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| **Aircraft Type and Registration:** | McDonnell Douglas DC-8-63F, 9G-MKO |
| **No & Type of Engines:** | 4 Pratt & Whitney JT3D-7 turbofan engines |
| **Year of Manufacture:** | 1971 |
| **Date & Time (UTC):** | 29 April 2003 at 1500 hrs |
| **Location:** | RAF Lyneham, Wiltshire |
| **Type of Flight:** | Public Transport (Cargo) |
| **Persons on Board:** | Crew - 4 Passengers - None |
| **Injuries:** | Crew - None Passengers - N/A |
| **Nature of Damage:** | Right main landing gear fractured |
| **Commander's Licence:** | Airline Transport Pilot's Licence |
| **Commander's Age:** | 43 years |
| **Commander's Flying Experience:** | 10,200 hours (of which around 3,500 were on type)<br>Last 90 days - 220 hours<br>Last 28 days - 85 hours |
| **Information Source:** | AAIB Field Investigation |

## Synopsis

The aircraft's right main landing gear suffered extensive fracturing of its shock-strut piston as the aircraft was making a 180° taxiing turn. Associated disruption to the landing gear scissor linkage allowed the landing gear truck to diverge approximately 45° from the aircraft's heading, but one of the broken parts of the piston remained jammed in the shock-strut cylinder and continued to support the aircraft. Around 90% of the specified overhaul life of the landing gear remained at the time of the accident.

Specialist examination indicated that the piston material was in accordance with the aircraft manufacturer's specification. The fractures had originated from a small pre-existing stress corrosion crack in an area of the surface where cadmium plating was absent. The crack had probably been initiated by abnormally high local stresses associated with a step in a blend radius in the region of the crack origin and with surface scratches in the area. These features should have been apparent during the last overhaul of the landing gear. The pre-existing crack, while small, was probably sufficient to cause the rapid extensive fracturing of the piston under normal operating loads, given the notch sensitivity of the high-strength steel from which it was made.

One safety recommendation, relating to the Federal Aviation Administration's oversight of overhaul organisations has been made.

## History of the flight

The aircraft had been prepared for a cargo flight from RAF Lyneham to the Middle East, with a take-off weight of 297,996 lb (135,170 kg). The weather at Lyneham was good, with a surface wind of approximately 200º/15 kt. Runway 24 was in use; this has an asphalt surface that was damp at the time of the accident. The runway is 148 feet (45 metres) wide; a paved protrusion on the right side of the runway at its start provides a turning area 206 feet wide and 200 feet long.

After taxiing to the runway, the aircraft backtracked to the turning area and commenced a 180° right turn to line-up for takeoff. With around 20° of the turn remaining the crew heard a loud noise and the aircraft came to a halt. Engine power was increased but the aircraft did not move; the crew realised that there was a technical problem and requested assistance. Ground engineers found that parts of the right main landing gear (MLG) had fractured and its truck had rotated approximately 45° left relative to the aircraft's heading. The aircraft was then shutdown. It remained on the runway for around 36 hours while the right MLG was replaced.

## Aircraft description

The McDonnell Douglas DC-8-63 is a four-engined freight aircraft of conventional layout with a tricycle landing gear. Maximum authorised taxi weight is 358,000 lb (162,388 kg) and maximum authorised take-off weight 353,000 lb (160,120 kg).

Manufacturer's data on ground manoeuvring indicates that the minimum pavement width for a 180° turn is 139 feet, using the maximum nose landing gear steering angle of 67°. Cautions were given in the Aircraft Maintenance Manual and the Operation Manual against using MLG wheel braking to tighten a turn.

## Main landing gear description

The MLG leg (Figure 1) consists of a cylinder/piston type oleo shock-strut. The shock-strut cylinder is attached to the wing structure and a four wheeled truck (bogie) is connected to the lower end of the piston by a pivot pin. The vertical load on the piston is supported by a pressurised nitrogen charge within the cylinder and shock absorption is provided by translation of the piston within the cylinder, controlled by regulating the flow of oil between chambers within the oleo. Rotation of the piston relative to the cylinder is prevented by a scissor linkage, consisting of two torsion links, connected together by a horizontal apex pivot joint. The lower torsion link is attached to the piston via a horizontal pivot joint and the upper torsion link is similarly pivoted onto the cylinder.

The piston is approximately 44 inches long and its main part is 9.5 inches in diameter with a 0.63 inch wall thickness. It is closed at its base and carries a number of integral clevis lugs. Two lugs extending below the piston base form a clevis for the truck beam pivot pin and the truck beam fits between these lugs. Two other lugs, formed in an integral boss on the inboard side of the piston near its base, form the clevis for the lower torsion link pivot pin. The bearing faces of these lugs, where they contact the torsion link, are chromium plated.

The piston is manufactured from 4340 steel, a relatively high-strength chromium-nickel-molybdenum steel alloy. It is machined from a forging defined by Douglas Material Specification (DMS) 1555H, and heat-treated to a ultimate tensile strength of 260-280 ksi (thousand pounds per square inch). This is followed by shot peening to reduce residual surface tensile stresses, by cadmium plating for corrosion protection and by hard chromium plating of load bearing regions. The overhaul life specified by the aircraft manufacturer for the shock-strut was 23,000 flight hours; no flight cycle limit was specified. Repair schemes include chromium plating of the plane faces of the cylinder and piston clevis lugs to which the torsion links attached.

### Maintenance background

Maintenance records indicated that at the time of the accident 9G-MKO (Serial Number 46147, previously registered as N811CK) had accumulated 48,076 flight hours and 16,533 flight cycles. Its last major check, a 25,000 hour D Check on 28 June 1998, had been 4,944 hours prior to the accident and its last C Check (3,000 hours), on 12 December 1999, had been 2,382 hours before the accident.

The right MLG (Part Number 5759101-5018, Serial Number HA002) had been fitted at the time of the C Check after it had been overhauled in the USA. It had been removed for overhaul from another aircraft and released to service in September 1999 as a zero time unit; documentation did not indicate that the shock-strut piston had been replaced during the overhaul. Subsequent to the C Check, the aircraft had operated for 453 flight hours on the USA register, before being purchased by the operator at the time of the accident, 1,926 flight hours before the accident. Thus at the time of the accident the right MLG had accumulated 2,379 flight hours since its overhaul, around 10% of the specified overhaul life.

### Wreckage examination

Initial examination revealed that the piston of the right MLG shock-strut had sustained extensive fracturing that had separated the piston into four main pieces and a number of fragments. A generally axial fracture passed through the base of the two lower torsion link clevis lugs and through the aft limb of the left (inboard) truck beam clevis lug (Figures 2 and 3). From this fracture, two fractures extended up the left side-wall of the piston. Around 9 inches from the base of the piston one of the fractures had become generally circumferential, separating the lower part of the piston into two pieces, detached from the remainder of the piston, with each piece including one of the truck beam clevis lugs.

The right hand piston piece remained jammed into the cylinder and continued to support the aircraft weight. The smaller piston piece, on the left side, remained attached to the lower torsion link. Further fractures extended up the piston (Figures 4 and 5) beyond the circumferential fracture and separated an additional 6 inch length of the piston, which also remained in situ.

The upper torsion link of the scissor linkage remained intact and connected to the cylinder. The lower torsion link had fractured at the apex pin connecting it with the upper torsion link. At its other end, the lower torsion link remained intact, but the unfractured forward limb of the left truck beam clevis lug had twisted around 40°. Deformation and markings indicated that both the torsion link fracture and the lug twisting had resulted from loads transmitted by the scissor linkage as the cylinder had rotated clockwise relative to the truck beam, as viewed from above.

### Detailed materials assessment

The shock-strut was disassembled and a detailed specialist assessment of the piston and lower torsion link fractures was made, together with an investigation of the piston material. This found that the majority of the fractures had resulted from sudden catastrophic overload. Directional features on the fracture surfaces indicated that the origin of the failure had been at the edge of the aft lug of the clevis for the lower torsion link, at a blend radius at the edge of the chromed face forming a bearing surface for the torsion link.

Metallographic examination of a sample taken from the region of the suspected origin found that the microstructure consisted of tempered martensite, which is typical of 4340 steel alloy. Energy dispersive X-ray (EDX) analysis of the sample yielded a material spectrum that was also fully consistent with 4340 steel alloy. Hardness testing showed that the average hardness was 539 HV30, equivalent to a tensile strength of 269 ksi and thus within the specified range.

Detailed examination of the suspected region of the fracture origin revealed an area of staining approximately 5 mm (0.197 inches) across and 2 mm (0.078 inches) deep at the edge of the fracture (Figure 6). Examination by scanning electron microscopy showed that the fracture surface outside of the stained region exhibited ductile dimples, characteristic of static overload failure. Within the stained region, crack branching and corrosion product deposits were evident on the surface and the fracture surface had a more intergranular morphology, characteristic of stress corrosion cracking.

The origin area showed signs of three distinct discoloration bands, with evidence that the initial crack had originated at two points (heavily stained areas). The markings indicated that these cracks had combined into a single crack, which had progressed over an area that was less discoloured and then over a further heavily stained area. There was no clear evidence as to the cause of the banding but it may indicate that the crack had been retarded during its growth for a period, possibly due to a temporary reduction in the stress in the component.

It was apparent from the examination that there was a step in the blend radius in the area of the crack origins and mechanical damage locally that was consistent with machining damage in the region (Figure 7). Additionally, EDX mapping showed that, while cadmium plating was generally present on the surface adjacent to the crack origin, it was absent near the edge of the crack (Figure 8). As cadmium is relatively soft and unlikely to spall off as a result of cracking, if correctly applied, it appeared that it had either not been applied in the area of the crack, had not been applied correctly or had been removed during service.

Inspection of the microstructure using nital etching of the surface around the fracture origin and of a polished transverse section through the origin showed no signs of grinding burns. Examination for evidence of shot peening of the surface was inconclusive.

The specialist analysis noted that steels with strength levels greater than a corresponding hardness of around 400 HV are acutely susceptible to stress corrosion cracking, a process where intergranular fracturing of a component under sustained tensile stress in a corrosive atmosphere can occur. A moist air environment would probably be sufficient to constitute such an atmosphere in the case of high-strength steels such as 4340. It was also noted that, while cadmium plating would prevent exposure to the corrosive atmosphere, and thus prevent stress corrosion cracking, in practice it was likely that service use would eventually cause the cadmium plating layer to be compromised in places.

The susceptibility to such cracking would be particularly increased by the presence of a stress raiser, such as a step or nick in the surface. A Boeing article on the Maintenance of High-Strength Alloy Steel Components, published in 2003, also noted the notch sensitivity (sensitivity to stress concentrations) of high-strength alloy steels, defined as steels that generally had been heat-treated above 180 ksi, and emphasised the necessity of proper rework practices. It noted that stress concentrations can lead to initiation of cracking by stress corrosion and other mechanisms, that abrupt changes in sections should be avoided and that finer surface finishes may be needed to eliminate unnecessary stress concentrations, especially in areas of machined radii.

The assessment concluded that 9G-MKO's right MLG piston material was 4340 steel alloy that had been correctly heat-treated. The piston had suffered catastrophic failure due to stress corrosion cracking that originated at a step in a blend radius in the truck beam pin lug where surface scratching damage was also present and cadmium plating was absent. The critical crack length in 4340 steel, ie the depth of crack which could result in rapid extension of the fracture under normal operating stresses, is small, and the catastrophic failure resulting from a stress corrosion crack only 2 mm deep was not considered unusual.

**Previous cases**

Information on two previous similar cases was found, detailed in McDonnell Douglas All Operator Letters (AOL), as follows:

> AOL 8-1182 (2 June 1994) that noted:
>
> *"One operator recently experienced failure of a right hand main landing gear (MLG) piston assembly while negotiating a 180 degree turn during aircraft positioning for takeoff. The results of the investigation, which was performed by the Civil Aviation Authority of the country in which the event occurred, concluded that the failure was due to overload. The overload was a direct result of a tight 180 degree turn on the runway while using differential braking. The speed at which the turn was made was also a contributing factor. In order to preclude additional failures of this nature, operators are reminded to adhere to the referenced MM and Operations Manual chapters which state:*
>
> *Reference (a):* [Aircraft Maintenance Manual (Chapter 9-2-0, page 1)] *"<u>CAUTION</u>: PIVOTING ABOUT A SET OF BRAKED WHEELS IS PROHIBITED".*
>
> *Reference (b):* [Operation Manual (Chapter 2-143, page 1)] *"<u>CAUTION</u>: DO NOT BRAKE MAIN WHEELS ON INSIDE OF A TURN TO OBTAIN A SMALLER TURN RADIUS THAN IS POSSIBLE BY MAXIMUM NOSEWHEEL STEERING"."*

AOL 8-1218 (12 October 1995) that noted:

*"A DC-8 operator reported that during takeoff the pilot noticed an abnormal vibration which prompted an aborted takeoff .*

*The aircraft proceeded to turn off the runway onto the taxiway. At that point, the pilot was unable to taxi the aircraft any further. Upon subsequent inspection, it was determined that the right hand main gear piston had fractured near the bogie beam pivot lugs.*

*Analysis of the failed piston revealed that the failure was attributed to stress corrosion cracking. The primary crack origin area of the piston occurred along the inner radius of the torque link [torsion link] attach lug. There were two secondary fractures located along the lower aft surface of the piston. Each of the fracture origins revealed a predominant intergranular mode of rupture most likely due to hydrogen assisted cracking.*

*The contributing factor in cases associated with stress corrosion cracking is an exposed surface exhibiting corrosion due to the loss of surface protection. The exposed surfaces of all landing gear components should be continually monitored for paint erosion or loss of cadmium plating to lessen the potential for additional failures. Maintaining an appropriate surface protection will preclude failures associated with stress corrosion. The chapters noted in referenced OHM* [Overhaul Manual] *provide the necessary information to touch up the primer and paint as well as to brush cadmium plate small areas of damage on landing gear components."*

**Discussion**

It was clear that, as the aircraft was making a 180° taxiing turn, the piston of the right MLG shock-strut had suffered extensive fracturing into a number of pieces from an origin in the region of the attachment lugs on the piston for the scissor linkage. The manoeuvre was likely to have increased the stresses in the fracture origin area, as indicated by one of the previous cases of MLG shock-strut piston fracture, but no evidence was available to indicate whether the design stress had been exceeded. However, there was clear evidence that the piston had a pre-existing crack that, while small, was probably deep enough to result in rapid catastrophic fracturing under normal operating stresses, given the notch sensitivity of the high-strength steel from which the piston was made.

The evidence indicated that the initial crack had resulted from a stress corrosion process, caused by the effects of a sustained tensile stress in the area in a damp environment. The step in the blending radius and the surface scratches in the region of the crack origin would have acted as local stress raisers that probably led to the initiation of the stress corrosion cracking and the local absence of cadmium plating would have assisted the process.

There were no signs that the step and the scratches could have been formed in service and it appeared likely that they had resulted from machining operations during overhaul of the piston, possibly when the chromium plating on the associated lug had been replaced. The reason for the local absence of cadmium plating could not be established but did not appear to have been due to wear in service. No evidence was available to determine when the features had been formed, but they should have been apparent during the last overhaul of the landing gear.

Detection of the pre-existing crack was unlikely to have been possible, given its location and size, without disassembly of the MLG. Only around 10% of the MLG life had been used since its last overhaul, and there was no maintenance requirement for the area to be subjected to detailed inspection in this period.

**Previous safety recommendation**

In December 2002 the nose landing gear outer cylinder of a Boeing 747-367 fractured as it taxied for takeoff from London's Heathrow Airport. The nose landing gear outer cylinder suffered a circumferential fracture as a result of fatigue cracking initiating from a groove at the upper edge of the internal diameter seal band. The groove had been inadvertently machined into 70% of the internal circumference during overhaul in the USA. The AAIB made two safety recommendations directed at the FAA as a result of the accident investigation. One recommendation concerned the retention of maintenance/overhaul records and the other (Recommendation 2004-70 made on 28 October 2004) was as follows:

> *Safety Recommendation 2004-70*
>
> *It is recommended that the Federal Aviation Administration (FAA) adopt a programme for performing targeted surveillance and increased oversight of overhaul practices at '14 Code of Federal Regulations Part 145' Repair Stations that are conducting repair, overhaul and rework of aircraft landing gears, to ensure that the manufacturer's overhaul manuals and instructions are followed and that appropriate quality assurance procedures are in place for the continued airworthiness of these components.*

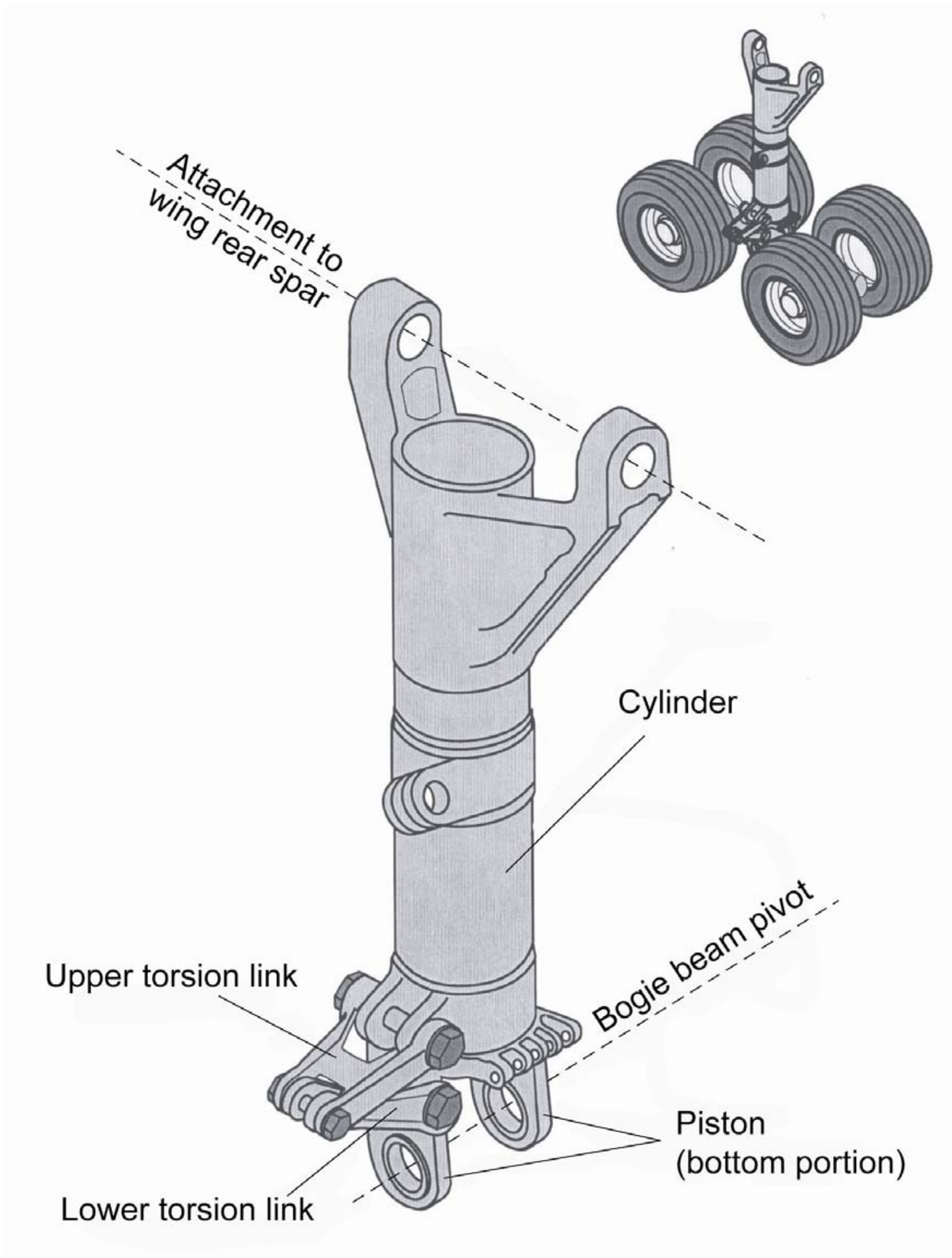
To date no response has been received from the FAA addressing either recommendation.

**Recommendation**

The evidence, gathered in the investigation into 9G-MKO's MLG failure, indicated that a poor profile and surface finish on part of the MLG shock-strut piston had probably been responsible for the initiation of a stress corrosion crack that would most likely have been undetectable in service but had led to extensive fracturing of the piston. These detrimental features should have been apparent during the last overhaul of the landing gear and the previous investigation highlighted similar areas for concern with the quality control at another overhaul agency. Even though the investigations highlighted problems in the quality control present within organisations involved with the overhaul of aircraft landing gear the problem of less than adequate quality control may be more widespread generally amongst FAA approved aircraft component overhaul organisations. The following recommendation is therefore made:

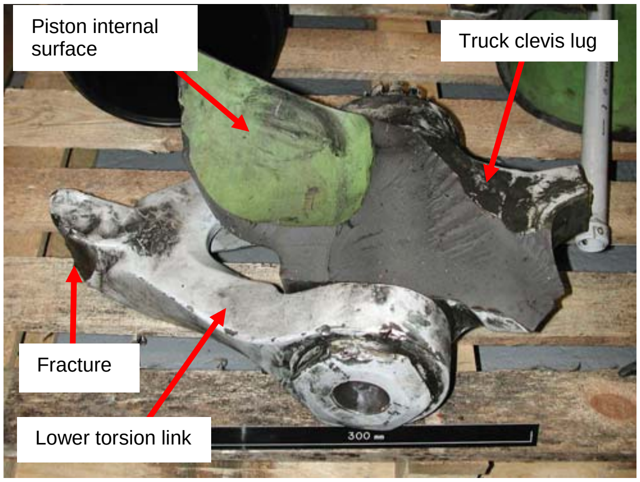
**Safety Recommendation 2005-04**

It is recommended that the Federal Aviation Administration (FAA) take measures aimed at ensuring that overhaul organisations approved by them have in place adequate standards of quality control.

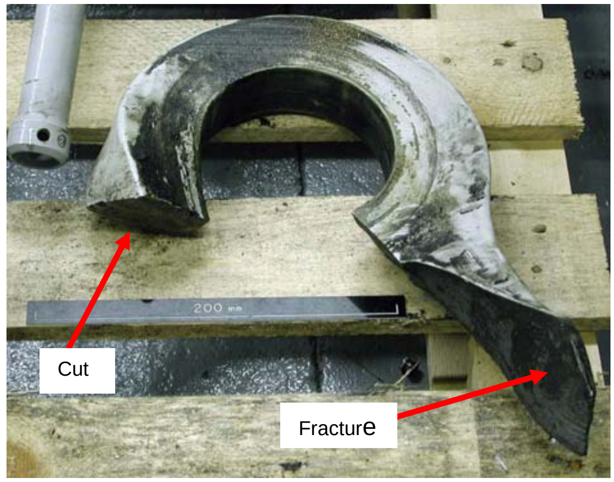



Main Landing Gear
**Figure 1**

## Main landing gear piston




Part of piston and lower torsion link
**Figure 2**

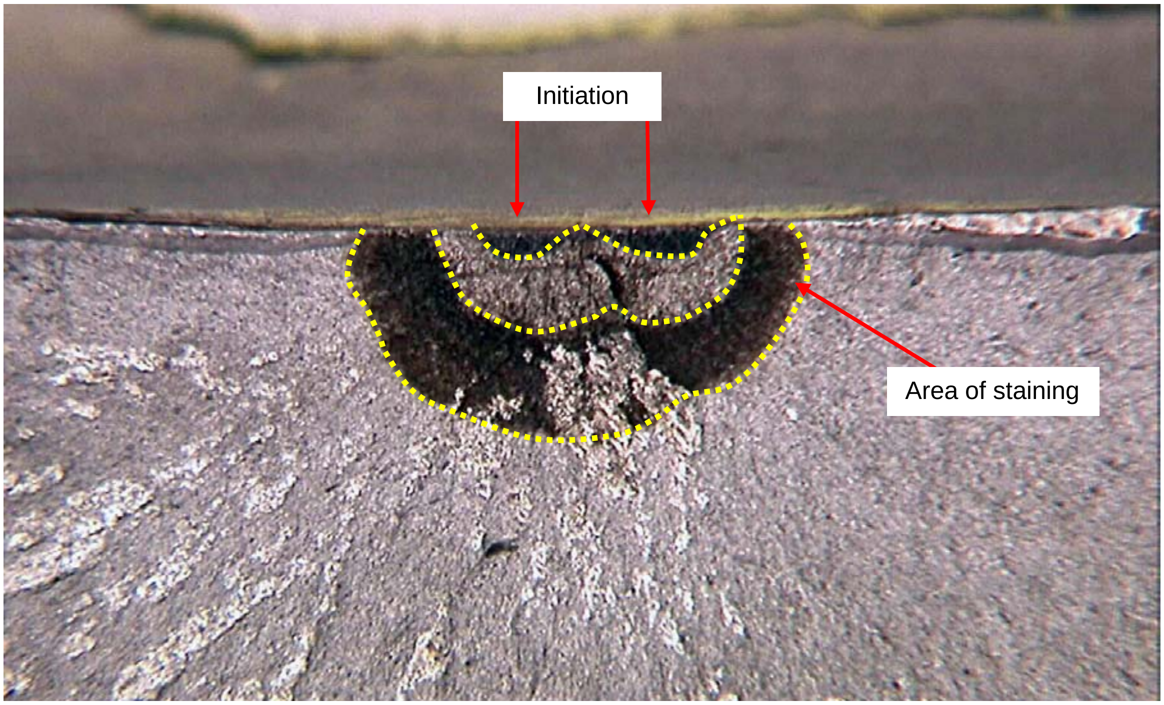



Truck clevis left lug
**Figure 3**

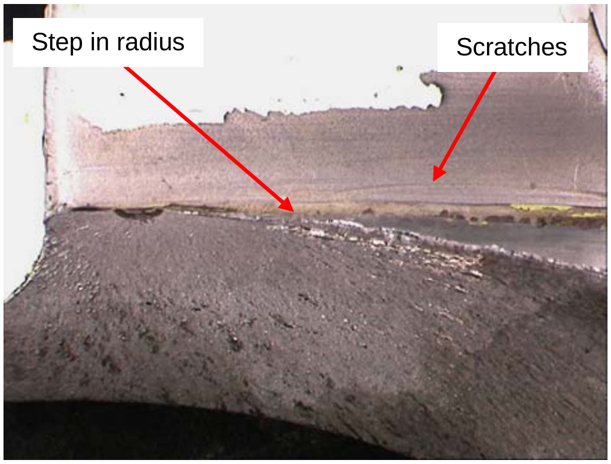

Piston upper part
**Figure 4**


Separated portion of piston
**Figure 5**

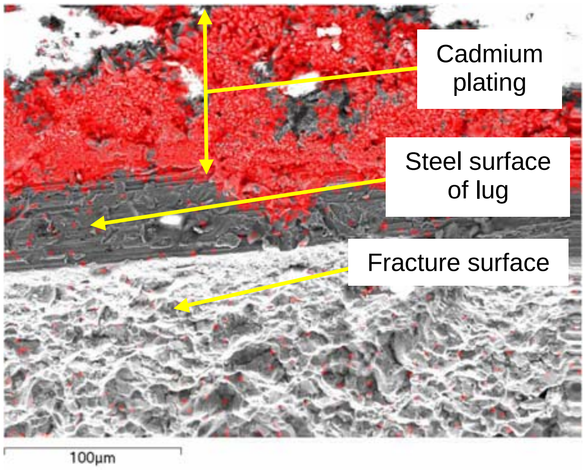
## Piston pre-existing crack




Initial crack
**Figure 6**




Lug radius showing step and poor machining
**Figure 7**




Surface at fracture origin showing cadmium distribution (red)
**Figure 8**