ICAS-88-2.11.4

MECHANICAL FAILURE ANALYSIS AS A MEANS OF IMPROVING QUALITY ASSURANCE IN THE AERONAUTICAL INDUSTRY

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Abstract

Failure analysis is capable of a valuable contribution to the improvement of the quality assurance of aeronautical parts. When failures of these parts are traced to defective manufacture, corrective measures may be taken to improve the quality assurance procedures applied, so as to prevent the re-occurance of such failures. This article presents six detailed case - histories of manufacture-originated failures which occured in the Israeli Air Force (IAF) and were investigated by the IAF Material and Process Engineering Laboratories. All these cases demonstrate the contribution of the laboratory findings to the improvement of the quality assurance systems concerned, in practice.

I. Introduction

Mechanical failure analysis is an interdisciplinary field, incorporating mainly the analysis of mechanical assemblies and metallurgy . The main sources of failures of mechanical components are design, manufacture, maintenance and operations. Thus, failure analysis is instrumental in the quality-assurance (Q/A) of the manufacture of these components and in the planning of their operational life.

The article discusses the role of failure analysis in Q/A of manufacture of aeronautical systems - engines, airframes, and utility systems. The determination of manufacturing defects and irregularities as primary causes of failures that occur before or during operations, often contributes directly to the improvement of the Q/A system concerned. Furthermore, an increased awareness of the "failure potential" of the production line is obtained by analyzing manufacture-originated failures.

This type of failures comprises some 25% of the scope of failures investigated by the IAF Laboratories. Using detailed case-histories of manufacture-originated failures, which occured in the IAF ,this work presents the contribution of failure analysis to Q/A systems in practice. The prime share of the laboratory work consisted of fractographic inspection of fracture/crack surfaces and of metallurgical characterization of failed components, i.e. their actual processing. Visual inspection, fractographic inspection, optical metallography, microhardness testing and chemical analysis methods (EDX, optical emission, Auger, etc) were all employed by the laboratory.

II. SELECTED CASE-HISTORIES OF MANUFACTURE-ORIGINATED FAILURES

Case #1 - Retainer Lock Rings of a Jet Engine

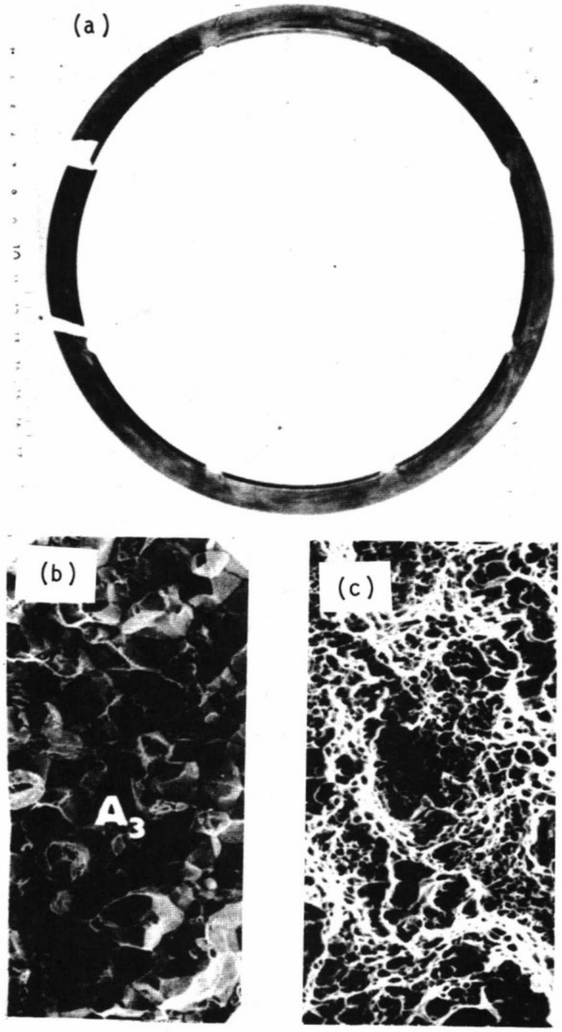
A large number of jet engine retainer-lock rings were reported to have failed upon assembly during overhaul operations. One fractured ring was recieved for laboratory examination (fig 1-a).

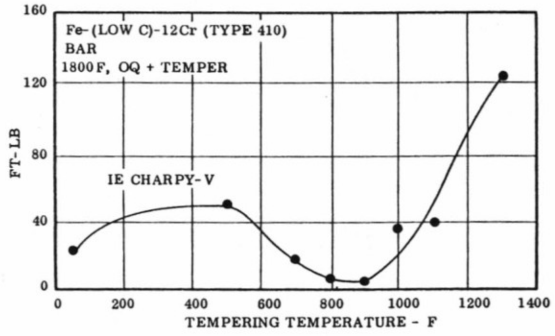
SEM (Scanning Electron Microscope) fractographic inspection revealed the fracture surfaces to be completely intergranular (fig 1-b). Material composition conformed to AISI-410 stainless steel while hardness was found to be 37-38[Rc]. The manufacturer's specification is the standard region of 30-36[Rc] ,whereas the next standard hardness region for that particular steel is 40-46[Rc]. The microstructure of the ring was indicative of quenched & tempered martensitic steel.

Because of the brittle nature of the fracture that occured at the onset of operational life, as well as the abnormal hardness detected, it was suspected that defective heat treatment had induced material brittleness(1). Thus a segment of the ring was re-treated according to MIL-H-6875 : preheated, quenched (austenized) and then tempered at 550±8°C (1020°F). The resulting hardness was 32-33[Rc]. An overload test produced a ductile , dimpled fracture (fig. 1-c) ,as expected. The segment was then re-treated once more in order to reconstruct the defective processing, tempering being performed at 475°C (885°F). This produced 38[Rc] hardness, the same as the hardness of the failed ring.

It was concluded that the repeated failures of the rings were the result of temper embrittlement which reduced the ductility and the flexibility of the rings. The effect of tempering temperature on the brittleness of AISI 410 steel is depicted by fig. 1-d.

For the restoration of defective rings, the IAF laboratories devised a restoration procedure consisting of thermal re-treating and of re-plating. This procedure was successfully applied to the rings at the overhaul depot.





(d) EFFECT OF TEMPERING TEMPERATURE ON IMPACT PROPERTIES OF TYPE 410.

Fig. 1: a) The fractured retainer-lock ring, as recieved (x0.35). b) An intergranular fracture surface (x150). c) A ductile dimpled fracture surface produced by the overload test (x400).

<u>Case #2- **A** Jet Engine Inlet-Case.</u>

Upon regular inspection it was noted that a welded metal strip had partially seperated from the strut of a jet engine inlet-case (fig. 2-a).

The seperation of a 15 cm strip segment occured along the two longitudinal EBW (Electron Beam Welding) seams on both sides of the strip. Macroscopic examination of the fracture surfaces, followed by SEM inspection, indicated that HCF (High Cycle Fatigue) cracks had initiated along the rear welding seam (fig. 2-b). Cracking then progressed transversely, towards the front seam.

Material composition of the strip and of the strut conformed to Ti-6Al-4V alloy .Hardness measured 31-33[Rc]. Metallographic cross-sections taken through the welding seams near the fracture area, showed partial joining or no joining of the strip and the strut along the front seam.This was due to a deviation of the welding beam from the joint, as indicated by the location of the heat-affected spot (fig. 2-c).

It was therefore concluded that an HCF failure of the strip resulted from a manufacturing error. The operator had partially missed the front joint of the strip along some 15 cm while electron-beam welding.Thus, the loose joint caused increased vibrations around the welded joint at the rear side of the strip.

It should be noted that the location and the geometry of the welding discontinuities were such that they were very difficult to detect, even using NDE (Non-Destructive Evaluation) techniques. This problem was further emphasized by the repeated failure of another inlet-case strut by strip seperation.

Corrective action taken by the manufacturer consisted of the introduction of a wider (defocused) beam and improved fixturing. Through these modifications, improved weld coverage and penetration were obtained. It is worth noting that no more failures of this type have been reported since then.

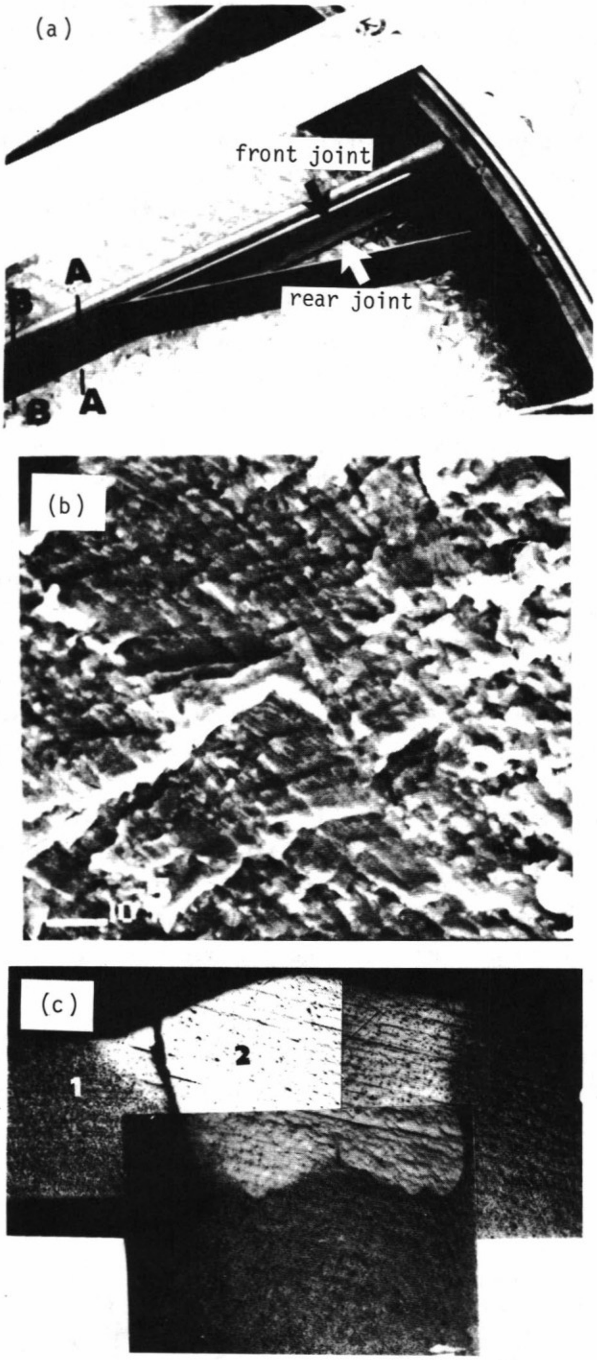


Fig. 2: a) Strip seperation at the inlet-case strut. The metallorgraphic cross-sections are indicated by dashed lines (x0.3). b) HCF striations at the fracture surface (x750). c) Section A-A showing no joining of the strip (1) and the strut (2). The heat-affected spot is light colored (x20).

Case #3- A Mount Bolt of a Helicopter Gearbox.

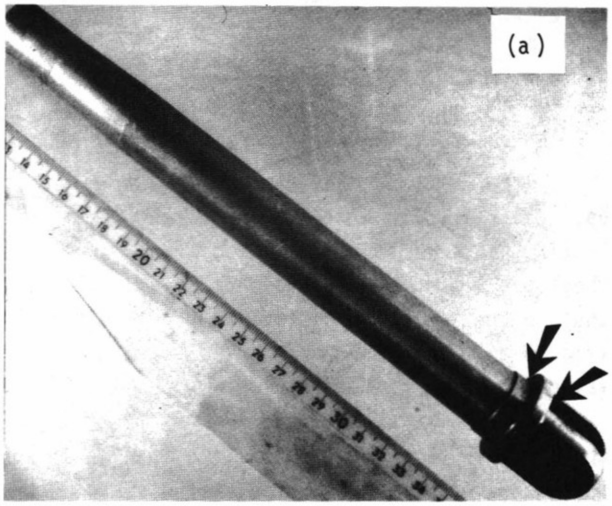
A gearbox mount bolt of a helicopter (fig. 3-a) was found broken following 15 operating hours in the course of two months since its installment.This bolt and another bolt from the same gearbox were recieved for failure investigation.

The failed mount bolt had fractured at the lugs (fig. 3-b) with visible fracture origins distributed along the circumference of the main fracture surface. SEM inspection showed the fracture surface to be predominated by an intergranular structure (fig. 3-c). Micro-pores and fine hairline features were observed on facets of some of the grains(2).

The failed bolt material composition conformed to AISI-4340 steel. The bolt was Cadmium plated in accordance with specification QQ-P-416 as required. However, hardness measured 48-49[Rc] and tensile strength measured 268 KSI, while specification was 40-43[Rc] (180-200 KSI) .The hardness of the other bolt was 48-50[Rc]. No further metallurgical anomalies were detected.

The above findings led to the conclusion that the bolt had failed in a hydrogen embrittlement mechanism due to the deviation in its hardness. The application of QQ-P-416 Cadmium plating is limited to steels of tensile strength lower than 240 KSI, because of the susceptibility of high-strength steels to hydrogen embrittlement even after being properly "baked" (see QQ-P-416, section 3.2.1).

The defective bolts were eventually traced to a batch that was not produced by the helicopter manufacturing company.Consequently the bolt was re-defined a "flight safety " item in order to ensure procurement through the original manufacturer only.



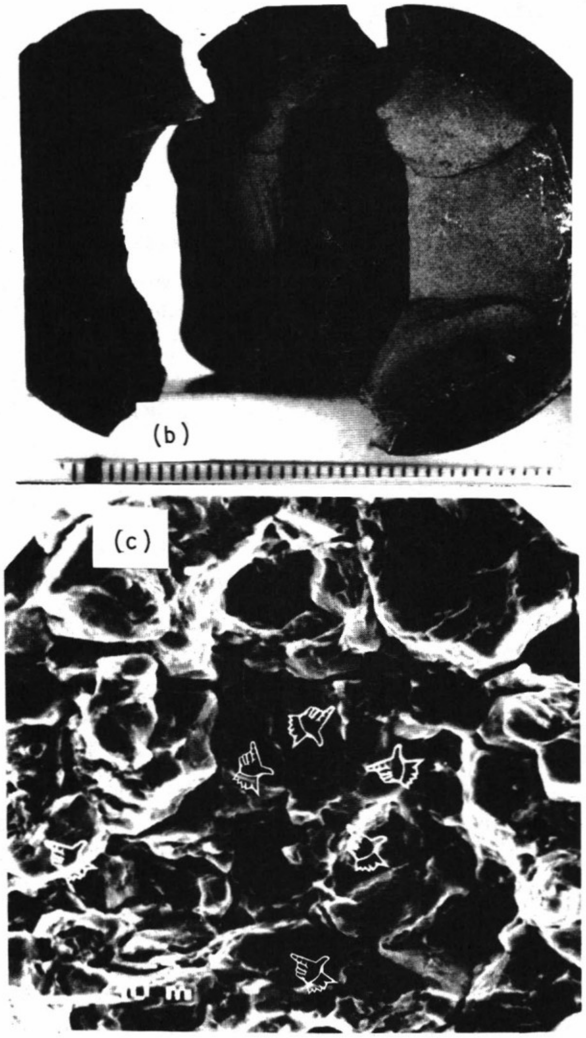


Fig. 3: a) A helicopter gearbox mount bolt with the failure location indicated (x0.75). b) The fracture surfaces of the failed bolt (x1.6). c) SEM micrograph showing the intergranular structure of the fracture. Micro-pores and hairline features, characteristic of hydrogen embrittlement, are indicated (x1125).

Case #4 - A Centrifugal Turbine Wheel of a Jet Starter Engine.

A jet starter engine exploded during start-up, following some 200 operational start-ups. Initial investigation revealed that the turbine wheel was broken across its center into two halves, thus bisecting the central bore longitudinally (fig. 4-a). This bore was manufactured by EDM (Electro-Discharge Machine) process.

Two dark areas, coarse in macroscopic appearance, were observed at the fracture surface on both sides of the EDM bore (fig. 4-b). Numerous short cracks, 0.2 mm long, were detected at the bore surface (fig. 4-c).

The microscopic fracture features of the dark areas, as observed by SEM, were characteristic of time-dependent crack propagation. The initial mode of propagation was intergranular, eventually changing into a mixed intergranular - transgranular mode. Fatigue striations were discernible on cristallographic planes up to 0.25 mm from the bore surface, and crack origins were identified along the bore's edge. Striation density was estimated at 300-800 striations/mm, probably indicating LCF (Low Cycle Fatigue).

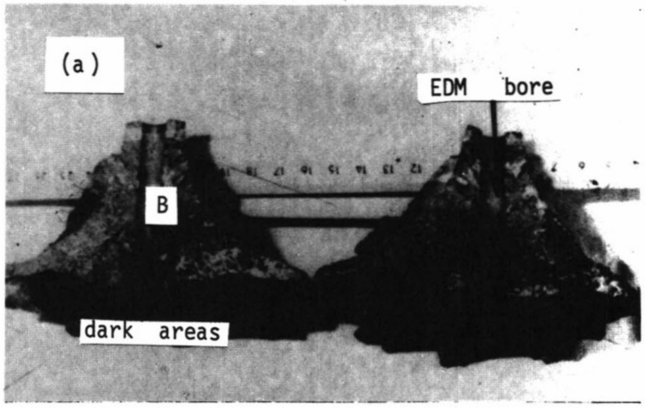
The turbine wheel material was Inconel 713 casting of hardness 34-35[Rc]. Metallographic inspection revealed an EDM recast layer, 2.5-70 microns deep, at the bore surface, as well as intergranular cracks traversing from this surface. Micro-hardness testing of the recast layer indicated 54[Rc] hardness.

Based on these findings and various inquiries at the manufacturer, the following conclusions were drawn(3):

1) The turbine wheel failure occured in a high stress, high temperature fatigue mechanism related to the number of engine start-ups.

2) Cracks were formed at the bore surface during the EDM process or during the initial turbine runs. Due to the hard and brittle recast layer ,the crack initiation process was essentially completed by the beginning of the turbine's operational life. This resulted in a severe reduction of the fatigue life of the turbine wheel.

The occurance of this failure, as well as several similar cases brought to the manufacturer's attention, impelled the manufacturer to make the following modifications :as a first step, a bore of smaller diameter would be formed by EDM. Then, the bore would be mechanically machined to its final dimensions, thus eliminating the recast layer problem.



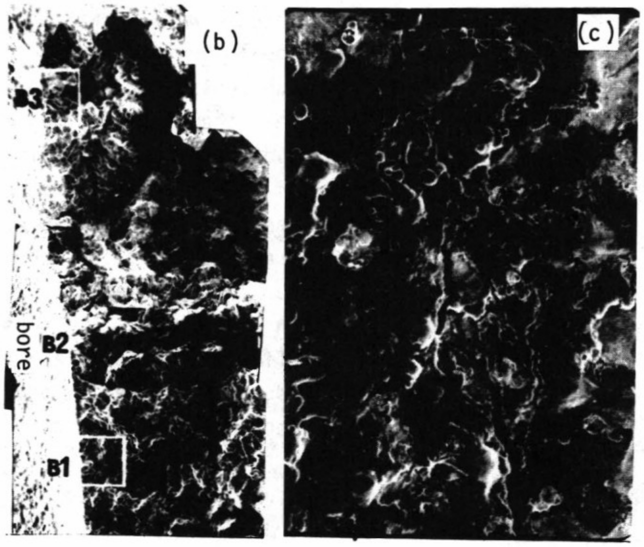


Fig. 4: a)The two halves of the failed turbine wheel (x0.35). b) A magnified view of area B, near the EDM bore (x7.5). c) Cracks at the bore surface (x75).

Case #5 - Tail Wheel Helical Springs.

Following the failure of a number of helical springs during handling before assembly, numerous other springs from the same batch were found cracked in storage.

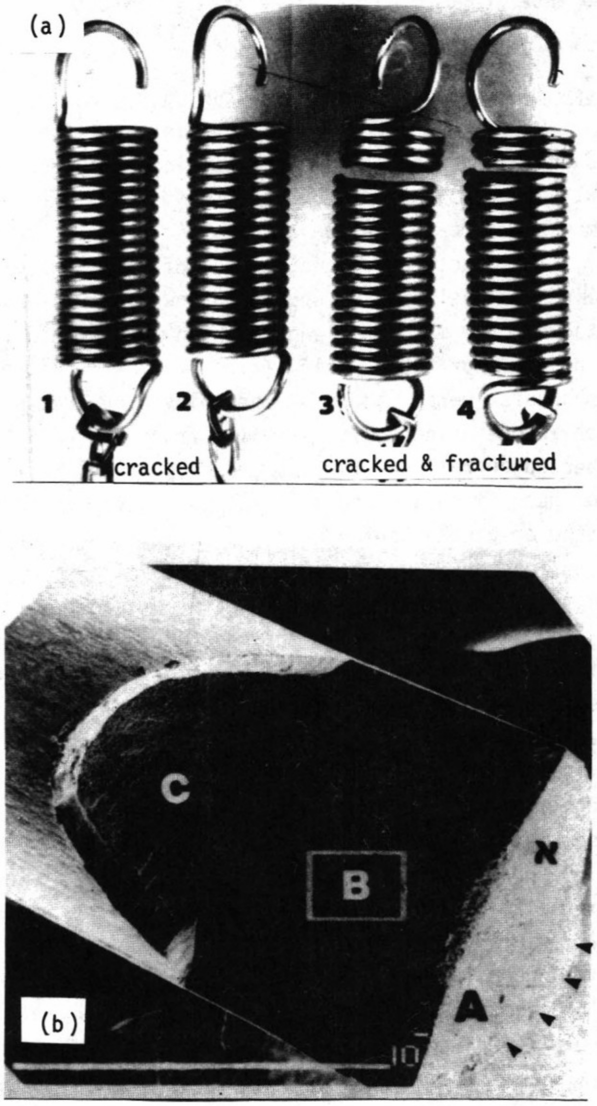
Visual examination of a sample of springs revealed a multiplicity of transverse fractures and cracks in the coils (fig. 5-a). Metallurgical inspection showed the springs to be made of AISI-6150 steel, heat treated to 48-52[Rc] and Cadmium plated in accordance with QQ-P-416, as required. The surfaces of the fractures and cracks, as observed using SEM ,were intergranular (figs. 5-b & 5-c). Penetrations of Cadmium plating were detected by EDX analysis at the edges of these surfaces.

The fractographic findings as well as the occurence of such failures in storage suggested that hydrogen embrittlement, induced by the electrolytic plating, might have contributed to the failure process. A qualitative hydrogen embrittlement test was performed, in which a cracked spring was subjected to a sustained load of 10.6 kg, this being the spring's upper design load limit. That spring failed after 11 hours at the location of one of the cracks.

Upon inquiry the manufacturer reported that two days had elapsed between the completion of the coil-winding procedure, and the ensuing stress relief treatement .It should be noted that the relevant specification strictly requires that stress relief be carried out "immeidiately" after winding the coils or within 24 hours. It was not possible to reconstruct the full manufacturing process in the lack of a factory routing system.

The failure of the springs was attributed to defective manufacturing, the cracks having occured during plating or prior to the electrolytic Cadmium plating process. The stress relief treatement as reported by the manufacturer, was deemed ineffective, for it was reasonable to assume that the failure resulted from high residual stresses combined with hydrogen embrittlement$^{(4)}$.

Following the investigation corrective measures were taken by the IAF Q/A facilities to ensure the manufacturing of these springs according to specification requirements.



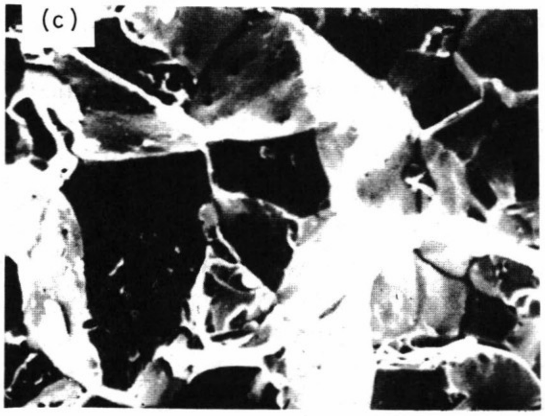


Fig. 5: a) A sample of failed springs (x0.5). b) A characteristic fracture of a coil. Areas A, B are intergranular. Cadmium penetrations were detected along the arrowed edge (x15). c) SEM micrograph of the intergranular structre of area A (x4000).

Case #6 - A Jet Engine Combustor Duct .

During a jet aircraft training flight a sudden explosion occured, resulting in an engine shut-down. It was discovered that part of the combustor duct was blown off.

The duct was constructed of two semi-cylindrical halves. The upper and lower halves were attached by mating flanges, which were welded to both sides of each half (fig. 6-a). All welded components were made of Chromoloy. The duct had been in operation only a small fraction of its expected operational lifespan. This particular duct had been manufactured by a sub-contractor and not by the original manufacturer.

The laboratory findings were as follows:

1)A fatigue crack was generated at the inner side of the weld joint of the upper half. The crack initiated at numerous origins and progressed radially outward to form a semi-elliptical through-crack, 83 mm long, in the cylinder wall. The rest of the fracture surface was characteristic of ductile overload. No surface irregularities were observed at the origins.

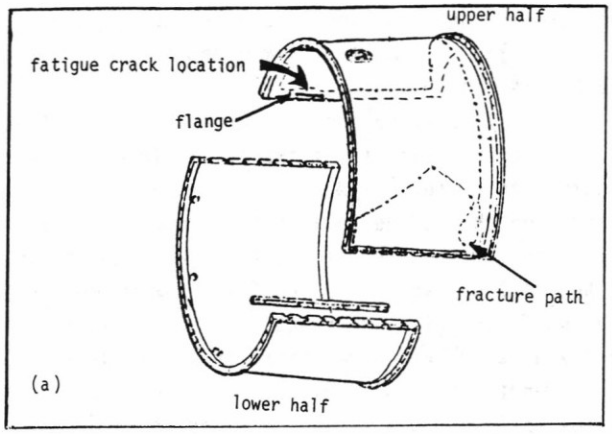
2) Secondary fatigue cracks were generated at the same location in the opposite joint of the upper half (fig. 6-b).

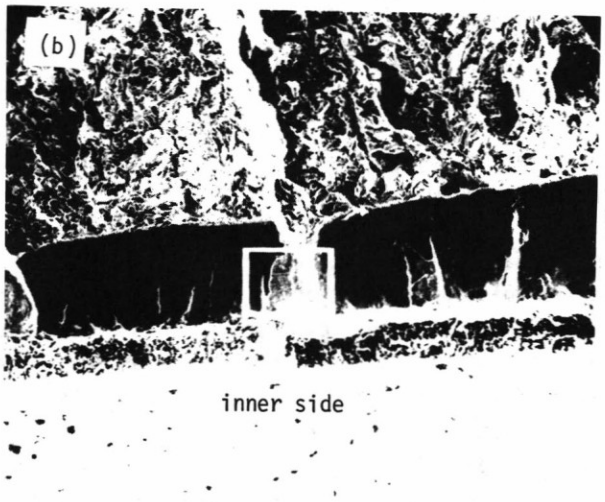
3) A considerable deviation from the specified joint geometry was measured at the failed side, i.e. 1.6 mm mismatch as compared with the max. allowable 0.8 mm, and approaximately 13° joint angle as compared with the max. allowable 7°. Similar deviations were measured at the other joints as well (fig. 6-c).

Further dimensional checks were carried out on a sample of operational ducts that were manufactured by the same sub-contractor. None of these ducts complied with the joint angle requirement, measurements reading as high as 14°.

Finite element stress analysis established that the dimensional deviations at the failed joint had increased the tensile stress level at the inner side of the joint by a factor of at least 2.5. Using fatigue crack growth analysis it was determined that the deviation in the joint angle had a major effect on the reduction of the fatigue life of the duct (the effect of the mismatch was secondary).

The IAF required the sub-contractor to re-qualify for the manufacture of the combustor duct. Hence, the flange welding process was modified so as to obtain better control of the joint geometry. The joint inspection technique was also improved. Furthermore ,each duct had been given individual documanetation for data of manufacture as well as depot operations.





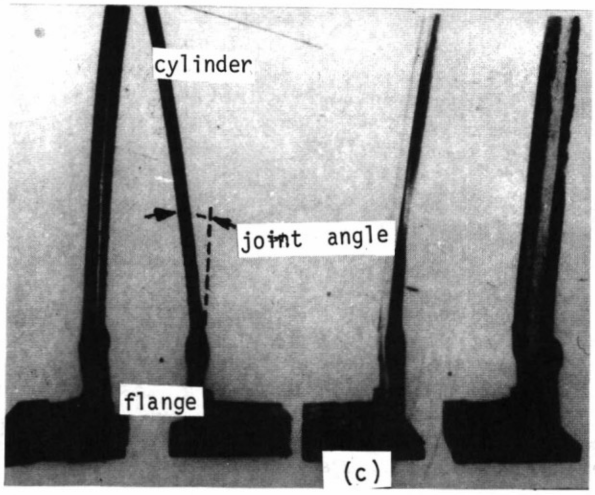


Fig. 6: a) A schematic drawing of the failed duct. b) SEM micrograph of secondary fatigue cracks (dark in color) penetrating from the inner side of the cylinder wall (x40). c) Transverse sections through the duct depict the geometric deviations at the joints .

III. Summary

The Q/A systems employed by either the manufacturer or the customer, are not always capable of predicting deficiencies in the manufacture design, the qualifying inspections and the procurement of aeronautical parts. Defective parts which passed all Q/A procedures will generally fail within a short period of operational life, as indicated by the above case-histories (see also fig.7). It is then that failure analysis is required in order to distingush manufacture-originated failures from others, and to provide the Q/A system with feedback information. This information serves to rectify the above mentioned deficiencies, as related by this article. Therefore, the incorporation of a failure analysis function in Q/A constitutes a substantial factor in the minimization of failures of the type presented herein.

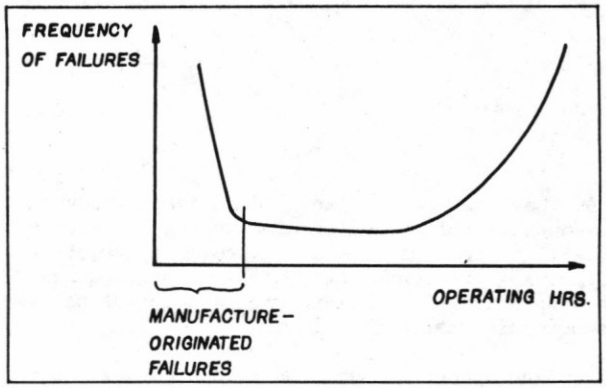


Fig. 7: A qualitative display of the frequency of failures in mechanical parts, as a function of operational life.

IV. References

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