminum plate.

Fig. 3 shows the outer surface of the panel and door after paint stripping and removal of the catches. From the outer surface there appears to be no damage to either the plate or door. However, examination of the inner surface showed extensive exfoliation corrosion on both the panel and door in the catch positions (Fig. 4). Cracking, which appeared to emanate from the catch position, was also observed in the stiffening ribs. A cross-section of the plate was taken through the worst area of corrosion and is shown in Fig. 5.

Exfoliation corrosion occurs in susceptible materials when attack takes place along selected grain boundaries, especially if they are highly elongated and form platelets, which are relatively thin. The corrosion product that is generated is relatively voluminous, which causes the layers of uncorroded material to split apart.

The material was checked and found to conform to the specification, which in this case was an aluminum alloy known to be susceptible to exfoliation corrosion. However, the extent of the corrosion, which was concentrated around the catch positions, indicated that the driving force had been greatest in this area. This increased driving force for corrosion was attributed to the stainless steel shims that were fitted beneath the catches. Although a paint system was present between the stainless steel and the aluminum, over the 25 years that this panel had been in service, the barrier between the dissimilar metals had broken down and allowed contact. Once contact had been established, a galvanic cell was formed in which the more noble metal (stainless steel) had accelerated the corrosion of the aluminum.

Case study: Hydrogen embrittlement

A bolt from an aircraft flap control unit fractured in the threaded region of the shank near the shoulder with the head upon installation after a major service. A metallurgical investigation was carried out to identify the cause of failure. The bolt was manufactured from cadmium-plated, high-strength steel. Material checks carried out on the bolt showed that it conformed to the required specification and was found to have an approximate ultimate tensile strength of 1380 MPa.

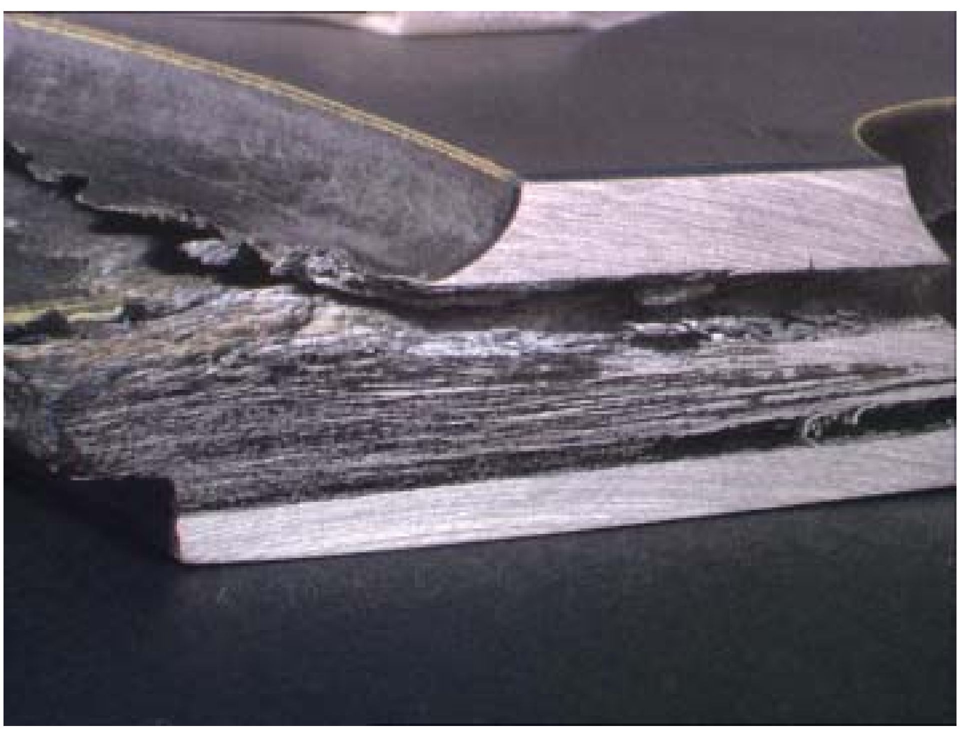

Fig. 5 Cross-section through the panel showing the exfoliation corrosion.

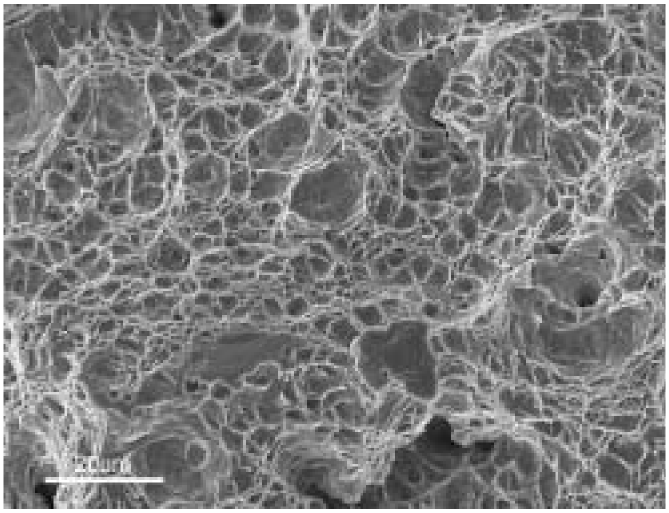

Fig. 6 Ductile fracture surface at the centre of the bolt.

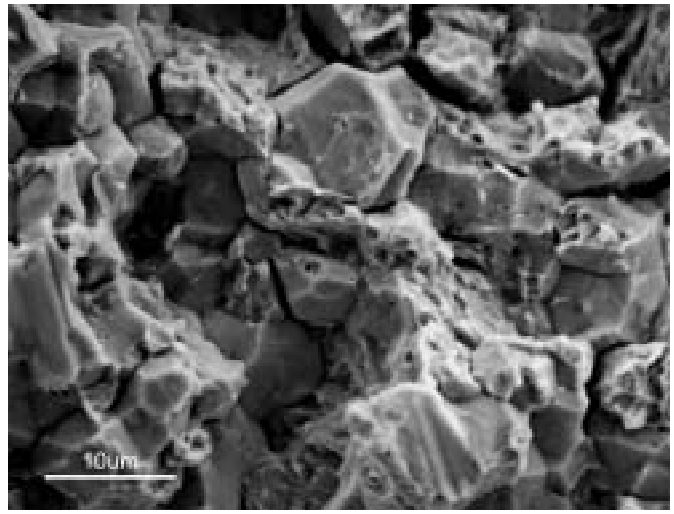

Fig. 7 Intergranular region of the fracture surface around the outer circumference of the bolt.

The fracture surface of the failed bolt was examined using SEM to identify the mode of fracture and determine if pre-existing defects were present that could account for the unexpected failure. The fracture surface exhibited two distinct modes of failure. The center of the bolt exhibited ductile features (Fig. 6), while the outer circumference exhibited intergranular features (Fig. 7). Both modes of crack growth were caused by static overload failure, but the ductile appearance at the center should have been present throughout. The intergranular region around the outer edge was suggestive of embrittlement, which had led to premature failure at loads below those anticipated.

The embrittlement in this case was attributed to the cadmium plating, which is applied to the bolts to provide corrosion protection to the steel. Hydrogen is evolved during the plating process, which becomes absorbed by the steel. The cadmium plating acts as a barrier to hydrogen diffusion at ambient temperature so that the hydrogen becomes 'trapped' in the steel. In high strength steels (>1100 MPa) this leads to embrittlement. To overcome this problem, high strength steel fasteners, which have been cadmium-plated, are baked at 175-205°C for 24 hours to allow hydrogen to diffuse through the cadmium. In this case, failure of the bolts was caused by insufficient baking after plating, which gave rise to hydrogen embrittlement.

Conclusions

Defect and failure investigations on aircraft structural components have an important role in improving aircraft safety. The identification of the primary cause of failure and the subsequent analysis enable recommendations for corrective action to be made that hopefully will prevent similar failures from occurring in the future. MT

REFERENCES

1. Campbell, G.S., and Lahey, R.T.C., (1983) *A survey of serious aircraft accidents involving fatigue fracture, Vol. 1 Fixed-wing aircraft*, National Aeronautical Establishment, Canada
2. Campbell, G.S., and Lahey, R.T.C., (1983) *A survey of serious aircraft accidents involving fatigue fracture, Vol. 2 Rotary-wing aircraft*, National Aeronautical Establishment, Canada