Lockheed L188C Electra, G-LOFD, 21 March 2001 at 1948 hrs

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Aircraft Type and Registration:	Lockheed L188C Electra, G-LOFD
No & Type of Engines:	4 Allison 501-D13 turboprop engines
Year of Manufacture:	1961
Date & Time (UTC):	21 March 2001 at 1948 hrs
Location:	Prestwick Airport, Scotland
Type of Flight:	Training
Persons on Board:	Crew - 2 - Passengers - None
Injuries:	Crew - None - Passengers - N/A
Nature of Damage:	Damage to left main landing gear, left propellers, left flap and tailplane
Commander's Licence:	Airline Transport Pilot's Licence
Commander's Age:	39 years
Commander's Flying Experience:	6,130 hrs (of which 560 were on type)
	Last 90 days - 57 hours
	Last 28 days - 42 hours
Information Source:	AAIB Field Investigation

History of the flight

The first officer was undergoing night flying exercises for his conversion training to the Electra aircraft and was in the right seat with the commander, as training captain, in the left seat. He had flown two normal night visual circuits, followed by an asymmetric power circuit and a go-around for a further circuit and landing on Runway 13 with No 1 engine retarded. The aircraft weight was about 68,600 lb and V_{AT} (threshold speed) was 'bugged' at 120 kt on the air speed indicator; the final approach was flown at V_{AT} + 20 kt.

The approach was normal and the aircraft was configured with the landing gear down and full flap. On short finals, the controller gave a surface wind check of 090°/12 kt. The aircraft crossed the threshold at V_{AT} and initiated the flare. However, the commander then considered that the first officer had held the aircraft in the flare for too long an interval and, in the light crosswind from the left, had allowed it to drift to the right of the centreline. The commander therefore took control and applied left rudder, left aileron and slight back pressure on the control yoke to regain the centreline and land. During the touchdown the left mainwheel touched first, followed by the right mainwheel and then the nosewheel. The touchdown appeared normal, but the commander was then immediately aware of a directional control problem. The aircraft was veering to the left to such an extent that he had to use full right rudder, in addition to asymmetric reverse thrust, to maintain the centreline. However, directional control became progressively more difficult as the aircraft decelerated. Because the Electra has no anti-skid facility, the wheel brakes are not normally used until the speed falls below 60 kt; in the event the commander thought that the speed had decreased to about 50 kt when he first applied the brakes. It then became apparent that the aircraft might leave the paved runway and so the commander applied full braking and maximum reverse thrust on all four engines to stop the aircraft on the left side of the paved surface. The aircraft came to rest about 10 feet from the edge of the runway and heading some 70° left of the runway direction.

The first officer pulled the four 'E Handles' to shut down all four engines and isolate their associated systems. Both pilots were not injured and vacated the aircraft through the main door.

The Airport Fire Service (AFS) attended the aircraft with four vehicles. Although there was no sign of fire the AFS deployed hoses, sidelines and dry powder media as a precaution.

Inspection of the aircraft on the runway

Runway 13 is 2,987 metres long and 46 metres wide, and has a displaced threshold with an elevation of 38 feet amsl. The concrete/asphalt surface of the runway is bordered by a hard surface for most of its length, and the aircraft had come to rest to the left of the runway edge on this border, some 1100 metres from the threshold. Initial examination of the aircraft showed that the lower (piston) oleo cylinder of the left main landing gear leg had fractured just above the axle, allowing separation of the axle and the two wheel/brake assemblies as a unit, as shown in Figure 1. The latter wheels assembly had caused relatively minor damage to the left flap and left tailplane after it had separated, and flash fires had occurred on the runway as hydraulic and oleo oil had been released from the broken gear leg and ignited by the friction sparks caused by contact of the fractured leg with the runway. The upper section of the piston cylinder had remained in the upper leg as the aircraft had decelerated and its fractured end had initially generated an intermittent groove in the runway surface which had progressively become more continuous as the aircraft had slowed. Just before the end of the ground roll, the stub of the left gear piston cylinder had struck and destroyed a runway edge light before crossing over a recessed drain aperture. As this occurred, the left wing had momentarily lowered further and caused all eight propeller blade tips on the left side of the aircraft to contact the runway surface.

Fractured piston cylinder examination

As described above, the upper fractured surface of the failed piston cylinder had been ground down by contact with the runway and so only the lower half of the fractured surfaces was available for examination. After recovery of the piston cylinder lower end, axle and wheels assembly to the AAIB facility at Farnborough, the unit was stripped down in preparation for detailed metallurgical examination.

The fractured surface was cut from the bottom section of the piston cylinder and examined visually. It was readily apparent that the fracture had initiated within the lowest area of the piston outer surface, close to the limit of normal travel of the piston (which was clearly indicated by the lower boundary of the polished area of the chromium plated surface). This location was on the inboard side of the piston and was characterised by corroded arcs on the fractured surface which extended some 2.5 mm into the material from the outer surface, as shown in Figure 2. Visual examination also revealed the presence of some parallel secondary cracking adjacent to the main fracture.

Examination of the fracture initiation zone in a Scanning Electron Microscope (SEM) revealed that the chromium plating on the surface exhibited brittle cleavage cracking and that the underlying steel in the corroded arc regions showed intergranular cracking, as illustrated in Figure 3. Beyond and between these areas the steel section had failed in overload.

Various non-destructive inspection techniques, including dye penetrant and magnetic particle tests, were used to examine the material around the fracture surface for evidence of additional cracking. Several areas of cracking of the steel substrate were found, and within the chromed surface on the front side of the leg, from 25 mm above the normal limit of travel to below this limit, a pattern of fine vertically orientated cracks were observed over the least worn regions. Almost all of the cracking in the steel was directly associated with cracks in the chromium plating, which was generally about 100µm thick in this area. Remote from the fracture initiation zone, the chromium thickness was reduced to between 25 and 40µm, probably due to wear. Some cracking in the plating was apparent in both the vertical and horizontal orientations, but this was not generally fully penetrating. Evidence of some remaining grit blasting debris was present in the chrome/steel interface, but the plating was strongly adherent to the parent steel substrate.

Chemical analysis of the piston material showed it to conform to the required specification of AISI 4340 steel, and hardness testing confirmed that it was of the required strength.

Discussion of the fracture mechanism

Hydrogen embrittlement and stress corrosion cracking (SCC) are two main causes of intergranular cracking in high strength steels. Cracking due to hydrogen embrittlement (caused by hydrogen diffusion into steel during plating processes) can result in a fracture surface very similar in appearance to that resulting from stress corrosion, but in this case SCC was considered the most likely cause. This was because most of the cracks identified were present in the regions of the piston routinely exposed to the general environment, rather than throughout the material examined which would have been more indicative of hydrogen embrittlement involvement.

SCC failures usually result from a field of cracks produced in a metal alloy under the combined influence of tensile stress (residual and/or applied) and a corrosive environment. The alloy is not attacked over most of its surface, but a system of intergranular cracks initiates at the stress corrosion site and propagates through the material grain structure over a period of time. Stress levels that produce SCC are well below the yield strength of the material and this mechanism is influenced by the level of stress, alloy composition, the type of corrosive environment and temperature.

The multiple orientation of the cracks evident on the failed piston indicated that internal stresses within the steel material were likely to be involved, as opposed to tensile stresses resulting from purely externally applied loads. The latter would result primarily from bending loads on the gear leg and it appeared that this mode of loading had propagated the final overstressing failure of the piston cylinder, after it had been weakened by the SCC propagation from the outer (cracked) chrome surface.

The distribution of the cracking detected in the chromium plating by NDT was consistent with excessive loads having been generated in the plating by grinding wheel operations during manufacture/refurbishment of this component. Whilst it is not unusual for some cracks to form in chromium 'as-plated', the many cracks observed were considered to have resulted from a combination of the plating process used and excessive grinding action. These cracks in the chrome surface had then allowed the underlying high strength steel to be exposed to corrosive conditions, inducing SCC.

Landing gear history

Records held by the aircraft manufacturer indicated that four previous gear leg piston cylinder failures had occurred during service operation of the Electra type which were similar to this failure on G-LOFD. Metallurgical examinations carried out on three of these failures had found that two could have been initiated by stress corrosion and/or hydrogen embrittlement cracks. The other failure was determined to have resulted from SCC. Improper thickness of the external chrome plating, and overhaul procedures which induced hydrogen embrittlement, were considered contributing factors.

The manufacturer stated that the Electra main landing gear was not fatigue-critical and that at the time that this aircraft type had initially been certificated there were no requirements to establish safe lives for the landing gear components. The high strength steel material was, however, acknowledged to be susceptible to corrosion and subsequent SCC, but proper maintenance and correct plating procedures during overhaul were deemed sufficient to inhibit such cracking. In the latter context, the manufacturer had issued Electra Service Information Letter (SIL) 88/SIL-88A in October 1974 to address related stripping and plating procedures, and believed that all operators were in possession of this SIL.

The authorised overhaul life for the main landing gear on G-LOFD was 16,000 flight hours. This component had achieved 15,400 hours since its last overhaul. The aircraft had flown for a total of 49,500 hours, with some 22,300 landings. It had been on the UK register since 1997, prior to which it had been operating with a Norwegian company. No re-working of this landing gear has been carried out since the aircraft had been on the UK register.