Australian Government
Australian Transport Safety Bureau

In-flight uncontained engine failure Airbus A380-842, VH-OQA

overhead Batam Island, Indonesia | 4 November 2010

Investigation

ATSB Transport Safety Report
Aviation Occurrence Investigation
AO-2010-089
Final – 27 June 2013

Australian Government

Australian Transport Safety Bureau

ATSB TRANSPORT SAFETY REPORT

Aviation Occurrence Investigation – AO-2010-089

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In-flight uncontained engine failure overhead Batam Island, Indonesia 4 November 2010 VH-OQA Airbus A380-842

Released in accordance with section 25 of the *Transport Safety Investigation Act 2003*

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Addendum

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SAFETY SUMMARY

What happened

On 4 November 2010, while climbing through 7,000 ft after departing from Changi Airport, Singapore, the Airbus A380 registered VH-OQA, sustained an uncontained engine rotor failure (UERF) of the No. 2 engine, a Rolls-Royce Trent 900. Debris from the UERF impacted the aircraft, resulting in significant structural and systems damage.

The flight crew managed the situation and, after completing the required actions for the multitude of system failures, safely returned to and landed at Changi Airport.

What the ATSB found

The Australian Transport Safety Bureau (ATSB) found that a number of oil feed stub pipes within the High Pressure / Intermediate pressure (HP/IP) hub assembly were manufactured with thin wall sections that did not conform to the design specifications. These non-conforming pipes were fitted to Trent 900 engines, including the No. 2 engine on VH-OQA. The thin wall section significantly reduced the life of the oil feed stub pipe on the No. 2 engine so that a fatigue crack developed, ultimately releasing oil during the flight that resulted in an internal oil fire. That fire led to the separation of the intermediate pressure turbine disc from the drive shaft. The disc accelerated and burst with sufficient force that the engine structure could not contain it, releasing high-energy debris.

What has been done to fix it

Following the UERF, the ATSB, Rolls-Royce plc, regulatory authorities and operators of A380 aircraft with Trent 900 engines took a range of steps to ensure that HP/IP hub assemblies with non-conforming oil feed stub pipes were identified and either removed from service, or managed to ensure their safe continued operation. Rolls-Royce also released an engine control software update that included an IP turbine overspeed protection system (IPTOS) that is designed to shut the engine down before the turbine disc can overspeed, in the unlikely event that a similar failure occurs.

Rolls-Royce has also made a range of changes to their quality management system to improve the way in which they manage non-conforming parts, both during the manufacturing process and when it has been identified that parts had unknowingly been released into service with non-conformances.

Safety message

The ATSB identified a number of issues during the manufacture of Trent 900 HP/IP hub assemblies that resulted in their release into service with non-conforming oil feed stub pipes. Those issues highlighted the importance of providing clear procedures during the manufacturing process and of personnel complying with those procedures. Even though modern civil turbine engines are very reliable, and UERFs are very rare events, the resulting damage from such a failure can be significant and the potential effects catastrophic. This accident represents an opportunity for the regulatory authorities to incorporate any lessons learned into their certification advisory material to enhance the safety of future aircraft designs.

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Figure D15 Courtesy of Coordinating Research Council report No. 635, Handbook of Aviation Fuel Properties, 2004 Third Edition.
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THE AUSTRALIAN TRANSPORT SAFETY BUREAU

The Australian Transport Safety Bureau (ATSB) is an independent Commonwealth Government statutory agency. The Bureau is governed by a Commission and is entirely separate from transport regulators, policy makers and service providers. The ATSB's function is to improve safety and public confidence in the aviation, marine and rail modes of transport through excellence in: independent investigation of transport accidents and other safety occurrences; safety data recording, analysis and research; fostering safety awareness, knowledge and action.

The ATSB is responsible for investigating accidents and other transport safety matters involving civil aviation, marine and rail operations in Australia that fall within Commonwealth jurisdiction, as well as participating in overseas investigations involving Australian registered aircraft and ships. A primary concern is the safety of commercial transport, with particular regard to fare-paying passenger operations.

The ATSB performs its functions in accordance with the provisions of the *Transport Safety Investigation Act 2003* and Regulations and, where applicable, relevant international agreements.

Purpose of safety investigations

The object of a safety investigation is to identify and reduce safety-related risk. ATSB investigations determine and communicate the safety factors related to the transport safety matter being investigated. The terms the ATSB uses to refer to key safety and risk concepts are set out in the next section: Terminology Used in this Report.

It is not a function of the ATSB to apportion blame or determine liability. At the same time, an investigation report must include factual material of sufficient weight to support the analysis and findings. At all times the ATSB endeavours to balance the use of material that could imply adverse comment with the need to properly explain what happened, and why, in a fair and unbiased manner.

Developing safety action

Central to the ATSB's investigation of transport safety matters is the early identification of safety issues in the transport environment. The ATSB prefers to encourage the relevant organisation(s) to initiate proactive safety action that addresses safety issues. Nevertheless, the ATSB may use its power to make a formal safety recommendation either during or at the end of an investigation, depending on the level of risk associated with a safety issue and the extent of corrective action undertaken by the relevant organisation.

When safety recommendations are issued, they focus on clearly describing the safety issue of concern, rather than providing instructions or opinions on a preferred method of corrective action. As with equivalent overseas organisations, the ATSB has no power to enforce the implementation of its recommendations. It is a matter for the body to which an ATSB recommendation is directed to assess the costs and benefits of any particular means of addressing a safety issue.

When the ATSB issues a safety recommendation to a person, organisation or agency, they must provide a written response within 90 days. That response must indicate whether they accept the recommendation, any reasons for not accepting part or all of the recommendation, and details of any proposed safety action to give effect to the recommendation.

The ATSB can also issue safety advisory notices suggesting that an organisation or an industry sector consider a safety issue and take action where it believes appropriate, or to raise general awareness of important safety information in the industry. There is no requirement for a formal response to an advisory notice, although the ATSB will publish any response it receives.

Report structure

Previous ATSB aviation occurrence investigation reports have closely followed the suggested report structure in the appendix of International Civil Aviation Organisation Annex 13. That suggested structure splits the report into four sections: factual information, analysis, conclusions and safety recommendations.

Annex 13 also states that the Final Report ‘may be prepared in the format considered to be the most appropriate in the circumstances’ (chapter 6). In this context, the factual information is divided into four main parts:

- Part 1—Occurrence flight;
- Part 2—Engine;
- Part 3—Engine failure sequence;
- Part 4—Manufacturing.

The ATSB analysis of this information is included throughout the report, either as ‘observations’ in grey filled observation boxes where appropriate, or as part of the analysis in Part 5 of the report. ‘Observations’ are also used to explain pertinent information to assist the reader in their understanding of the factual information presented.

TERMINOLOGY USED IN THIS REPORT

Occurrence: accident or incident.

Safety factor: an event or condition that increases safety risk. In other words, it is something that, if it occurred in the future, would increase the likelihood of an occurrence, and/or the severity of the adverse consequences associated with an occurrence. Safety factors include the occurrence events (e.g. engine failure, signal passed at danger, grounding), individual actions (e.g. errors and violations), local conditions, current risk controls and organisational influences.

Contributing safety factor: a safety factor that, had it not occurred or existed at the time of an occurrence, then either: (a) the occurrence would probably not have occurred; or (b) the adverse consequences associated with the occurrence would probably not have occurred or have been as serious, or (c) another contributing safety factor would probably not have occurred or existed.

Other safety factor: a safety factor identified during an occurrence investigation which did not meet the definition of contributing safety factor but was still considered to be important to communicate in an investigation report in the interests of improved transport safety.

Other key finding: any finding, other than that associated with safety factors, considered important to include in an investigation report. Such findings may resolve ambiguity or controversy, describe possible scenarios or safety factors when firm safety factor findings were not able to be made, or note events or conditions which 'saved the day' or played an important role in reducing the risk associated with an occurrence.

Safety issue: a safety factor that (a) can reasonably be regarded as having the potential to adversely affect the safety of future operations, and (b) is a characteristic of an organisation or a system, rather than a characteristic of a specific individual, or characteristic of an operational environment at a specific point in time.

Safety action: the steps taken or proposed to be taken by a person, organisation or agency in response to a safety issue.

EXECUTIVE SUMMARY

Key investigation outcomes

On 4 November 2010, an uncontained failure of the intermediate pressure turbine disc occurred in the No. 2 engine, of an Airbus A380 aircraft, registered VH-OQA, overhead Batam Island, Indonesia. The failure was the result of an internal oil fire within the Rolls-Royce Trent 900 engine that led to the separation of the intermediate pressure turbine disc from its shaft. The fire started when oil was released from a crack in the pipe that supplied oil to the high pressure/intermediate pressure (HP/IP) turbine bearing chamber. The Australian Transport Safety Bureau (ATSB) found that the oil pipe cracked because it had a thin wall from a misaligned counter bore that did not conform to the design specification.

Immediately after the accident, and then progressively during the preliminary stages of the investigation, the ATSB interacted with a number of organisations who advised of various proactive safety actions taken to prevent a recurrence. The ATSB monitored those proactive safety actions to ensure that the safety issues indicated by the accident were being addressed. In particular, the ATSB identified a safety issue that affected all Trent 900 series engines in the world-wide fleet. On 1 December 2010, the ATSB issued a safety recommendation to Rolls-Royce plc to address that issue and published a preliminary investigation report that set out the context for that recommendation.

As a result of its continuing investigation, the ATSB identified a number of factors that resulted in the manufacture and release into service of non-conforming oil feed pipes. Those factors occurred over a number of years, and highlighted the importance of manufacturers providing clear procedures and of personnel complying with those procedures. Throughout the investigation, the ATSB continued to work with the aircraft and engine manufacturers to ensure that any identified safety issues were addressed and actions taken to prevent a similar occurrence.

The ATSB also identified that this accident represented an opportunity to extend the knowledge base relating to the hazards from uncontained engine rotor failure (UERF) events and to incorporate that knowledge into airframe certification advisory material in order to further minimise the effects of a UERF on future aircraft designs. As a result, the ATSB issued recommendations to the European Aviation Safety Agency and the United States Federal Aviation Administration, recommending that both organisations review the damage sustained by the aircraft in order to incorporate any lessons learned from this accident into the certification compliance advisory material.

The occurrence flight

The aircraft departed Changi Airport, Singapore on a scheduled passenger flight to Sydney, Australia. About 4 minutes after take-off, while the aircraft was climbing through about 7,000 ft, the flight crew heard two ‘bangs’ and a number of warnings and cautions were displayed on the electronic centralised aircraft monitor (ECAM).

Initially, the ECAM displayed a message warning of turbine overheat in the No. 2 (inner left) engine. That warning was followed soon after by a multitude of other messages relating to a number of aircraft system problems. After assessing the situation and completing a number of initial response actions, the flight crew were cleared by ATC to conduct a holding pattern to the east of Changi Airport. While in the holding pattern, the flight crew worked through the procedures relevant to the messages displayed by the ECAM. During that time the flight crew were assisted by additional crew that were on the flight deck as part of a check and training exercise.

The aircraft sustained significant impact damage to the left wing by fragments and debris from the UERF, and fuel was leaking from the damaged left wing fuel tanks. However, after completing the ECAM procedures and performing some aircraft controllability checks, the flight crew landed the aircraft safely at Changi Airport.

After landing, fuel continued to leak from the left wing tank. The risk associated with this leak was minimised by the airport emergency services by applying large quantities of water and foam below the left wing while the aircraft's engines were shut down.

The No. 1 (outer left) engine continued to run following the normal shut-down procedure. Because of the still-running engine and leaking fuel on the left side, the passengers were disembarked via a set of stairs on the right side of the aircraft. The disembarkation was completed about 2 hours after the aircraft landed. Numerous unsuccessful attempts to shut down the No. 1 engine were made by the flight crew, maintenance engineers and the airport emergency services using different methods . The engine was finally shut down about 3 hours after the aircraft landed by pumping firefighting foam directly into the engine inlet.

The ATSB found that the flight crew and cabin crew managed the event as a competent team in accordance with standard operating procedures and practices.

Damage to the aircraft

The aircraft sustained damage from a large number of disc fragments and associated debris. The damage affected the aircraft's structure and a number of its systems.

The ATSB found that a large fragment of the turbine disc penetrated the left wing leading edge before passing through the front spar into the left inner fuel tank and exiting through the top skin of the wing. The fragment initiated a short duration low intensity flash fire inside the wing fuel tank. The ATSB determined that the conditions within the tank were not suitable to sustain the fire.

Another fire was found to have occurred within the lower cowl of the No. 2 engine as a result of oil leaking into the cowl from the damaged oil supply pipe. The fire lasted for a short time and self-extinguished.

The large fragment of the turbine disc also severed wiring looms inside the wing leading edge that connected to a number of systems.

A separate disc fragment severed a wiring loom located between the lower centre fuselage and body fairing. That loom included wires that provided redundancy (back –up) for some of the systems already affected by the severing of wires in the wing leading edge. This additional damage rendered some of those systems inoperative.

The aircraft's hydraulic and electrical distribution systems were also damaged, which affected other systems not directly impacted by the engine failure.

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Engine failure sequence

The crack in the oil feed pipe was a result of fatigue and had developed over some time. On the occurrence flight, the crack grew to a size that allowed oil in the pipe to be released into the buffer space between the bearing chamber and the hot air surrounding the IP turbine disc. The oil was released as an atomised spray and the air within the buffer space was sufficiently hot for the oil to auto-ignite.

The resulting fire propagated through the bearing chamber buffer space and eventually impinged upon the IP turbine disc drive arm, resulting in the separation of the disc from the drive shaft. Following the separation of the disc, the engine behaved in a manner different to that anticipated by the manufacturer during engine design and testing. The disc accelerated to a speed in excess of its structural capacity and burst into three main segments, with sufficient force to puncture the engine case at high velocity.

The ATSB found that the crack formed in a section of the HP/IP bearing chamber oil feed pipe, known as the oil feed stub pipe. That section of pipe contained an area of reduced wall thickness, which resulted from the misalignment of a counter bore machined into the end of the stub pipe during manufacture. A detailed engineering analysis found that the stresses generated in the oil feed stub pipe were sensitive to the wall thickness, which in turn had a significant effect on the pipe's fatigue life. The ATSB found that the alignment of the counter bore did not conform to the design specification.

Following the event, the engine manufacturer determined that a significant number of HP/IP bearing support assemblies were produced with oil feed stub pipes that did not conform to the design specification. Those assemblies had been released into service and were fitted to a number of engines in the Trent 900 fleet. As a result, on 1 December 2010 the ATSB issued a safety recommendation for Rolls-Royce plc to address this issue. Rolls-Royce took action to fully address the recommendation.

Manufacture of non-conforming oil feed stub pipes

The ATSB found that the misalignment of the counter bores was the result of movement within the HP/IP bearing support assembly during manufacture and that a number of opportunities existed during the design and manufacture processes where the misaligned oil feed stub pipe counter bores could have been identified and managed. Those opportunities were missed for a number of reasons, but generally because of ambiguities within the manufacturer's procedures and the non-adherence by a number of the manufacturing staff to those procedures.

During the development of the Trent 900 engine, a manufacturing datum (or reference) was introduced to specify the location of the oil feed stub pipe counter bore. That manufacturing datum was intended to replace the datum from the design specification, as the design datum became inaccessible when the oil feed stub pipe was fitted to the HP/IP bearing support assembly. The manner in which the manufacturing datum was represented on the manufacturing drawings resulted in its position not being constrained to the location of the oil feed stub pipe. This had several flow-on effects that made it more difficult to detect the misaligned counter bores and the effect that it had on the wall thickness of the oil feed stub pipe.

During an inspection of the first HP/IP bearing support assembly manufactured in 2005, the manufacturing drawings were referenced rather than the design definition drawings. The manufacturer's procedures for the inspection contained ambiguity that may have influenced the inspector’s decision to use the manufacturing drawings. As a result, the lack of constraint on the manufacturing datum was not identified and the HP/IP bearing support assembly entered into production without having properly shown compliance with the design specification.

The HP/IP bearing support assembly was released into service without the non-conforming oil feed stub pipe being reported in accordance with the manufacturer’s non-conformance management procedures. Due to an absence of some of the inspection records for the HP/IP bearing support assembly in the No. 2 engine from VH-OQA, the ATSB could not determine exactly why the non-conformance was not reported.

Similarly, records for HP/IP bearing support assemblies from other Trent 900 engines produced around that time had not been retained by the manufacturer. The ATSB therefore could not determine how they were released without the non-conformances being reported. However, the ATSB identified that a culture existed within the manufacturer’s facility that produced the HP/IP bearing support assemblies where it was considered acceptable to not report what were considered to be ‘minor’ non-conformances.

In 2007, the manufacturer identified that a number of components had left the facility with unreported non-conformances and carried out a major quality investigation. After that investigation, a number of newly manufactured non-conforming oil feed stub pipe counter bores were identified and reported by manufacturing personnel. However, due to a difference between the reference datum used by the manufacturer’s automated measuring machines and the datum specified on the drawings, the engineers assessing the effect of the non-conformance misunderstood how the non-conformance would affect the wall thickness of the oil feed stub pipe.

In March 2009, a manufacturing engineer identified that oil feed stub pipe counter bores were misaligned in previously manufactured and released HP/IP bearing support assemblies. The engineer was the first to identify the effect that misalignment of the counter bore had on the wall thickness of the pipe.

In an attempt to estimate the potential size of the misalignment within all of the previously-manufactured HP/IP bearing support assemblies, the engineer measured all nine of the HP/IP bearing support assemblies available at the facility at that time and performed a statistical analysis on those measurements. Based on that analysis, the engineer applied for an engineering assessment to establish the suitability of those non-conforming oil feed stub pipes for continued use. This application was submitted through the manufacturer’s non-conformance management system. There was a level of uncertainty inherent within the statistical analysis method used in that assessment. The uncertainty was not effectively communicated to, or understood by, the engineers that assessed the application and ‘retrospective’ approval was given to permit those non-conforming HP/IP bearing support assemblies to remain in service.

The manufacturer’s procedure for such retrospective approvals (known as retrospective concessions) required that the application be approved by both the Chief Engineer and the Business Quality Director. For a reason that could not be positively identified by the ATSB, neither the Chief Engineer’s nor the Business Quality Director’s approval for the oil feed stub pipe counter bore retrospective concession were sought. This omission denied the Chief Engineer the opportunity to assess what risk the non-conformance may have had upon the fleet of in-service Trent 900 engines.

The ATSB found no other opportunities where the potential for cracking in the oil feed stub pipes could have been identified and the non-conformances managed prior to the UERF on VH-OQA.

Minimisation of hazards resulting from an uncontained engine rotor failure

The United States Federal Aviation Administration (FAA) and the European Aviation Safety Agency (EASA) had promulgated advisory material for the minimisation of hazards resulting from uncontained engine rotor failures (UERF). That material was designed to assist manufacturers in meeting the requirement of the relevant certification standard. The advisory material was based on service experience, including accident investigation findings. The damage to VH-OQA exceeded the parameters of the model provided in the advisory material for predicting the likely damage from a UERF. Information from the accident represents an opportunity to incorporate any lessons learned from this accident in the advisory material. The ATSB therefore issued recommendations to EASA and FAA to address this opportunity.

Investigation process

The ATSB's investigation covered a range of complex issues to determine the factors involved in the occurrence. This could not have been achieved without the expertise and cooperation of:

- Air Accident Investigation Bureau of Singapore
- Airbus SAS
- Civil Aviation Safety Authority of Australia
- European Aviation Safety Agency
- French Bureau d'Enquêtes et d'Analyses pour la sécurité de l'aviation civile
- Indonesian National Transportation Safety Committee
- Qantas Airways Ltd
- Rolls-Royce plc
- United Kingdom Air Accident Investigation Branch
- United Kingdom Civil Aviation Authority.

The ATSB acknowledges and is grateful for the cooperation received from those organisations.

ABBREVIATIONS

AC	Alternating Current
ACARS	Airborne Communication Addressing and Reporting System
ACJ	Advisory Circular Joint
ACMS	Aircraft condition monitoring system
AES	Airport emergency services
AIA	Aerospace Industries Association
AMSL	Above mean sea level
AAIB	Air Accidents Investigation Branch (United Kingdom)
AAIB of Singapore	Air Accident Investigation Bureau of Singapore
ANSU	Aircraft network server unit
APU	Auxiliary power unit
ARAC	Aviation Rule making Advisory Committee
ATC	Air traffic control
ATPL	Airline Transport Pilot Licence
ATSB	Australian Transport Safety Bureau
BEA	Bureau d’Enquêtes et d’Analyses pour la sécurité de l’aviation civile
Cat	Category
CMM	Coordinate measuring machine
CMS	Centralised maintenance system
CNC	Computer numerical control
CRM	Crew resource management
CSM	Customer service manager
CVR	Cockpit voice recorder
DAR	Definition alteration request
DE	Design engineer
DGAC	Direction générale de l’aviation civile
DOA	Design Organisation Approval
EASA	European Aviation Safety Agency
EBHA	Electrical backup hydraulic actuator
ECAM	Electronic centralised aircraft monitoring
EEC	Engine electronic controller
EHA	Electro-hydrostatic actuator
EMU	Engine monitoring unit
ENG	Engine
ESN	Engine serial number

FAA	Federal Aviation Administration (United States)
FADEC	Full authority digital engine control
FAI	First article inspection
FAIR	First article inspection report
FCEPM	Flight Crew Emergency Procedures Manual
FCOM	Flight Crew Operating Manual
FCTM	Flight Crew Training Manual
FDR	Flight data recorder
FDU	Fire detection unit
FMECA	Failure modes effects and criticality analysis
FMS	Flight management system
GQP	Group quality procedure
HCAS	Hucknall Casings and Structures
HP	High pressure
HP3	High pressure stage 3
Hz	Hertz
IAQG	International Aerospace Quality Group
ICAO	International Civil Aviation Organization
IP	Intermediate pressure
IP8	Intermediate pressure stage 8
IPT	Integrated Programme Team
IPTOS	Intermediate pressure turbine overspeed protection system
ISO	International Organisation for Standardisation
JAR	Joint Aviation Requirements
JAR-E	Joint Aviation Requirements – Engines
kN	Kilonewton
kPa	Kilopascal
LPA	Landing performance application
LP	Low pressure
LPTOS	Low pressure turbine overspeed system
MIE	Minimum ignition energy
mJ	Millijoule
MPa	Megapascal
MQI	Major quality investigation
N1	Low pressure assembly shaft speed
N2	Intermediate pressure assembly shaft speed
N3	High pressure assembly shaft speed
NASA	National Aeronautics and Space Administration

NGV	Nozzle guide vane
NMSB	Non-modification service bulletin
No.	Number
NTSB	National Transportation Safety Board (United States)
NTSC	National Transportation Safety Committee (Indonesia)
OBU	Operating business unit
OIS	Onboard information system
OIT	Onboard information terminal
OP	Operation Number
P30	High Pressure Compressor Delivery Pressure
PA	Public address
PED	Portable electronic device
pph	Pounds per hour
POA	Production Organisation Approval
psi	Pounds per square inch
QMS	Quality management system
RA	Radio altitude
REP	Aircraft Systems Report
RPM	Revolutions per minute
SACMSR	Smart aircraft condition monitoring system recorder
SAR	Safety alert report
SCU	Supply Chain Unit
SFAR	Special Federal Aviation Regulation
TCAF	Turbine cooling air front
TCAR	Turbine cooling air rear
TS&D	Transmissions Structures and Drives
UK	United Kingdom
UK CAA	United Kingdom Civil Aviation Authority
US	United States
UTC	Coordinated Universal Time
VHF	Very high frequency
VQAR	Virtual quick access recorder
WDAR	Wireless digital aircraft condition monitoring system recorder

1 FACTUAL INFORMATION: OCCURRENCE FLIGHT

1.1 Sequence of events

1.1.1 In-flight engine failure

On 4 November 2010, at 01:56:47 Coordinated Universal Time (UTC),[1] an Airbus A380 aircraft (A380), registered VH-OQA and being operated as Qantas flight 32, departed from runway 20 centre (20C) at Changi Airport, Singapore, for Sydney, Australia. On board were five flight crew,[2] 24 cabin crew and 440 passengers.

Following a normal takeoff, the crew retracted the landing gear and flaps and changed the thrust setting to climb. At about 02:01, while maintaining 250 kt in the climb and passing 7,000 ft above mean sea level (AMSL), the crew heard two, almost coincident 'loud bangs'. The captain immediately selected altitude and heading hold mode on the auto flight system control panel.

The crew reported that there was a slight yaw and the aircraft levelled off in accordance with the selection of altitude hold. The captain stated that he expected the aircraft's autothrust system to reduce thrust on the engines to maintain 250 kt as the aircraft levelled off. However, the autothrust system was no longer active, so he manually retarded the thrust levers to control the aircraft's speed.

Initial ECAM actions

The electronic centralised aircraft monitoring (ECAM)[3] (Figure 1) displayed a message indicating a No. 2 engine turbine overheat warning. Soon after, the ECAM began displaying multiple messages (Appendix A). The captain confirmed with the other flight crew that he had control of the aircraft and instructed the first officer to commence the procedures as presented on the ECAM.

The procedure for the overheat message was to move the affected engine's thrust lever to the idle position and to monitor the situation for 30 seconds. During that 30 second monitoring period, at 02:02:00, the crew transmitted a PAN[4] to Singapore air traffic control (ATC). The first officer also reported observing an ECAM warning of a fire in the No. 2 engine that was displayed for about 1 to

1 Singapore local time was UTC + 8 hours.

2 The flight crew comprised the normal operating crew (captain and first officer) with the addition of a second officer for crew relief purposes. This flight also carried a check captain under training and a supervising check captain, who was supervising the check captain under training. All flight crew were located in the cockpit.

3 The ECAM provides information to the crew on the status of the aircraft and its systems. It also presents the required steps of applicable procedures when an abnormal condition has been detected by the monitoring system.

4 An internationally recognised radio call announcing an urgency condition which concerns the safety of an aircraft or its occupants but where the flight crew does not require immediate assistance.

2 seconds, before the ECAM reverted to the overheat warning and reinitiated the 30 second monitoring period. As part of the turbine overheat procedure, the crew elected to shut down the No. 2 engine. During the shutdown procedure, the ECAM displayed a message indicating that the No. 2 engine had failed.

A damage assessment as part of the engine failure procedure suggested that the damage to the No. 2 engine was serious and the flight crew discharged one of the engine's two fire extinguisher bottles. Contrary to their expectation, the flight crew did not receive confirmation that the fire extinguisher bottle had discharged. They repeated the procedure for discharging the fire extinguisher and again did not receive confirmation that it had discharged.

The flight crew recalled that, after a brief discussion, they followed the procedure for discharging the second fire extinguisher bottle into the No. 2 engine. After completing that procedure twice, they did not receive confirmation that the second bottle had discharged. They then elected to continue the engine failure procedure, which included initiating a process of fuel transfer.

The engine/warning display indicated that the No. 2 engine had changed to a failed mode, No. 1 and 4 engines had reverted to a degraded mode[5] and that No. 3 engine was operating in an alternate mode (Figure 1).

Figure 1: Engine/warning display

Image source: Image taken during the occurrence flight; supplied by a flight crew member.

Managing the situation

The flight crew discussed the available options to manage the situation, including an immediate return to Singapore, climbing or holding. As the aircraft remained controllable, and there was ample fuel on board, it was decided that the best option would be to hold at the present altitude while they processed the ECAM messages and associated procedures. The flight crew recalled frequently reviewing this decision and assessing the amount of fuel on board.

The flight crew contacted ATC and advised that they would need about 30 minutes to process the ECAM messages and associated procedures, and requested an

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5 Degraded or alternate engine mode indicates that some air data or engine parameters are not available.

appropriate holding position in order for that to occur. ATC initially cleared the flight crew to conduct a holding pattern to the east of Singapore. Following further discussion amongst the flight crew, ATC was advised that a holding area within 30 NM (56 km) of Changi Airport was required. ATC acknowledged that requirement and directed the aircraft to a different area to the east of the airport and provided heading information to maintain the aircraft in an approximately 20 NM (37 km) long racetrack holding pattern at 7,400 ft (Figure 2). ATC also advised of reports that a number of aircraft components had been found by residents of Batam Island, Indonesia.

Figure 2: Flight path during the event

Image source: Google Earth.

Observations of the aircraft's damage from the cabin

At the time of the engine failure, the seat belt sign was on and all cabin crew and passengers were seated. Cabin crew and passengers observed damage to the aircraft's left wing and fuel escaping from that wing. The customer service manager (CSM) and other cabin crew attempted to contact the captain at that time to report the observed damage, however, the flight crew did not respond.

While the flight crew continued to process the ECAM messages and associated procedures, the second officer went into the cabin to visually assess the damage. As the second officer moved through the cabin, a passenger, who was also a pilot for the operator, brought his attention to a view of the aircraft from the vertical fin-mounted camera that was displayed on the aircraft's in-flight entertainment system. That display showed a fuel leak from the left wing.

The second officer proceeded to the main (lower) deck and observed damage to the left wing and fuel leaking from the wing. He recalled that the leak appeared to be coming from underneath the wing in the vicinity of the No. 2 engine, and that the trail was about 0.5 m wide (Figure 3). He reported that he could not see the turbine area of the No. 2 engine from anywhere within the cabin. The second officer returned to the cockpit and reported his observations to the other members of the flight crew.

The captain and the supervising check captain made a number of public address (PA) announcements during the flight to inform the cabin crew and passengers of the situation and to provide updates when required.

Figure 3: Fuel leaking from the left wing

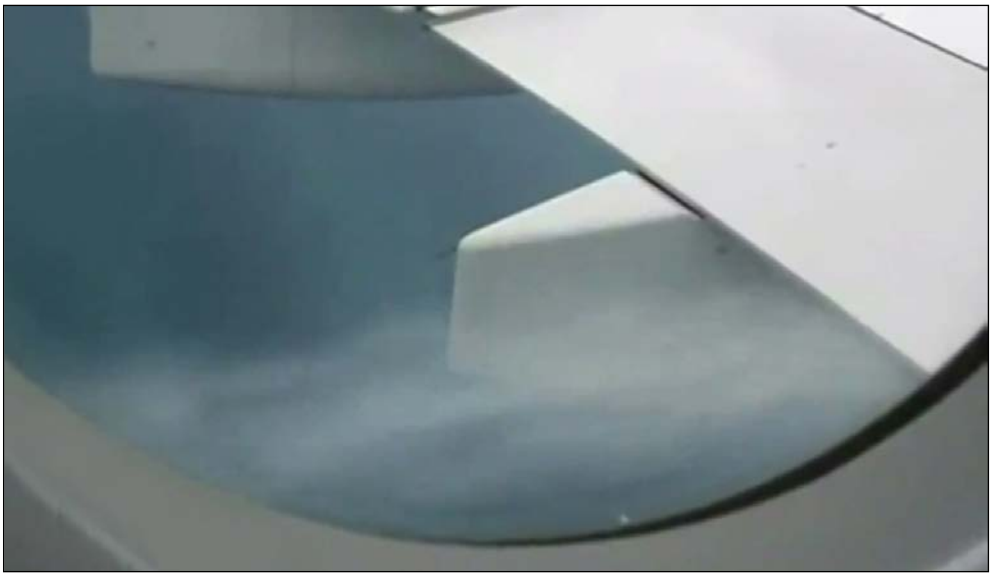

Image source: Supplied by a passenger.

Completion of ECAM procedures

The flight crew reported that, during their assessment of subsequent multiple fuel system ECAM messages, they elected not to initiate further fuel transfer as they were unsure of the integrity of the fuel system. In addition, the flight crew could not jettison fuel due to damage to the fuel management system.

The flight crew also received an aircraft communications addressing and reporting system (ACARS)[6] message from the aircraft operator that indicated that multiple failure messages had been received from the aircraft by the operator. At the time, the flight crew were busy managing the ECAM messages and procedures, and only had time to acknowledge but not respond to that ACARS message.

It took about 50 minutes for the flight crew to complete all of the initial procedures associated with the ECAM messages.

Preparations for landing

On completion of the ECAM procedures, the flight crew assessed the aircraft's systems to determine their functionality and the effect of any failures or inoperative

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6 A wireless communication system that was used to transmit and receive data and text messaged to and from the aircraft.

systems on further operations. This included attempting to calculate the landing distance using the landing performance application (LPA)[7] (section 1.14).

The inoperative wing leading edge lift devices, reduced braking function, reduced number of operational spoilers and inactive left engine thrust reverser[8] resulted in an abnormal landing configuration, which in turn affected the landing distance calculation. After some initial difficulty in calculating the landing distance required due to the high number of system and flight control malfunctions, the flight crew determined that a landing within the distance available on runway 20C at Changi Airport was achievable and proceeded on that basis.

The flight crew advised ATC that on landing they required emergency services, and that the aircraft was leaking fuel from the left wing. The captain called the CSM on the interphone to advise him of the potential for a runway overrun and evacuation, and the CSM and cabin crew prepared the cabin for this possibility.

Prior to leaving the holding pattern, the crew discussed the controllability of the aircraft and conducted a number of manual handling checks. The crew determined that the aircraft remained controllable, and advised ATC that they required a long final approach to runway 20C.

The approach and landing

The flight crew progressively configured the aircraft for the approach and landing and conducted further controllability checks in each new configuration. As a result of the damage to the aircraft, extending the landing gear required use of the emergency extension procedure.[9]

Singapore ATC radar vectored the aircraft to a position 20 NM (37 km) from the threshold of runway 20C and provided for a progressive descent to 4,000 ft. The captain set engines No. 1 and 4 to provide symmetrical thrust, and controlled the aircraft's speed with thrust from No. 3 engine.

Following the engine failure and subsequent system damage, the autopilot remained operational. The captain, as the handling pilot, continued to fly the aircraft using the autopilot during the time that the crew managed the ECAM procedures. The captain manually disconnected the autopilot to conduct control checks to assess the handling qualities of the aircraft, before re-engaging it. The autopilot disconnected twice during the approach—these were automatic disconnections in response to pre-set functions within the autopilot system relating to the aircraft's angle of attack. When the autopilot disconnected for the second time (at about 800 ft) the captain elected to leave it disconnected and manually fly the aircraft for the remainder of the approach.

The aircraft touched down at 03:46:47 and the captain applied the brakes and selected reverse thrust on the No. 3 engine. The flight crew observed that the deceleration appeared to be 'slow' in the initial landing roll, but that with the braking effort being used and reverse thrust, the aircraft began to slow. The captain recalled feeling confident that, as the speed approached 60 kt, the aircraft would be

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7 A computer application within the aircraft's onboard information system (OIS) used to calculate aircraft landing performance.

8 The A380 is equipped with thrust reversers on the inboard (No. 2 and 3) engines only.

9 An alternative method of extending the landing gear using gravity.

able to stop in the remaining runway. The No. 3 engine was gradually moved out of maximum reverse thrust and manual braking was continued until the aircraft came to a stop about 150 m from the end of the runway. The aircraft was attended by emergency services.

1.1.2 Events on the ground following the landing

On completion of the landing roll, the flight crew commenced shutting down the remaining engines and, when the final engine master switch was selected OFF, the aircraft's electrical system went into a configuration similar to the emergency electrical power mode. That rendered all but one of the aircraft's cockpit displays inoperative (blank), and meant that there was only one very high frequency (VHF) radio available to the crew.

It was reported that, just before the cockpit displays went blank, a number of the flight crew noticed that the left body landing gear brake temperature was indicating 900 °C, and rising. After some initial confusion about which radio was functioning, the first officer contacted the emergency services fire commander, who asked for the No. 1 engine to be shut down. The first officer responded that they had done so already, but was advised again by the fire commander that the engine continued to run.

The flight crew briefly discussed the still-running No. 1 engine and recycled the engine master switch to OFF, but the engine did not shut down. In response, the flight crew decided to press the engine fire push button and then fire extinguisher bottles in an attempt to shut down the engine. This was also ineffective and the engine continued to run. At this stage, the fire commander indicated that there appeared to be fuel leaking from the aircraft's left wing. The first officer advised the commander of the hot brakes and requested that fire retardant foam be applied over the leaking fuel. The firefighters had already commenced laying a foam blanket over the fuel leak in accordance with the airport emergency services standard operating procedures.

The flight crew then discussed the options for disembarking the passengers. The captain made a PA announcement to the cabin crew and passengers to advise them of the situation, and that the emergency services were dealing with a fluid leak from the left side of the aircraft. After accessing the necessary checklists relating to the evacuation of the passengers and crew, and with the knowledge that the fire risk was being managed and had decreased, the crew decided that a precautionary disembarkation via stairs on the right of the aircraft would be the safest course of action.

The flight crew elected to use a single door for the disembarkation so that the passengers could be accounted for as they left the aircraft and to keep the remainder of the right side of the aircraft clear in case of the need to deploy the escape slides. They also decided to leave the remaining doors armed, with cabin crew members at those doors ready to activate the respective escape slides until all of the passengers were off the aircraft. About 13 minutes after the aircraft landed, the flight crew asked the fire commander to have stairs brought to the right of the aircraft and to arrange for buses to move the passengers to the terminal. Consideration of how to shut down the No. 1 engine continued, with some flight crew members contacting the operator via mobile phone to seek further assistance.

Stairs arrived at the aircraft about 35 minutes after landing and the first bus arrived about 10 minutes later. Passengers commenced disembarking from the aircraft via the No. 2 main (lower) deck forward door about 50 minutes after the aircraft touched down. The last passengers disembarked the aircraft about 1 hour later.

Throughout the disembarkation, the flight crew, under advice from the operator's maintenance personnel, attempted to shut down the No. 1 engine through various alternative means. That included activating a series of circuit breakers in the aircraft's equipment bay and reconfiguring the transfer valves in the aircraft's external refuelling panel to transfer fuel away from the engine. All of these attempts were unsuccessful.

Maintenance personnel also attended the aircraft and attempted a number of methods to shut down the engine, each without success. Finally, the decision was taken by the operator to 'drown' the engine, initially with water and then with fire fighting foam from the airport emergency services fire vehicles (Figure 4). The No. 1 engine was reported to have been shut down at 06:53:00, about 3 hours after the aircraft landed.

Figure 4: Fire-fighters 'drowning' the No. 1 engine with foam

Image source: Supplied by the Air Accident Investigation Bureau (AAIB) of Singapore.

1.2 Injuries to persons

There were no reported injuries to the crew or passengers. There were no confirmed injuries to persons on Batam Island.

1.3 Damage to the aircraft

Initial inspection revealed that the No. 2 engine sustained an uncontained failure in the turbine region and that debris from the engine damaged the airframe and systems.

1.3.1 No. 2 engine damage

There was damage to the cowling and thrust reversers towards the rear of the engine and sooting on the rear section of the cowl (Figure 5). There was extensive damage to the turbine region of the engine, detail of which is covered in Part 2 of this report.

Figure 5: General damage to the No. 2 engine

1.3.2 Airframe damage

Liberated material from the uncontained engine failure resulted in significant damage to the airframe structure and systems. The structural damage included the:

- No. 2 engine support structure (Figure 6)
- left wing upper and lower skins, front spar and internal ribs (Figure 7 and Figure 8)
- left wing-to-fuselage joint (indentation and scoring).

Figure 6: Damage to the No. 2 engine support structure

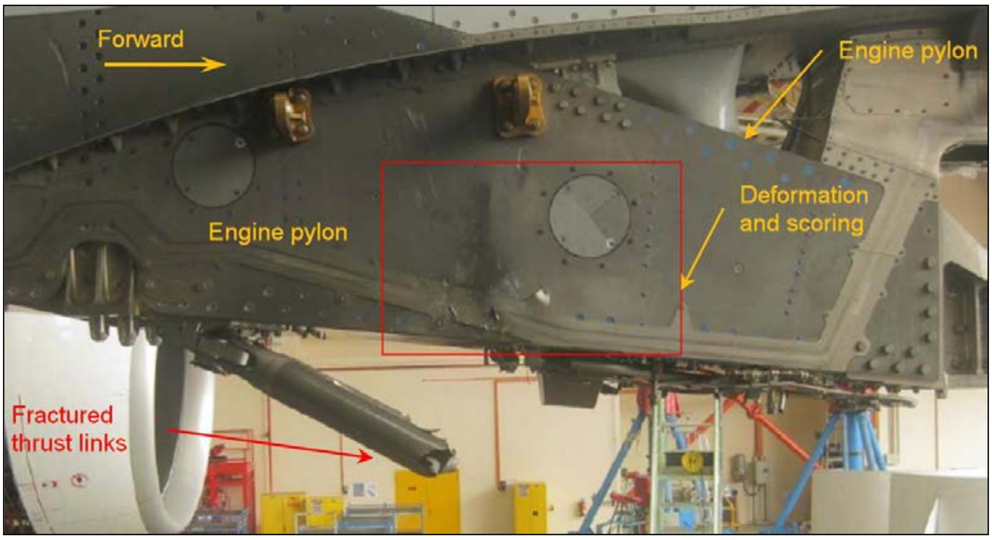


Figure 7: Examples of damage to the left wing

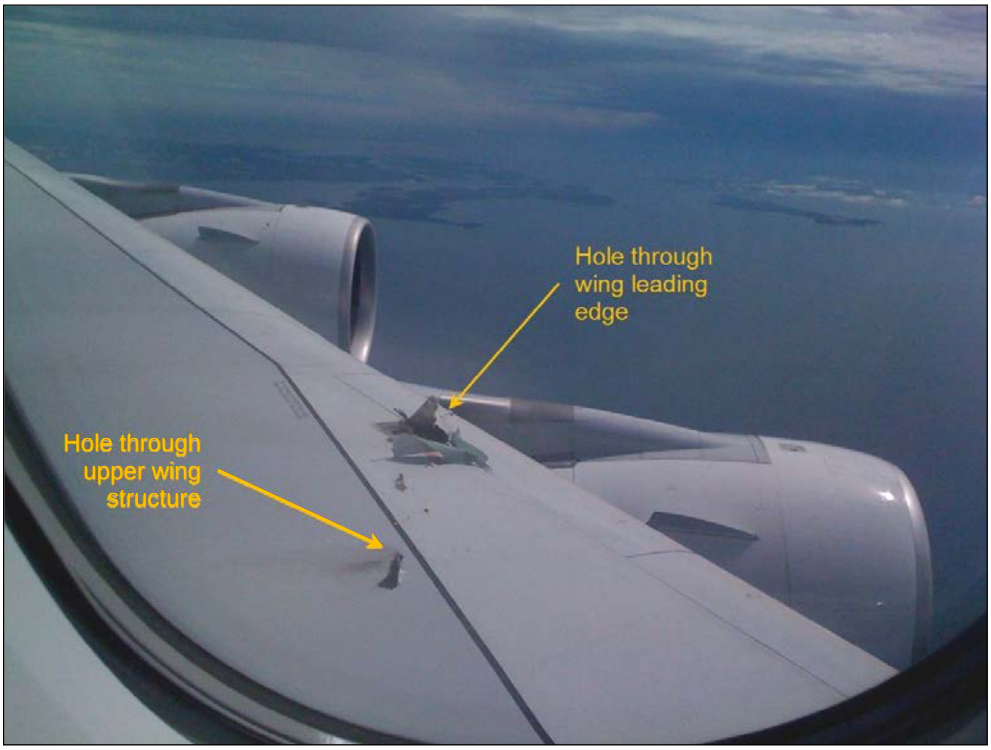
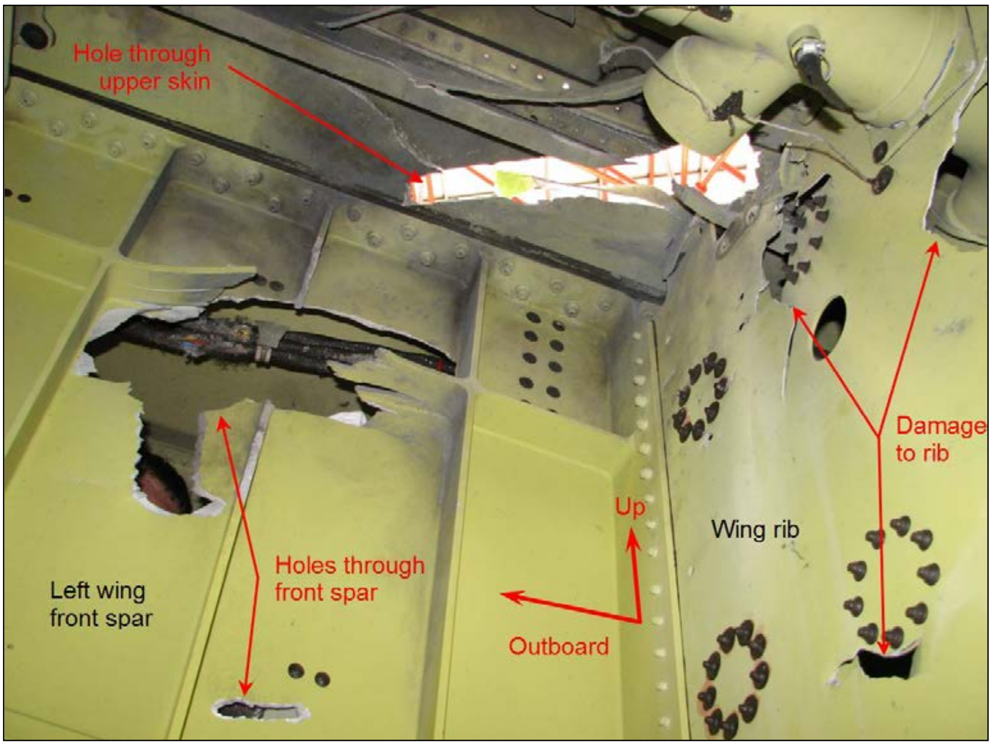


Image source: Supplied by a passenger.

Figure 8: Example of internal damage to the left wing (looking forward and up)



In addition to the damage to the primary structure, the left wing lower skin, fuselage skins and vertical and horizontal stabilisers contained numerous indentations and score marks from debris impacts. Further information on the airframe damage is contained in Appendix B.

System effects

The debris from the engine failure directly damaged a number of systems and a number of other systems were affected as a result. Although some systems sustained direct mechanical damage, most of the affected systems were damaged through debris impact to the respective wiring looms. Two main wiring looms were impacted by debris; one running through the leading edge of the left wing (Figure 9), and the other in the belly fairing (Figure 10). This amounted to damage to about 650 wires in total.

Figure 9: Example of wiring damage in the left wing (looking rearwards)

Figure 10: Damage to wiring in the belly fairing

The following systems were affected either as a direct result of the damage from the engine failure, or due to actions taken by the flight crew as part of the ECAM procedures:

- Hydraulic power, as a result of:
  - the loss of function to the aircraft's green hydraulic system
  - reduced redundancy within the aircraft's other (yellow) hydraulic system.
- Electrical power, resulting from:
  - a loss of electrical power generation at engines No. 1 and 2

- the loss of one of the aircraft's four alternating current (AC) systems
  - the inability to connect the aircraft's auxiliary power unit (APU) generators on the ground.
- Flight controls, through:
  - reduced aileron and spoiler function
  - the loss of wing leading edge slats and droop nose function.[10]
- Engine control, as a result of:
  - loss of autothrust function
  - reduction in the automatic control function to engines No. 1, 3 and 4.
- The landing gear normal extension function was no longer available.
- Braking, through a:
  - reduction of function in the right wing gear brakes (including anti-skid)
  - loss of function to the left wing gear brakes.
- Partial loss of bleed air, resulting from damage to the:
  - left wing system ducting
  - APU bleed air ducting.
- Fuel system, resulting from:
  - fuel leakage from the No. 2 engine feed tank
  - a loss of function to the engines No. 1 and 2 low pressure fuel shutoff valves
  - the loss of function to the No. 1 engine high pressure shutoff valve
  - loss of function of numerous fuel system components (valves and/or pumps)
  - degradation of the fuel quantity management system
  - a reduction in capability of the automatic and manual fuel transfer function
  - disabling the fuel jettison system.
- Engine fire protection, as a result of a loss of function to one of the two extinguisher bottles in engines No. 1 and 2.

Further information on the damage to aircraft's systems is contained in Appendix B.

1.4 Other damage

A significant amount of debris from the aircraft fell over about 1.5 square km on Batam Island (Figure 11). A number of engine and other components were recovered from the island with the assistance of the local residents, Indonesian police, the National Transportation Safety Committee (NTSC) of Indonesia and the engine manufacturer.

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[10] A leading edge lift augmentation device.

Figure 11: Debris distribution on Batam Island

Image source: Google Earth.

A section of the intermediate pressure turbine disc (about 40% of the disc) from the No. 2 engine was recovered from a property on Batam Island (Figure 12). Despite a search carried out over several days, no other significant fragments of the turbine disc were recovered. A number of other engine components were recovered, including sections of engine cowling and other components from the No. 2 engine.

Several buildings and other property were damaged by the debris.

Figure 12: Property damage from a section of the No. 2 engine turbine disc

Image source: Courtesy of the 'Posmetro' newspaper, Indonesia.

1.5 Personnel information

All crew reported that they were adequately rested prior to commencing duty on the morning of the occurrence.

Each flight crew member's aeronautical experience at the time of the occurrence is outlined in the following paragraphs.

1.5.1 Captain

The captain held an Airline Transport Pilot (Aeroplane) Licence (ATPL(A)). He was endorsed on and operated Boeing 747, 747-400 and Airbus A330 aircraft before commencing operations on the A380. The captain completed his A380 endorsement on 11 April 2008 and his last proficiency check prior to the occurrence was completed on 14 October 2010. He held a valid Class 1 Medical Certificate with a requirement to wear reading glasses for vision correction.

The captain's aeronautical experience is outlined in Table 1.

Table 1: Captain's aeronautical experience

Total flying hours	15,140.4
Total flying hours in the A380	570.2
Total flying hours in the last 90 days	78.1
Total flying hours in the last 30 days	34.1
Total flying hours in the last 7 days	Nil

1.5.2 First officer

The first officer held an ATPL(A) and was endorsed on and had operated Boeing 747, 767, Airbus A330 and A340 aircraft before commencing operations on the A380. He completed his A380 endorsement on 29 August 2008 and his last proficiency check prior to the occurrence was completed on 16 September 2010. The first officer held a valid Class 1 Medical Certificate with no restrictions.

The first officer's aeronautical experience is outlined in Table 2.

Table 2: First officer's aeronautical experience

Total flying hours	11,279.5
Total flying hours in the A380	1,271.0
Total flying hours in the last 90 days	127.5
Total flying hours in the last 30 days	35.3
Total flying hours in the last 7 days	35.2

1.5.3 Second officer

The second officer held an ATPL(A) and was endorsed on and had operated Boeing 747 aircraft before commencing operations on the A380. He completed his A380 endorsement on 15 February 2009 and his last proficiency check prior to the occurrence was completed on 27 October 2010. The second officer held a valid Class 1 Medical Certificate with no restrictions.

The second officer's aeronautical experience is outlined in Table 3.

Table 3: Second officer's aeronautical experience

Total flying hours	8,153.4
Total flying hours in the A380	1,005.8
Total flying hours in the last 90 days	151.7
Total flying hours in the last 30 days	34.7
Total flying hours in the last 7 days	33.6

1.5.4 Check captain

The check captain held an ATPL(A) and was endorsed on and had operated Boeing 747, 747-400 and 767 aircraft before commencing operations on the A380. He completed his A380 endorsement on 1 March 2009 and his last proficiency check prior to the occurrence was completed on 31 October 2010. The check captain held a valid Class 1 Medical Certificate with a requirement for reading correction to be available while exercising the privileges of the licence.

The check captain’s aeronautical experience is outlined in Table 4.

Table 4: Check captain’s aeronautical experience

Total flying hours	20,144.8
Total flying hours in the A380	806.4
Total flying hours in the last 90 days	151.7
Total flying hours in the last 30 days	34.7
Total flying hours in the last 7 days	6.4

1.5.5 Supervising check captain

The supervising check captain held an ATPL(A) and was endorsed on and had operated Boeing 747, 747-400, 767 and Airbus A330 aircraft before commencing operations on the A380. He completed his A380 endorsement on 27 June 2008 and his last proficiency check prior to the occurrence was completed on 1 November 2010. The supervising check captain held a valid Class 1 Medical Certificate with a requirement for reading correction to be available while exercising the privileges of the licence.

The supervising check captain’s aeronautical experience is outlined in Table 5.

Table 5: Supervising check captain’s aeronautical experience

Total flying hours	17,692.8
Total flying hours in the A380	1,345.9
Total flying hours in the last 90 days	189.3
Total flying hours in the last 30 days	59.9
Total flying hours in the last 7 days	10.6

1.6 Aircraft information

1.6.1 General

The aircraft was a low-wing, high capacity transport category aircraft that was manufactured in France in 2008 (Figure 13). The aircraft could carry 259,471 kg of usable fuel (at 0.80 kg/litre) and about 450 passengers on two decks in various classes. The aircraft information is summarised in Table 6.

Figure 13: Airbus A380-842

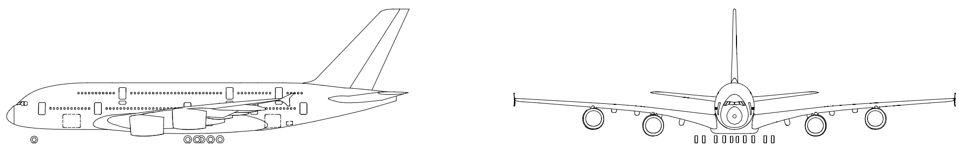

Image modified from an Airbus-supplied model

Table 6: Aircraft information

Manufacturer	Airbus
Type	A380-842
Serial number	0014
Total hours	8,533.02
Total cycles	1,843
Year of manufacture	2008
Certificate of Registration	4 September 2008
Certificate of Airworthiness	18 September 2008
Maximum take-off weight	569,000 kg
Actual take-off weight	464,040 kg
Maximum landing weight	391,000 kg
Actual landing weight	431,712 kg

1.6.2 Engine

The aircraft was fitted with four Rolls-Royce plc RB211 Trent 972-84 high-bypass turbofan engines. The engines were numbered 1 to 4 from left to right (Figure 14). A description of the engine and details of the damage sustained by the No. 2 engine are contained in Part 2 of this report.

Figure 14: Engine positions

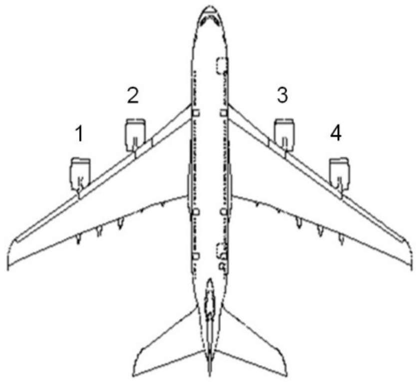

Image modified from an Airbus-supplied model

1.7 Meteorological information

Meteorological observations were taken at Changi Airport every 30 minutes. Between 00:00:00 and 04:00:00 the surface winds varied from 3 to 6 kt and were from the south-south-west. No significant weather was observed within the vicinity of the airport. Weather radar images indicated that there were no areas of precipitation between Singapore and Batam Island during the occurrence.

The flight crew reported weather conditions that were consistent with the recorded data, and that the flight was conducted in visual conditions.

1.8 Communications

1.8.1 Communication to and from the aircraft

In flight

Communication between the aircraft and ATC was through VHF radio. All of the communications with and by ATC were recorded and, on examination, found to be clear, with no communication issues reported.

On the ground

As previously discussed, following completion of the engine shutdown procedure, there was only one VHF radio available for flight crew communication with the ATC ground controller and the airport emergency services (AES) commander. It took the crew 2 to 3 minutes to establish direct communication with the AES commander via VHF radio. The AES commander reported that he was absent from his vehicle for a short period, during which he was assessing the situation and initial response of the fire fighting resources.

The flight crew used a mobile telephone to communicate with the operator's maintenance personnel.

1.8.2 Communication between the cabin and the cockpit

A number of cabin crew reported that they attempted to contact the flight crew immediately after the engine failure using the cabin interphone system, but were unsuccessful. This included an attempt by the CSM to contact the flight crew via the EMERGENCY contact selection on the cabin interphone system. The emergency contact function illuminated a light on the cockpit overhead panel (Figure 15) and sounded the flight deck horn. The flight crew reported that they associated the emergency contact-activated flight deck horn with the ongoing warnings from the ECAM and so cancelled the horn without recognising its association with the cabin interphone system emergency contact function.

Figure 15: Location of the cabin emergency call light

Observation:

Given the crews' attention on the ECAM and the associated procedures, the location of the cabin emergency call light and the tone of the warning horn were not sufficient to obtain the flight crew's attention. Although this did not result in an unsafe situation, the lack of conspicuity of the cabin emergency call function delayed the transfer of potentially important information to the flight crew.

1.9 Aerodrome information

1.9.1 Runways

Changi Airport had three parallel runways oriented north-north-east to south-south-west on magnetic headings of 023°/203°. Those runways were designated 02/20 left (L), centre (C) and right (R) indicating their relative position when looking along the runway. Runways 20C and 20R were the longest runways available, both being 4,000 m long and 60 m wide. The aircraft departed from and landed on runway 20C.

1.9.2 Emergency services

The airport was designated as a category 10 aerodrome[11] for the purposes of rescue and fire fighting support and had a full-time AES.

In responding to the PAN call from the aircraft, the local AES turned out 8 fire fighting vehicles and 20 firefighters to deal with the aircraft and passenger safety. Additional ground staff attended the aircraft in response to requests for stairs and busses. In dealing with the on-ground phase, the AES reported that they used 60,000 L of water, 3,500 L of fire fighting foam and 18 kg of dry chemical powder.

1.10 On-board recorded information

1.10.1 Flight recorders

The aircraft was equipped with the following flight recorders:

- a flight data recorder (FDR)
- a cockpit voice recorder (CVR)
- a wireless digital aircraft condition monitoring system recorder.

The fitment of FDR and CVR equipment was mandatory for this aircraft. The parameters to be recorded on the FDR and the audio to be recorded on the CVR were defined by regulation. The recorded flight and audio data was stored within each recorder's crash-protected memory module.

Initial examination of the FDR data showed that it contained recorded data from the entire flight; this was consistent with the capability of the recorder.

The CVR audio recording was of 2 hours duration, as required by the relevant legislation. This recording commenced during the landing approach and continued during the subsequent ground operations.

1.10.2 Isolation of the CVR

After the passenger disembarkation process was completed, representatives from the Air Accident Investigation Bureau (AAIB) of Singapore approached the flight crew to request the isolation of the CVR. The representatives were concerned that, as the No. 1 Engine was still running, the CVR could still be powered and audio data from 2 hours previous was being overwritten. The flight and ground crews attempted to isolate the CVR but without success. The CVR using the endless loop principle continued recording, consequently overwriting the oldest audio data.

At the time of the occurrence, the operator did not have, and was not required by Australian aviation regulation to have a procedure that would enable the CVR to be isolated in conditions of this nature. In contrast, Singapore legislation stated that reasonable measures may be taken to preserve any object or evidence deemed

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11 Category 10 was defined in the International Civil Aviation Organization (ICAO) *Annex 14 – Aerodromes* as having the capacity to provide fire fighting and rescue support for aircraft with wingspans of between 76 m and 89 m, with a maximum fuselage width of 8 m. The A380 has a wingspan of 79.75 m and a fuselage width of 7.14 m.

necessary for the purposes of the investigation. See Appendix C for further discussion on this subject.

Observations:

The CVR recording capability met the 2 hour requirement; however, in this occurrence this was not adequate to record both the engine failure event and the subsequent ground operations. CVR audio prior to and around the time of the engine failure would have been extremely useful to this investigation in examining the crew resource management response to the event and exact ECAM sequences. The CVR audio would have been able to provide a correlation with other sources of flight information allowing the ATSB to more quickly develop an understanding of the technical issues and the flight crew's response.

The overwriting of the CVR in this event was cited in a Working Paper presented by the ATSB to the ICAO flight recorder working panel in October 2012, to support the panel proposal for increased CVR duration. This proposal resulted in a recommendation by the panel to the ICAO Air Navigation Committee in January 2013[12] to extend the mandated CVR recording capability to 15 hours.

Two aircraft network server units (ANSU) formed part of the aircraft's computer network architecture and contained duplicate recorded data sets from the FDR, aircraft condition monitoring system and the aircraft's centralised maintenance system (CMS). These data sets included a virtual quick access recorder, smart aircraft condition monitoring system recorder (SACMSR), aircraft system reports and post-flight reports from the CMS.

Additional recorded information was available from the engine monitoring unit (EMU) and engine electronic control (EEC) on each engine. The EMU and EEC data included detailed information specific to the operation of that engine.

Details of the recording systems and analysis of the data contained in those systems is in Appendix C.

1.11 Fire

1.11.1 No. 2 engine external fire

Following the uncontained engine failure, an external fire developed in the lower left region of the No. 2 engine nacelle. A portion of the thrust reverser assembly was burnt in the fire (Figure 16).

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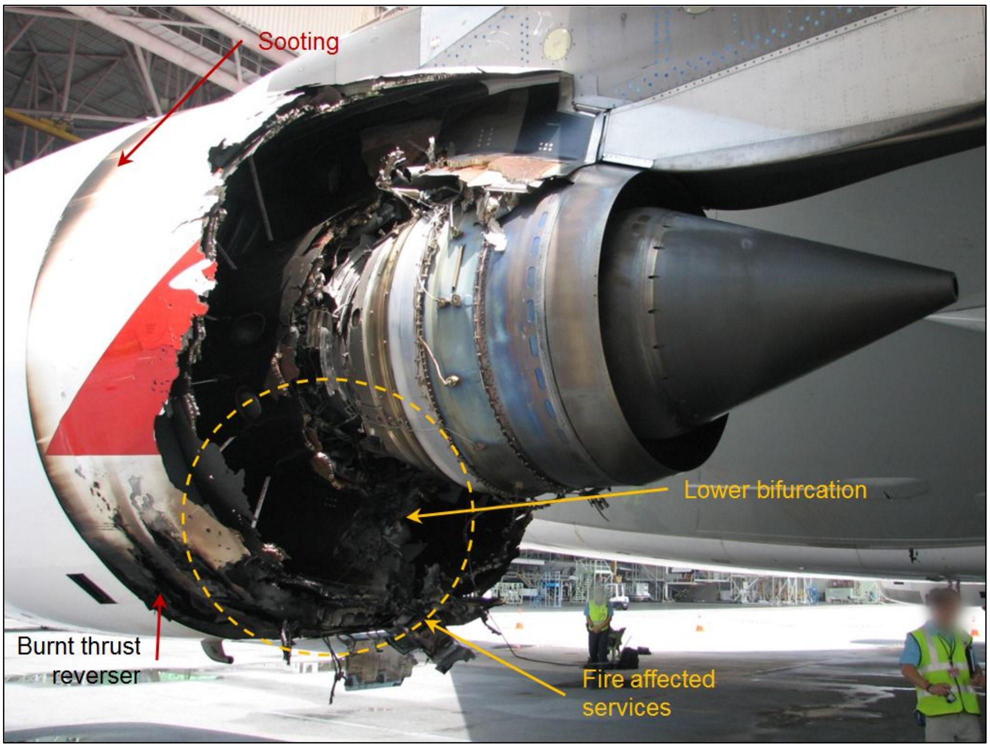
12 AN-WP/8697 dated 29/01/13 amending Annex 6 Part 1.

Figure 16: No. 2 engine showing fire damage to the lower left engine nacelle



Initial on-site inspection identified that the fire was most intense in the lower left thrust reverser section and in the lower bifurcation[13] (Figure 17 and 18). Sooting, glazing and the effects of heat were observed on a number of electrical, air, oil and hydraulic supply lines and the nacelle anti-ice bleed air ducts within the core section, lower bifurcation and accessories section (on the fan casing). A significant amount of oil was smeared on the lower panel of the thrust reverser cowl (Figure 18).

13 A vertical space within the thrust reverser section where the left and right half of the thrust reverser cowls join. The lower bifurcation provides a space for the various service pipes to be routed between the core section of the engine and the accessories section on the fan case.

Figure 17: Example of fire damage in lower the bifurcation (looking to the forward and right of the No. 2 engine)

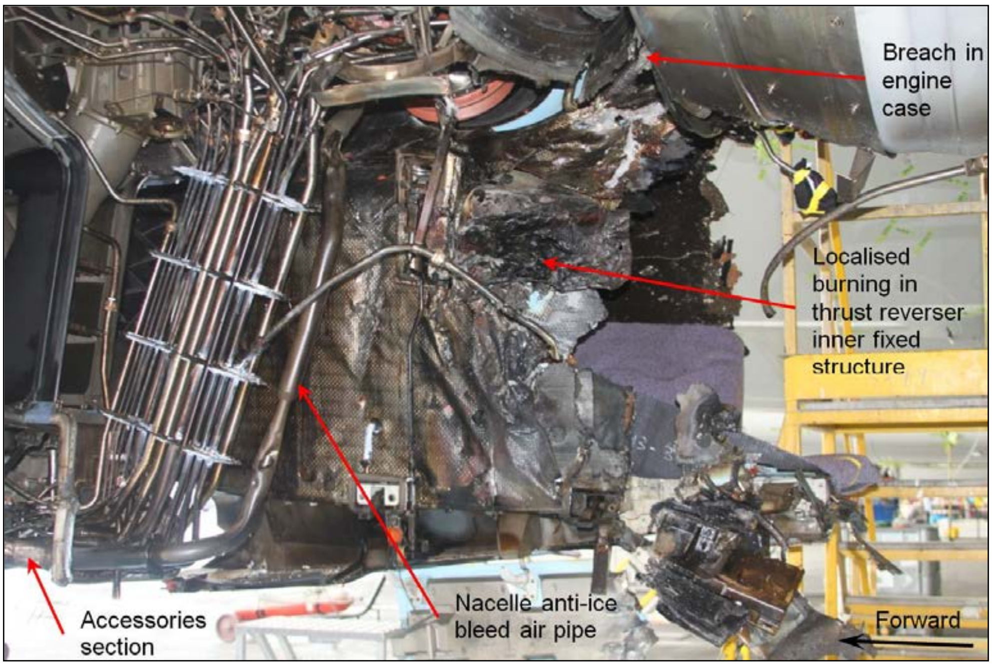
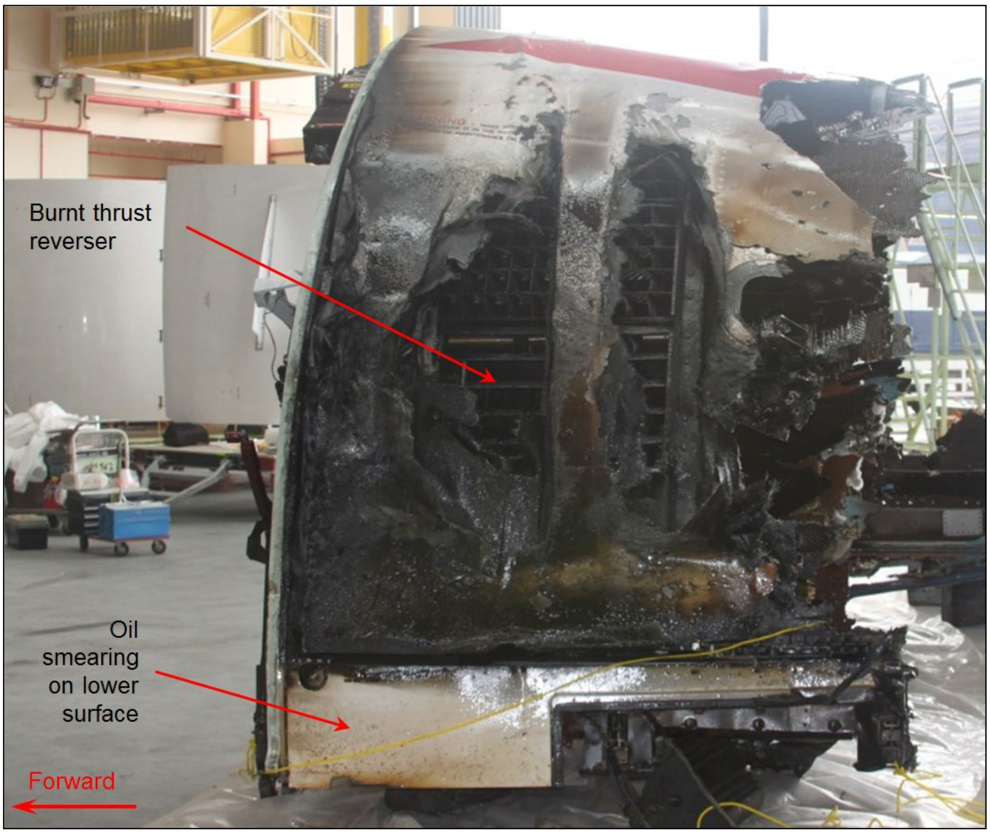
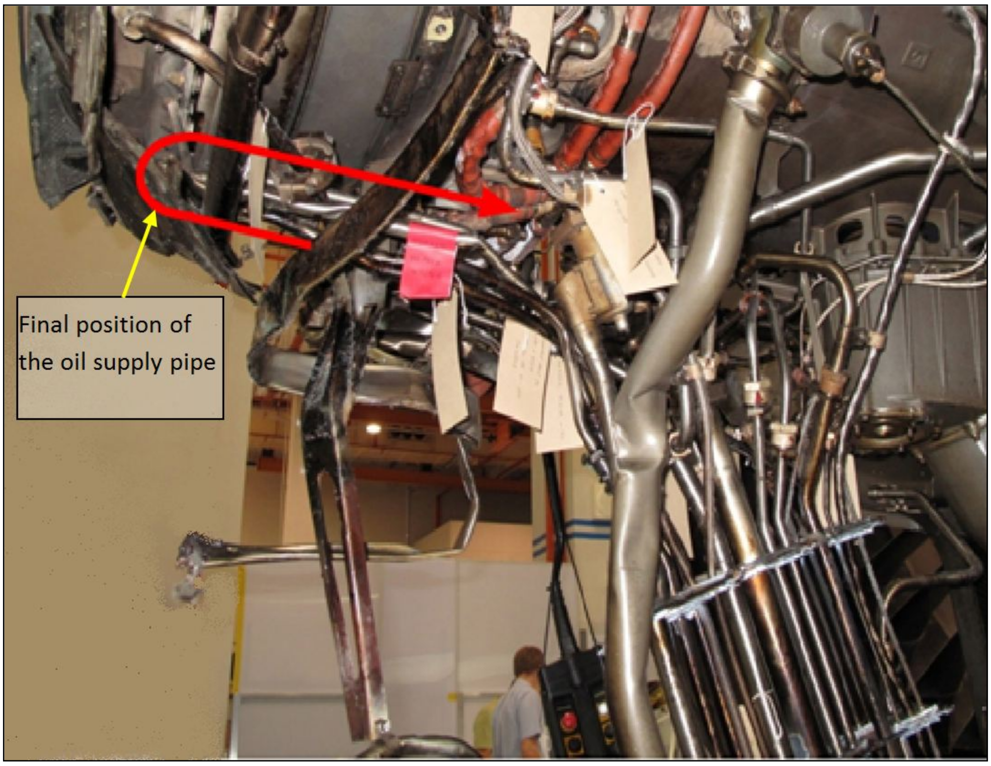


Figure 18: Left thrust reverser section (looking at the lower surface of the engine)



During the subsequent engine inspection, it was noted that an oil supply pipe to the HP/IP and low pressure (LP) turbine bearings that was located within the lower bifurcation had been severed and was kinked forward (Figure 19). That damage was directly below the breach in the engine case and was consistent with being a consequence of the uncontained engine failure. The damage to the supply pipe was also coincident with a region of intense burning on the thrust reverser inner fixed structure.

Figure 19: Representation of the position of the oil supply pipe to the HP/IP and LP turbine bearing chambers (severed and bent forwards during the engine failure sequence)



The nacelle anti-ice bleed air duct passed through the lower bifurcation. The bleed air was taken from the engine's high pressure compressor and had a temperature of around 500 °C. The duct was not insulated and there were a number of holes in the duct.

The ATSB recovered less that 0.5 L of oil from the No. 2 engine. The recorded data indicated that the engine oil quantity decreased from 13.5 L to a small quantity in around 45 seconds following the uncontained failure.

Engine fire detection system and warnings

The engine fire detection system consisted of five detectors as shown in Figure 20. The detectors could provide any of three signals to the fire detection unit: NORMAL, FAULT or FIRE. Each of those detectors had two signal channels, or 'loops' (A and B) that sent a signal to the fire detection system.

Figure 20: Fire detector layout

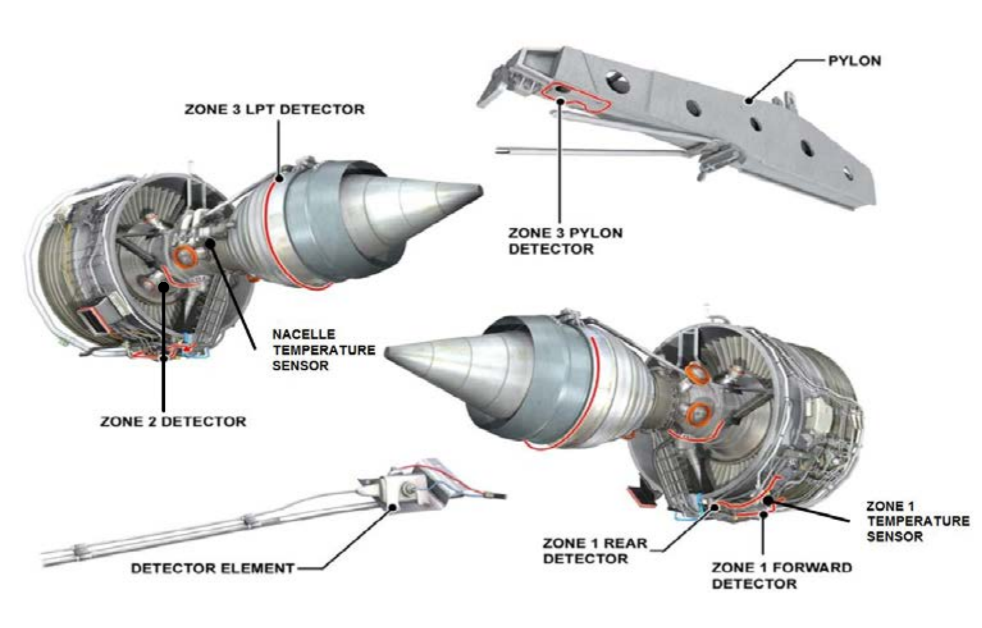


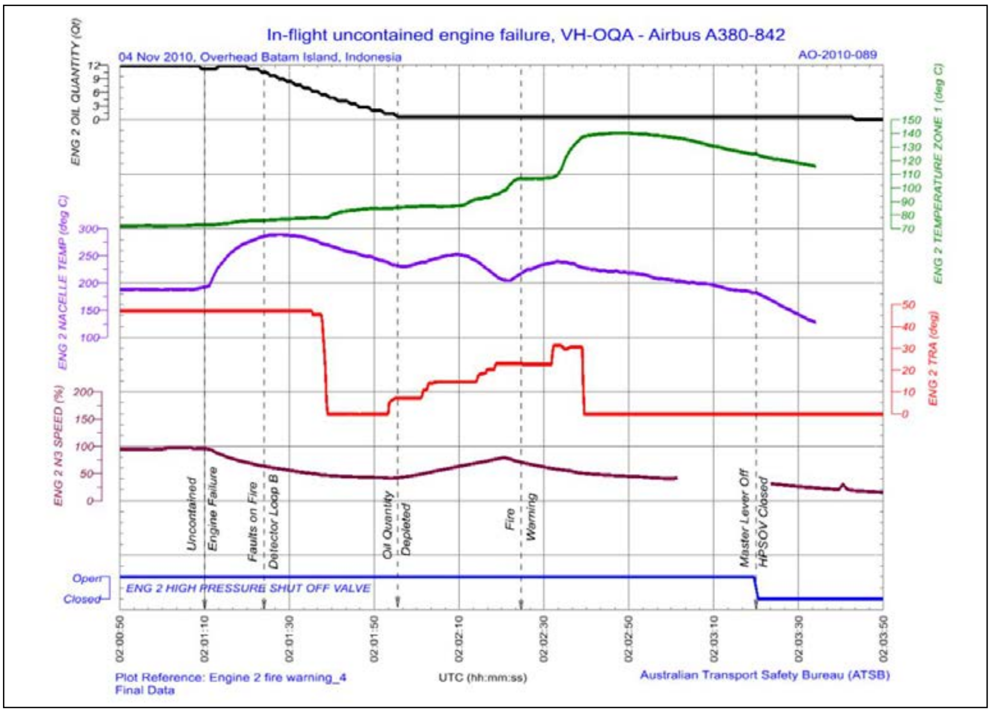
Image source: Rolls-Royce RB211-Trent 900 Line and Base Maintenance training guide.

During the event, the power supply to No. 2 engine loop A was interrupted when the cable was cut by debris, resulting in a FAULT condition on all of the fire detectors in that loop. The recorded data showed that faults were also detected in loop B of the Zone 2 and 3 detectors. This meant that the only remaining operable engine fire detectors were in zone 1 (the fan casing).

The recorded data showed that at 02:01:52 the No. 2 engine thrust lever was advanced (Figure 21). There were increases in the zone 1 temperature that followed further thrust lever inputs, suggesting a correlation. That apparent correlation became more significant at higher thrust lever settings. When the thrust lever was returned to idle at 02:02:39 the temperature in that zone stabilised.

At 02:02:25 a fire warning was displayed for 2 seconds before being reset. Figure 21 shows that around this time the temperature in zone 1 had increased to about 107 °C and shortly after rapidly increased further, reaching a maximum of about 140 °C before slowly decreasing.

Figure 21: Engine fire – key data



Engine fire suppression system

The engine fire suppression system consisted of two fire extinguishing bottles for each engine, a control subsystem and a distribution subsystem (Figure 22). A post-flight inspection of the aircraft revealed that one of the No. 2 engine's fire bottles had discharged.

The aircraft manufacturer determined that the damage to the wiring from the uncontained engine failure had prevented the discharge of the second bottle and the provision to the flight crew of an indication of the status of the bottles. This prevented an accurate determination of when the crew attempted to activate the fire suppression system.

Figure 22: Fire suppression system for engine No. 1 (similar system for engines No. 2, 3 and 4)

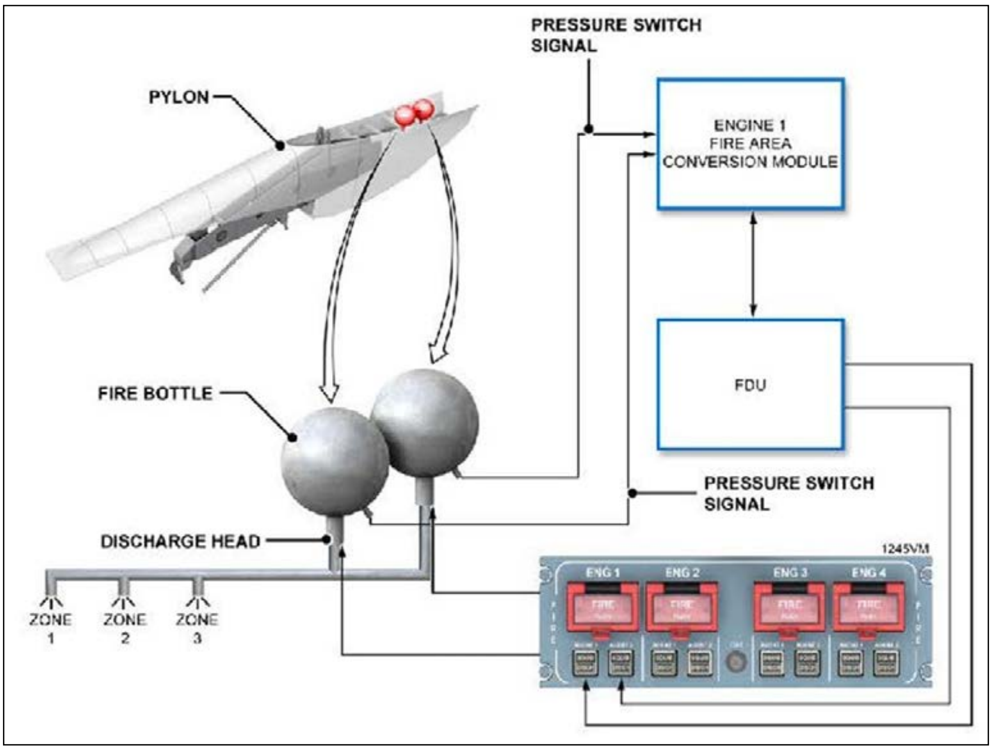


Image source: Rolls-Royce RB211-Trent 900 Line and Base Maintenance training guide
Note: FDU = fire detection unit

Observations:

The evidence was consistent with the commencement of a fire in the engine No. 2 thrust reverser lower bifurcation duct when an oil line was severed by ejected material during the uncontained engine failure. The specific ignition source was unable to be conclusively determined, however, the region had a number of sources of hot air (nacelle anti-ice bleed air and hot gases released from the engine after the failure) and hot surfaces that were sufficient to ignite any leaking oil in that area.

Although there was no direct measurement of the temperature in the area, a nearby fan casing measurement showed that the temperature in the region continued to rise after the engine oil quantity had depleted. This is consistent with a local fire, where the pipe had been severed, and a larger area fire in the lower section of the thrust reverser. This was itself consistent with oil pooling in the bottom of the nacelle where an intense fire burnt the thrust reverser and entered the accessory gearbox area (zone 1). It is likely that the rear fire detector in zone 1 triggered the No. 2 engine fire warning.

Although the temperature in zone 1 levelled out then slowly decreased when the thrust levers were returned to idle, a direct relationship between the zone 1 temperature and thrust lever movement could not be established with certainty. Other possible reasons for the reduction in the zone 1 temperature included the consumption or loss overboard, and therefore removal as an energy source, of the pooled oil in the lower bifurcation, or as a result of the discharge of the fire bottle.

1.11.2 Internal wing flash fire

Immediately following the engine failure, a passenger who was seated on the upper left deck of the aircraft, reported observing a fire coming from the hole through the left upper wing structure (Figure 7). The passenger reported to the ATSB that he observed a fire that lasted for about 5 to 6 minutes. None of the flight crew reported observing any fire, any evidence of a fire, or being aware of a fire in that area of the left wing.

The ATSB examined the left wing and found a region inside the left inner fuel tank that had a dark residue on the wall surfaces (Figure 23). A similar dark residue was also found behind the hole in the exterior upper wing surface. Samples of that residue were tested in an independent laboratory, where those residues were determined to be consistent with products of combustion. The painted surfaces immediately underneath the residue did not exhibit any thermal damage, discolouration or blistering.

Figure 23: Dark residue in left inner wing fuel tank (looking up and forward)

The ATSB conducted a detailed inspection of the left inner fuel tank region to identify potential sources of ignition. The surface temperature of the turbine disc fragment and the frictional effects associated with the passage of the fragment would have been sufficient to ignite the fuel vapour within the tank volume. No other potential sources of ignition were identified as being sufficient to have initiated the fire.

The ATSB determined that the conditions within the tank were suitable for a short duration (flash) fire; however, they were not suitable to sustain the fire. A detailed report of the ATSB's examination and analysis of the fire in the left inner fuel tank is at Appendix D.

Observation:

The ATSB determined that the passenger had a clear view of the No. 2 engine through the damaged section of the wing. The duration of the fire observed by the passenger probably indicated that they observed the No. 2 engine external fire (as discussed in section 1.11.1) through the holes in the left inner fuel tank, rather than the flash fire.

1.11.3 Brake overheat

The AES reported that after the aircraft came to rest, they observed smoke emitting from the left body gear wheels. The emergency responders described an 'incipient fire' developing in the landing gear, and that the tyres had deflated. As a precaution, they discharged dry powder fire retardant in that area.

Even though there was evidence that components of the left wing body gear wheels were affected by high temperatures, a post-flight inspection revealed no evidence of a fire, such as melting or charring, in any of the affected wheels.

1.12 Survival aspects

There were no injuries as a direct result of the uncontained engine failure and the cabin environmental systems remained functional during the occurrence and return to Changi Airport.

1.12.1 Portable electronic devices

After the passengers boarded, as part of the normal departure procedures, the cabin crew made a public address announcement requiring that all portable electronic devices (PEDs) were to be switched off and could not be turned on again until the aircraft had reached cruising altitude and the seat belt sign had been turned off. The engine failure occurred prior to the aircraft reaching cruise altitude and, after the failure, the seat belt sign remained on for the rest of the flight. Therefore, the conditions under which passengers could use PEDs were not met and permission to use them was not given. The cabin crew reported that additional PA announcements were made following the engine failure instructing passengers to remain seated with their seat belts fastened and to keep PEDs switched off.

Video and still images showed that some of the passengers did not comply with the crew's instructions, such as:

- moving about the cabin when they had been instructed to remain seated
- using PEDs during flight, and in particular during the approach, landing and post landing phases, when instructed to switch them off.

Observation:

Although the images taken by the passengers during the flight provided supplemental information to the ATSB, the use of PEDs has a number of known safety risks. Specific to this accident is the potential for interference with aircraft systems and the risk of passenger distraction at critical times during the emergency situation.

Although it is difficult to replicate the effects from interference from PEDs on aircraft systems, there is strong anecdotal evidence to show that they can have a detrimental effect.[14] This can include erroneous navigation information, false warnings, reduced crew confidence in warning systems and crew distraction.

During an emergency situation, crew members provide instructions to passengers regarding their safety. Many of those instructions will be significantly different to the normal announcements made during a flight and contain specific information and

[14] Balfe N. and Head T (2004). *Passenger Use and Perception of Personal Electronic Equipment on Board Aircraft*. International Journal of Applied Aviation Studies, 4, Number 2. FAA Academy, Oklahoma City, OK.

instructions not normally provided to passengers. It is important for the safety of all passengers that these instructions are listened to and understood. Human attentional resources are a single-channel, filtered and limited resource[15] and PEDs have been shown in numerous research studies to have a significant distracting effect. In an emergency situation, such as an evacuation, actions need to be carried out quickly and there can be insufficient time for crew to be repeating information to passengers distracted by their PEDs.

1.12.2 Passenger disembarkation

The operator's *Flight Crew Emergency Procedures Manual* (FCEPM) categorised the emergency egress of an aircraft's occupants into three basic classes: unprepared emergencies, prepared emergencies and precautionary disembarkation. Aside from an egress in the case of an unprepared emergency, the captain was responsible for determining which class of emergency egress was to be used. The highest level of urgency was an evacuation, where the crew were required to use all available exits, specific commands and set procedures to ensure that the aircraft was evacuated as rapidly as possible. This method of disembarkation also presented the greatest risk of passenger injury.[16]

The FCEPM defined a precautionary disembarkation as an emergency situation where passengers are disembarked as quickly as is deemed safe, however, the pace of the disembarkation is slower in an effort to minimise potential injury to passengers or crew. The FCEPM also stated that this form of disembarkation can be upgraded quickly to a full evacuation if the circumstances change.

As both the evacuation and the precautionary disembarkation methods of emergency egress were classed as 'prepared on the ground' emergencies, they both used a common initial alert to the cabin crew. This was achieved through a specific PA announcement that placed the cabin crew in the 'alert phase'.[17] While in the alert phase, the flight crew were required to assess the safety of the aircraft, taking into account the internal and external environments and determine the type of egress to be used should an emergency egress become necessary. The alert phase may take several minutes while the flight crew carry out the applicable shutdown procedures and establish communications with the AES to obtain information on the external environment around the aircraft.

Following the landing, the flight crew actioned the applicable evacuation checklist, assessed the information that had been provided to them by the AES and decided that a precautionary disembarkation using stairs would be the most appropriate and safest method. The captain briefed the CSM on the plan and addressed the passengers regarding the disembarkation.

The initial plan was to use the main deck 1R door for the disembarkation and that door was disarmed in preparation for the arrival of a set of stairs. However, the

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15 Wickens, C.D. and McCarley, J.S. (2008). *Applied attention theory*. CRC Press: Boca Raton, FL.

16 A safety study of emergency evacuations carried out by the US National Transportation Safety Board (NTSB/SS-00/01, 27 June 2000) found that 8 per cent of the people involved in the evacuations studied sustained injuries during evacuation (2 per cent serious and 6 per cent minor). A number of the injuries were related to the emergency type (for example, smoke inhalation from a fire) while others were directly related to the evacuation, such as fractures.

17 During the alert phase, all cabin crew were to remain at their assigned stations, with all doors armed. This allows the crew to immediately activate the escape slides should they be needed.

stairs were actually placed in position at the main deck 2R door. There were hurried communications between the cabin and flight crew to ensure that there was no accidental deployment of a door slide when main deck 2R door was opened. The flight crew also instructed the cabin crew to prevent any subsequent evacuation from the left side of the aircraft while the No. 1 engine continued to run.

To ensure that passengers safely entered a controlled environment outside the aircraft and did not wander into areas where a significant hazard still existed, the disembarkation was carried out in a managed fashion by the AES. That management included passengers being disembarked from the aircraft in groups to fill a single bus load at a time.

Observation:

Given that there was no indication of an immediate threat to the safety of those on board, and that the option of an immediate evacuation remained throughout, the crew's decision to evacuate via the stairs likely provided the safest option. With the uncontrolled No. 1 engine, fuel leakage hazard and the large number of passengers, the airport emergency services action to control the passengers in proximity to the aircraft reduced risk to the passengers themselves, the crew and emergency services.

1.13 Tests and research

1.13.1 Simulation of the occurrence in a fixed base simulator

Following the occurrence, Airbus conducted a simulator session to assess the aircraft's handling characteristics after the uncontained engine failure. That simulator session was carried out at the Airbus facilities in France during April 2011 and included participants from Airbus, the Direction générale de l'aviation civile (DGAC)[18], the Bureau d'Enquêtes et d'Analyses pour la sécurité de l'aviation civile (BEA)[19] and, acting on behalf of the ATSB, one of the pilots from the occurrence flight.

The simulations were carried out in an A380 fixed base simulator (that is, a simulator without motion capability) using a certified aerodynamic model. The simulator was connected to and controlled by actual computers of the standard fitted to the occurrence aircraft and were configured to represent the state of the aircraft during the return and landing after the event. There were a number of slight differences in how the systems were failed as compared to the effect of the uncontained engine failure (entire systems were failed rather than specific wires being disconnected or cut to simulate the damage from the liberated engine debris). According to the manufacturer, this did not change the characteristics, but may have affected the presentation of ECAM messages.[20]

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[18] The DGAC is the French agency with responsibility for the regulation of civil aviation under its jurisdiction.

[19] The BEA is the French agency with responsibility for technical investigations into civil aviation accidents or incidents under its jurisdiction. The BEA was an accredited representative to the ATSB investigation under ICAO Annex 13.

[20] This was considered acceptable because the purpose of the session was not to examine the flight crew responses to the ECAM.

The design of the simulator set-up was aimed at replicating the occurrence aircraft's responses during the event as accurately as possible. The primary purpose of the simulation session was to assess the aircraft's:

- flight envelope protections
- handling qualities and controllability at a range of lateral fuel imbalances
- control on the ground and required landing distance with maximum braking and full reverse thrust selected on No. 3 engine versus differential braking.

In all cases the flight envelope protections functioned as anticipated. In addition, it was determined that the flight crew could maintain control of the aircraft in all fuel imbalance configurations tested. The effect on the aircraft's controllability in those configurations was rated as 'minor'.[21]

Each of the simulator scenarios concluded when the aircraft had landed and come to a full stop. On each landing the flight crew applied maximum braking, with full reverse thrust applied to the No. 3 engine and directional control via the rudder pedals. On each occasion the simulated aircraft remained near the runway centreline and stopped in the second third of the runway with 1,000 to 1,500 m of runway remaining.

Observation:

The primary aim of the simulation session was to test the aircraft's handling qualities with the known airframe and systems degradation resultant from the uncontained engine failure. Despite significant system and structural damage following the uncontained engine failure, the simulation identified that the aircraft had sufficient redundancy to continue safe operation.

1.13.2 Engineering analysis of the occurrence

An engineering analysis by the airframe manufacturer applied the conditions from the event flight using the certified aircraft model, but with the following differences:

- maximum braking effort was applied for the duration of the landing roll
- full right thrust reverser was applied down to 75 kt, followed by idle reverse thrust until coming to a full stop (*Flight Crew Operating Manual* (FCOM) procedure).

The total estimated landing distance was calculated in the manufacturer's analysis to be 2,300 m. This left a 1,700 m margin in the case of a landing on the 4,000 m runway 20C at Changi Airport.

Observation:

The difference between the manufacturer's engineering analysis of the landing distance and the actual aircraft stopping distance may primarily be attributed to the flight crew not applying maximum braking for the duration of the landing. As maximum braking effort was not required to stop the aircraft within the confines of the runway, the risks associated with maximum effort braking were reduced.

[21] Based upon the Cooper-Harper rating scale. The Cooper-Harper rating scale is a standardised aircraft handling quality rating scale that was developed by the US National Advisory Committee for Aeronautics (the predecessor to National Aeronautics and Space Administration (NASA)).

1.14 The landing distance performance application

The onboard information system (OIS) could be operated through an onboard information terminal (OIT) that was incorporated into the captain's and first officer's instrument panels, or through synchronised laptop computers stowed at each of the flight crew stations (Figure 24). The OIS included an operations library application that supported a complete suite of aircraft operations documentation, navigation charts, and performance applications. The performance applications enabled the flight crew to determine aircraft performance data including take-off and landing data. The program for determining landing performance was referred to as the landing performance application (LPA). Certain OIS functions, including the LPA, were also available on a third laptop computer stowed at the rear of the cockpit.

Figure 24: Onboard information system

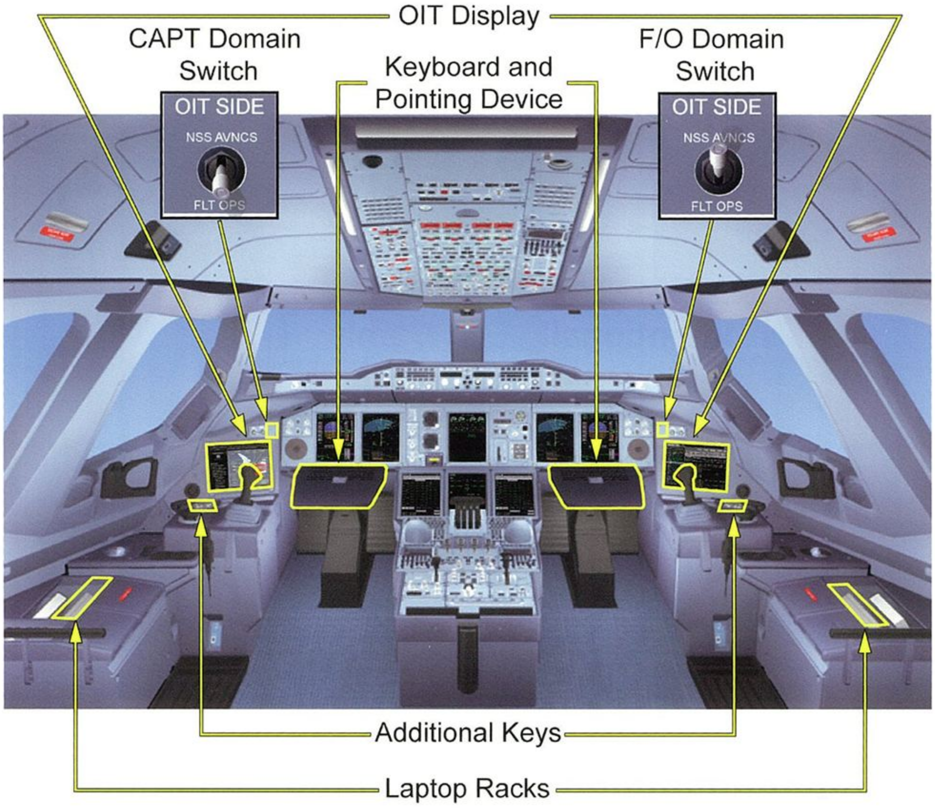

Image modified from an Airbus-supplied model

The pre-landing procedures required the flight crew to enter information into the flight management systems (FMS), including the intended approach and landing runway, the aircraft's landing configuration and the weather conditions for the arrival airport. The FMS data, which included the aircraft's weight and centre of gravity, as well as the ECAM list of any system failures that affected landing performance, were automatically synchronised with the captain's and first officer's OITs. The flight crew could manually modify the synchronised system data and add or remove data as required.

1.14.1 LPA operational logic

In determining the landing distance, the LPA calculations accounted for aircraft system failures by multiplying the calculated landing distance by an applicable factor. The LPA also contained a degree of ‘conservatism’ through the application of an operational coefficient, to account for variations in landing technique between the average pilot and the ideal procedure.

The manner in which that conservatism was applied by the LPA differed depending on whether the landing weight was above, or at or below, the maximum landing weight.[22] When the weight was at or below the maximum landing weight, the operational coefficient was applied to each of the system failure factors. However, when the landing weight was above the maximum landing weight, the LPA only applied the operational coefficient once. That was, the operational coefficient for weights at, or below, the maximum landing weight was more conservative than for weights above the maximum landing weight. This could result in calculated landing distances that were greater for lower weights.

1.14.2 LPA calculations during the occurrence

During the preparations for landing, the additional flight crew attempted to determine the landing distance for a landing at Changi Airport runway 20C using the third laptop. They manually entered data including the aircraft’s maximum landing weight,[23] the runway and weather conditions and nine inoperative aircraft systems. In calculating the landing distance, the LPA applied the additional operational coefficient, and therefore the inherent conservatism nine times, reflecting the number of inoperative aircraft systems entered. The landing distance calculated was greater than the landing distance available and the LPA provided a ‘no result’ message.

The crew recalculated the landing performance data, but this time used the actual aircraft weight. This revised calculation indicated that a landing on runway 20C was feasible, with 100 m of runway remaining. The crew elected to proceed on that basis.

1.15 Flight envelope protection alerts during the final approach

Reports from the flight crew and data from the FDR indicated that the aircraft activated three flight envelope protection alerts during the approach and landing. The first two were ‘low energy’ aural alerts, with the first being triggered as the aircraft passed 1,000 ft radio altitude (RA)[24] and the second at 362 ft RA. On each occasion the captain appropriately responded by increasing thrust.

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22 In an emergency, a landing above the aircraft’s maximum landing weight was permitted, but required a subsequent maintenance inspection.

23 The actual landing weight was 41 T in excess of the maximum landing weight.

24 Altitude determined by radar altimeter.

The third warning, a stall warning associated with the aircraft's angle of attack,[25] was generated at 3 ft RA, immediately prior to the aircraft touching down on the runway.

To prevent a loss of control from an aerodynamic stall, the A380 flight control system in NORMAL law included a number of low energy protections. Those protections were based on the angle of attack and included a low energy warning and alpha[26] floor. According to the FCOM:

> The objectives of the low energy alert and of the alpha floor function are to protect the flight path angle by providing means to achieve a proper level of energy:
>
> - The low energy alert is the first level of protection. It requires crew action, for manual thrust increase.
>
> - The alpha floor function is the second level of protection. It automatically sets maximum thrust.

As a result of the damage to the aircraft systems, primarily the loss of the leading edge slats, the aircraft's control system reverted from NORMAL to ALTERNATE 1A law. Although this included an ECAM message to advise the crew that the flight envelope protections had been lost, the only protection lost was alpha floor. Under ALTERNATE 1A law, the stall warning was restored.[27]

Post-accident analysis by Airbus identified that, as a result of the control system laws and the damage to the aircraft, all three warnings were genuine and that the flight envelope margins were maintained.

Observation:

The warnings produced by the aircraft's flight control computers were genuine and provided for the appropriate margins to avoid an aerodynamic stall during the approach.

The increase in thrust setting by the captain, in response to the 'low energy' aural alerts, were in accordance with the operating procedures.

1.16 Crew inability to shut down the engine

1.16.1 Guidance material

The guidance in ICAO Annex 14 – Vol 1 - *Aerodromes* included information regarding airport rescue and fire fighting services at an airport. Additional detailed guidance was provided in the ICAO Airport Services Manual, Part 1 – *Rescue and Fire Fighting*.[28] Although neither publication included any material directly related to shutting down an uncontrolled engine, the following guidance was provided in the ICAO Airport Services Manual:

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25 Angle of attack is the angle of the wing relative to the oncoming airflow.

26 Alpha (α) is used to represent angle of attack.

27 Stall warning was not required for NORMAL law as the system would automatically provide the protection to prevent the aircraft from stalling.

28 ICAO Document No. 9137

12.2.9 *Confined turbine engine fires (jet)*. Fires confined to the combustion chambers of turbine engines are best controlled when the flight crew is in a position to keep the engine turning over and it is safe to do so from the viewpoint of aircraft evacuation and other safety considerations. Fire fighters will have to stand clear of the exhaust but may have to protect combustibles from exhaust flames. Fires outside the combustion chambers of turbine engines but confined within the nacelle are best controlled with the built-in extinguishing system. If the fire persists after the built-in system has been expended and the turbine shut down, halon or dry chemical may be used to attempt extinguishment.

12.2.10 Foam or water spray should be used externally to keep adjacent aircraft structure cool. Foam should not be used in the intake or exhaust of turbine engines unless control cannot be secured with other agents and the fire appears to be in danger of spreading.

12.2.11 Rescue and fire fighting personnel should stay at least 7.5 m from the intake of an operating turbine engine to avoid being sucked in, and 45 m from the rear to avoid being burned from the blast.

12.3.24 *Engine running.* It is often necessary to keep at least one engine operating after the aircraft has come to a stop in order to provide lighting and communications aboard the aircraft. This will hamper aircraft rescue operations to some extent and consideration should be given to this problem.

1.16.2 Previous industry occurrences

The flight crew's inability to shut down the No. 1 engine on the ground increased the risk to firefighters, rescue personnel, and crew and passengers being evacuated or disembarked from the aircraft. There have been a number of previous events where a flight crew has been unable to shut down an engine and there were survivors or firefighters in attendance at the aircraft:

1982 – A McDonnell Douglas DC10 had a runway excursion and entered a body of water adjacent to the airport. The nose of the aircraft was detached during the impact. The No. 2 engine remained in full reverse thrust for approximately 30 minutes during which time the evacuation of the aircraft took place. The thrust from the No. 2 engine interfered with the operation of the two rearmost slides.[29]

1993 – A Boeing 747 departed the runway during the landing roll and entered an adjacent lagoon. The subsequent water ingress into the electronics bay in the nose rendered all control to the No. 1 engine inoperative. The No. 1 engine continued to run at above ground idle for some time until it was stopped by the local AES by drowning the engine using foam and water.[30]

2000 – A Convair 580 turboprop departed the runway during the landing roll. The flight crew were unable to shut down the left engine as a result of damage to the aircraft. The crew attempted to use three different methods

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[29] National Transportation Safety Board - Aircraft Accident Report - NTSB-AAR-82-15, World Airways, Inc, Flight 30H, McDonnell Douglas DC-10-30CF, N113WA, January 23, 1982.

[30] BEA Aircraft Accident Report – F-TA930913, Air France Boeing B747-428, 13 September 1993, Papeete-Faaa Airport, Tahiti.

of shutting down the engine, all of which were unsuccessful. The engine continued to run for 15 minutes during which time the passengers were evacuated from the aircraft. The engine finally stopped when it overheated due to a lack of lubricating oil.[31]

2007 – During an engine run as part of pre-acceptance tests an Airbus 340 was severely damaged when the aircraft impacted engine-blast suppression walls surrounding the run-up bay. The cockpit of the aircraft was separated from the cabin and two engines continued to run and could not be stopped using either the engine FIRE push buttons or the engine master switches in the cockpit. One engine was able to be stopped 2 hours and 45 minutes after the accident by spraying water and foam into the intake. The other engine was too close to an external structure to be able to get sufficient water/foam sprayed into the intake to shut it down. It finally stopped 9 hours after the accident when the fuel supply in the feed tank was exhausted.[32]

1.16.3 Airport emergency services procedures

At the time of the occurrence, the local AES did not have a procedure or guidance for shutting down an uncontrolled engine.

After attempting all of the operational and engineering methods for shutting down the No. 1 engine, and following consultation with the operator, a decision was made to inject water and foam directly into the intake. Although the delivery rate of water alone was not sufficient to shut the engine down, the use of foam was successful at drowning the engine.

Observation:

Although the guidance material did not provide specific material on a process for shutting down an uncontrolled engine, it did include information on the precautions to be taken when an engine is operating. This provided a level of safety to rescue and fire fighting personnel and aircraft occupants in the case of an operating engine during an emergency response, as was the case with this occurrence.

The implications of the continued running of the No. 1 engine were mitigated in this occurrence by the actions of the crew and emergency services personnel. There may be situations where this is not possible. Previous similar occurrences indicate that a number of airport emergency services either had a procedure, or determined a suitable means of shutting down an uncontrolled engine or protecting personnel while it remained running.

Airport emergency services in general may benefit from reviewing their procedures to ensure that, should the nature of the emergency dictate, they have the capability and processes in place to be able to manage an uncontrolled engine.

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31 Transportation Safety Board – Aviation Occurrence Report – A00Q0133, Runway Excursion, Hydro-Quebec Convair Liner 340 (580) C-GFHH, 27 September 2000.

32 BEA Aircraft Accident Report – Airbus A340-600, Registered F-WWCJ, 15 November 2007, Toulouse Blagnac Airport, France.

1.17 Crew performance

A key aim of crew resource management (CRM) is to minimise or manage crew workload. The additional flight crew that were present on the flight deck during the accident flight were resources available to provide support to the primary flight crew of the captain and first officer. The primary flight crew processed the ECAM messages and followed procedures in accordance with their training. They used other flight crew members at opportune times to share tasks not essential to flight safety and gather information to assist in their decision making. This included communication with the cabin and obtaining information in relation to the damage to the aircraft. The operator's training regime is structured around a prescribed number of flight crew being present in the flight deck. In this case the presence of additional flight crew members did not interfere with the primary flight crew operating in accordance with their experience and training.

CRM advocates a primary flight crew's use of all available resources including people, information and equipment. Therefore, the safe outcome of the accident flight was not only contingent on the primary and supporting flight crew but also on the efforts of the CSM and cabin crew.

Salas and colleagues[33] provided guidance on the factors that make up a competent team from a CRM perspective (Table 7).

Table 7: Identified CRM skills exhibited by a competent team

CRM Skill Identified	Definition
Communication (also known as Closed-loop communication)	Ability of two or more team members to clearly and accurately send and receive information or commands and to provide useful feedback.
Briefing (Mission analysis, Planning)	Ability of team members to develop plans of action by organising team resources, activities and responses to ensure tasks are completed in an integrated and synchronised manner.
Backup behaviour (Advocacy)	Ability of team members to anticipate the needs of others through accurate knowledge about each other's responsibilities, including the ability to shift workload between members to create balance during periods of high workload or pressure.
Mutual performance monitoring (Workload management)	Ability of a team member to accurately monitor other team members' performance, including giving, seeking and receiving task-clarifying feedback.
Team leadership (Management)	Ability of a team leader to direct and coordinate the activities of team members, encourage team members to work together; assess performance; assign tasks; develop team knowledge, skills and abilities; motivate; plan and organise; and establish a positive team atmosphere.
Decision making (judgment, problem solving)	Ability of team members to gather and integrate information, make logical and sound judgments, identify alternatives, consider the consequences of each alternative, and select the best one.

33 Salas, E., Prince, C., Bowers, C., Stout, R., Oser, R.L., & Cannon-Bowers, J.A. (1999). A methodology for enhancing crew resource management training. *Human Factors, 41,* p.163.

CRM Skill Identified	Definition
Task-related assertiveness (confidence, aggressiveness, authoritarian)	Willingness/readiness of team members to communicate their ideas, opinions and observations in a way that is persuasive to other team members and to maintain a position until convinced by the facts that other options are better.
Team adaptability (flexibility)	Ability of team members to alter a course of action or adjust strategies when new information becomes available.
Shared situation awareness (shared mental models, situation assessment)	Ability of team members to gather and use information to develop a common understanding of the task and team environment.

Overall, the flight and cabin crew behaviours on the occurrence flight were consistent with the behaviours identified above, indicating that both the flight crew and cabin crew could be classified in this instance as a team performing to the level of a competent team. The majority of skills defined were exhibited on the flight deck and in the cabin of the aircraft during the accident flight. Examples of these behaviours included:

- The captain (in conjunction with the rest of an experienced flight crew) made critical decisions regarding aircraft controllability, completion of the ECAM procedures, preparing for return and landing of the aircraft and passenger disembarkation.
- The CSM dealt efficiently and effectively with a minor medical issue involving a passenger and their medication, and ensured that all cabin crew were aware of the developing situation and what their duties entailed by personally visiting each station and briefing all crew.
- The CSM reported controlling the communications between the flight deck and cabin to maintain one point of contact.
- Communication between all crew members and between crew and the passengers was rapid, thorough and provided the necessary information to keep all fully informed.
- Flight crew and cabin crew worked very well together within and across their teams to ensure the safe outcome from an emergency situation.

Observation:

On the flight deck, the supporting flight crew provided valuable input and assistance to the primary flight crew—in terms of conducting PA announcements, liaison with the cabin crew and visual observations of damage.

Although the additional flight crew were a valuable resource, had they not been available the primary flight crew would have likely responded to the situation in a similar manner. However, the gathering of information to assist in decision making would have required the use of alternative resources and methods. This may have resulted in prolonging the airborne time before landing or it is also possible that the flight crew may have shed tasks not essential to flight safety. This was unlikely to have affected the safety of the flight because the crew’s training and the aircraft manufacturer’s procedures required them to complete prescribed tasks before attempting to land.

1.18 Aircraft certification

Certification of any product indicates that it complies with a recognised standard. For an aviation product, such as an aircraft or aircraft engine, the recognised standard is a design standard prescribed by a relevant National Airworthiness Authority (NAA).[34] For the Airbus A380 airframe and the Trent 900 engine, the NAA was the European Aviation Safety Agency (EASA).

When a design organisation wishes to produce a new aircraft product, they make application to the relevant NAA for a type certificate. There are basically four phases to the type certification process:[35]

- The standard by which the aircraft product will be assessed is negotiated and established. That agreed standard comprises the applicable airworthiness codes established by the NAA[36] and any special conditions[37] under which the type certification application will be assessed. This standard will not change for that aircraft product, even if the design standard changes after the standard has been agreed upon.
- The NAA and the applicant agree on the certification program. The application is generally limited to a specific time period,[38] during which the design organisation must establish that the proposed aviation product is in compliance with the agreed design standard.
- Compliance with the standard is demonstrated through the completion of the certification program.
- On successful completion of the certification program, a number of documents relating to the results of the certification program as well as continued airworthiness are developed and submitted. Finally, the NAA issues a ‘Type Certificate’. The type certificate is a document which, along with the Type Certificate Data Sheet (TCDS) and referenced documents, defines the aircraft product and any limitations associated with its operation.

At the time of the certification application for the A380 and Trent 900, certification oversight and regulation was being transitioned from the European Joint Aviation Authorities to EASA. With respect to the A380, the Type Certificate was issued on 12 December 2006 and the TCDS noted that the relevant certification standard was Joint Aviation Regulation (JAR) Parts 1 and 25 at change 15.[39]

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34 National Airworthiness Authorities are government statutory authorities in each country that are responsible for the certification of aircraft as part of their duties. For example, in Australia the Civil Aviation Safety Authority is the NAA, in the United States the Federal Aviation Administration (FAA) is the NAA, and so on.

35 The FAA and Industry Guide to Product Certification (2nd ed.) Federal Aviation Administration: 2004. Available at http://www.faa.gov/aircraft/air_cert/design_approvals/media/cpi_guide_ii.pdf.

36 The codes are typically those which are effective on the date of the application.

37 Special conditions are often prescribed for novel or unusual design features, or where the regulations do not contain adequate or appropriate safety standards.

38 For large aircraft, this period is normally 5 years.

39 The TCDS for the A380, A.110, identified the certification application date for the accident aircraft type as 20 Dec 2001 and the certification date as 12 December 2006.

The A380 was also issued with a US FAA Type Certificate on the same date, which noted that the applicable certification standard was US Code of Federal Regulations, Title 14 Aeronautics and Space Part 25 *Airworthiness Standards: Transport Category Airplanes* (14 CFR 25).[40]

1.18.1 Airframe certification – Uncontained Engine Rotor Failure

The certification standard for Uncontained Engine Rotor Failure

The section of the European and US airframe design certification standards for the Airbus A380 that dealt with Uncontained Engine Rotor Failure (UERF) were respectively JAR 25.903 and 14 CFR 25.903(d)(1). Both standards stated:

> (d) *Turbine engine installations.* For turbine engine installations –
>
> > (1) Design precautions must be taken to minimise the hazards to the aeroplane in the event of an engine rotor failure …

Advisory material, AMJ 20-128A and AC 20-128A, both titled *Design Considerations for Minimizing Hazards Caused by Uncontained Turbine Engine and Auxiliary Power Unit Rotor Failure*, was provided by both EASA and the FAA, describing a method of demonstrating compliance with the requirements of JAR 25.903(d)(1) and 14 CFR 25.903(d)(1) respectively. This advisory material included guidance on design precautions aimed at minimising the hazards to an aeroplane resulting from a UERF. The AMJ and AC both stated:

> These guidelines are based on service experience and tests but are not necessarily the only means available to the designer.

AMJ 20-128A was used by the aircraft manufacturer to show compliance of the A380 aircraft design with the requirements of JAR 25.903(d)(1).[41]

The non-containment of significant high-energy debris resulting from a UERF was defined as a 'hazardous effect' under JAR – Engines (JAR-E), Section 2 *Acceptable means of compliance and interpretations,* Advisory Circular - Joint (ACJ) E 510.42. ACJ E 510 required the likelihood of a failure resulting in a 'hazardous effect' to be 'extremely remote', which in turn was to be based on a predicted probability of occurrence of not more than 10^{-8} per hour of engine operation (or one event in every 100,000,000 hours of engine operation).

Although the probability of a UERF event is considered extremely remote, the advisory material was based on the premise that a UERF had occurred. Analysis of the effects of that UERF were then necessary, and additional methods of hazard minimisation were required to reduce the effect of a UERF on the aircraft.

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40 FAA Type Certificate A58NM

41 JAR 25 and AMJ 20-128A have since been superseded by EASA certification specification (CS) 25 and Acceptable Means of Compliance (AMC) 20-128A, respectively. The content of those documents was, for all intents and purposes, the same. JAR 25 and AMJ 20-128A remain applicable to the A380 as its certification basis.

42 For installation on a Transport Category aircraft, JAR-25 required that engines were Type Certificated to JAR-E.

The advisory material provided a description of a safety analysis, the objective of which was to minimise the hazards to the aircraft. The safety analysis includes the use of and assessment of practical design considerations and precautions, and a numerical assessment of the residual risk,[43] based upon a fragmentation model. If the airframe design meets the objectives of the safety analysis, then it has minimised the hazards from a UERF and therefore, meets the certification requirement.

The safety analysis accepted that after a UERF event:

> Some degradation of the flight characteristics of the aeroplane or operation of a system is permissible, provided the aeroplane is capable of continued safe flight and landing.[44]

The practical design considerations included the placement of critical components, where practicable, outside of the areas subject to UERF debris. Where this is not practical, the designers were to reduce the risks by practices such as system multiplication, redundant multipath wiring and the use of shielding from the airframe or supplemental shielding. Specific accepted design precautions included:

- Fire warning and extinguishing systems. Following a UERF event, the fire warning systems of all engines should continue to be operational and the extinguishing systems of the remaining engines should also remain operational.
- Flammable fluid shut-off valve. The guidance identified that the only effective means of extinguishing an uncontrolled engine fire following a UERF event may be to shut off the supply of all flammable fluid to that engine. Therefore, the operation of the isolation valves for fuel and hydraulic fluids should be assured. This required the designer to locate these isolation valves outside of the UERF impact areas for an engine, and ensure that any valve actuation controls that were routed through the impact areas were redundant and appropriately separated.

The numerical assessment of the residual risk examined a number of fragmentation scenarios. Associated with each scenario was a specific hazard ratio that identified the probability that the scenario in question could result in catastrophe.[45] The assessment required the designer to plot fragment trajectories from each of these scenarios, calculate the probability of a catastrophic outcome, and show that this probability was less than the specified ratio.

For other than the duplicated or multiplicated systems, that were contained within a specific debris impact area, the safety analysis was predicated upon assessment of the trajectory of a single one-third disc fragment, a single intermediate fragment and

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43 Residual risk is the risk remaining after all practical design considerations have been taken to minimise the hazards from the effects of a UERF.

44 AMJ 20-128A defined ‘continued safe flight and landing’ as: ‘the aircraft is capable of continued controlled flight and landing, possibly using emergency procedures and without exceptional pilot skill or strength with conditions of considerably increased flight crew workload and degraded flight characteristics of the aeroplane.’

45 Defined under JAR 25.1309 and associated guidance as a failure condition that would result in multiple fatalities, usually with the loss of the aeroplane.

a limited percentage of small fragments.[46] Each of the fragment sizes had a specific spread angle that defines their relevant impact areas. The spread angle was the angle measured, fore and aft, from the plane of rotation of the failed rotating disc. Each of the scenario assessments was conducted separately.

Evolution of the Acceptable Means of Compliance

The guidance material, providing an acceptable means of compliance, reflected the evolving knowledge base gained from service experience. This included several aircraft accident investigations conducted by the US National Transportation Safety Board (NTSB).

On 22 September 1981 a McDonnell Douglas DC-10 sustained structural and systems damage following a UERF event during the take-off roll at Miami International Airport, Florida. The flight crew rejected the takeoff and safely stopped the aircraft. There were 15 crew and 71 passengers on board and there were no injuries.[47]

As a result of this accident, on 16 April 1982, the NTSB issued recommendation A-82-038 to the US Federal Aviation Administration (FAA), which stated:

> The NTSB recommends that the Federal Aviation Administration: expedite the publication of guidance material for acceptable means of compliance with 14 CFR 25.903(d)(1), which includes compliance documentation by failure mode and effect analysis, provides for rotor fragment energy levels and paths based on cases of severe in-service damage, and reflects advances in analytical techniques and concepts which have taken place since certification programs of the early 1970's.

In response to this recommendation, on 9 March 1988 the FAA issued Advisory Circular (AC) 20-128 *Design Considerations for Minimizing Hazards caused by Uncontained Turbine Engine and Auxiliary Power Unit Rotor and Fan Blade Failures*. The NTSB closed the recommendation on 24 May 1988, stating that:

> The Safety Board has accepted the FAA’s Advisory Circular (AC) 20-128 as fulfilling the technical aspects of A-82-38. This safety recommendation has been classified as "Closed--Acceptable Action."

On 19 July 1989 a UERF event occurred on a McDonnell Douglas DC-10 during cruise, which resulted in the loss of all hydraulic systems that powered the aircraft’s flight controls. The flight crew experienced severe difficulties controlling the aircraft, which subsequently crashed during an attempted landing at Sioux Gateway Airport, Iowa. There were 11 crewmembers and 285 passengers on board. One flight attendant and 110 passengers were fatally injured. [48]

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[46] AMJ 20-128A permitted the use of an alternative model that examined a single one-third fragment in place of the one-third disc fragment and intermediate fragment scenarios, but over a greater impact area.

[47] National Transportation Safety Board. Aircraft accident report: NTSB/AAR-82-3, Air Florida Airlines, Inc. McDonnell Douglas, Inc., DC-10-30CF, N101TV, Miami International Airport, Miami, Florida, September 22, 1981.

[48] National Transportation Safety Board. Aircraft accident report: NTSB/AAR-90/06, United Airlines Flight 232, McDonnell Douglas DC-10-10, Sioux Gateway Airport, Sioux City, Iowa, July 19, 1989.

During its investigation of the accident the NTSB issued a series of safety recommendations that were targeted at uncontained engine failures. One of those recommendations concerned improving the advisory material concerning the minimisation of hazards resulting from UERFs.

Recommendation A-90-170 stated:

> The NTSB recommends that the Federal Aviation Administration: analyze the dispersion pattern, fragment size and energy level of released engine rotating parts from the July 19, 1989, Sioux City, Iowa, DC-10 accident and include the results of this analysis, and any other peripheral data available, in a revision of AC 20-128 for future aircraft certification.

In response to this recommendation, on 25 March 1997 the FAA issued AC 20-128A *Design Considerations for Minimizing Hazards caused by Uncontained Turbine Engine and Auxiliary Power Unit Rotor Failure*. This AC was harmonised with the guidance provided by the European Joint Aviation Authorities and cancelled AC 20-128. On 26 August 1997, the NTSB classified A-90-170 as ‘closed--acceptable action’, based on the FAA's actions.

Service experience relating to UERFs

Another recommendation from the Sioux Gateway Airport DC-10 accident investigation, A-90-172, stated:

> The NTSB recommends that the Federal Aviation Administration: create the mechanism to support a historical data base of worldwide engine rotary part failures to facilitate design assessments and comparative safety analysis during certification reviews and other FAA research.

In response to this recommendation, on 10 September 1998 the FAA advised the NTSB that:

> The FAA will preserve & retain the overall data base for uncontained engine failures generated under the aircraft catastrophic failure prevention program. The data will be available for future design assessments, safety analyses, & research.

On 8 December 1998 the NTSB classified A-90-172 as ‘closed--acceptable action’, based on the FAA's actions.

In addition the Aerospace Industries Association conducted a study of uncontained rotor failures and their consequences over the period 1969 to 2006. The study’s findings were released in a report in January 2010.[49] The intent of the report was to provide ‘a single rotor burst database for high bypass turbofans, as recommended by the NTSB recommendation A-90-172’.

The report found that there were 67 recorded disc burst events in the period studied, of which 58 were uncontained by the engine nacelle. A breakdown of those failures

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49 Aerospace Industries Association (AIA), *AIA report on high bypass ratio turbine engine uncontained rotor events and small fragment threat characterization 1969 - 2006, Volume 1*, January 2010. Available at: http://www.aia-aerospace.org/assets/aia_rotor_burst_small_fragment_committee_report_vol1.pdf

indicated that 46 involved first generation[50] engines and 12 second generation engines. The second generation engines were 'those designed in the 1980s with the understanding and incorporation of Lessons Learned from the first generation.'[51]

The failure of the No. 2 engine on VH-OQA was the first UERF involving a third generation engine. Third generation engines were those 'designed to incorporate the Lessons Learned from the second generation.'[52]

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50 ibid. pg 21. 'First generation' Turbofan engines that were designed in the late 1960s. These include: the JT9D, RB211-22B, and the CF6-6, CF6-50 and CF34-3.

51 ibid. pg 21. 'Second generation' turbofan engines include: PW2000, CFM56-2, CFM56-3 and CFM56-5, and RB211-5XX.

52 Ibid. pg 21. 'Third generation' turbofan engines include: GE90, CFM56-7, CF34-10, PW4000 and RB211 Trent series.

2 FACTUAL INFORMATION: ENGINE

Part 2 of this report focuses on the aircraft's No. 2 engine that sustained an uncontained failure on 4 November 2010. Information is provided on the structure of the engine and the damage resulting from the uncontained failure that will provide a background for understanding the engine failure sequence as described in Part 3.

2.1 Trent 900 description

2.1.1 Mechanical arrangement

The Trent 900 is a three-shaft, high-bypass ratio turbofan engine with variants ranging in maximum thrust from 334.3 kN (75,152 lb) to 374.1 kN (84,098 lb). The three primary rotating assemblies in the Trent 900 are (Figure 25):

- a low pressure (LP) compressor (fan) connected by a shaft to a five-stage LP turbine
- an intermediate pressure (IP) compressor connected by a shaft to a single stage IP turbine
- a high pressure (HP) compressor driven by a single-stage HP turbine.

Each rotating assembly was supported by bearings at the front and rear of each shaft.

The No. 2 engine on VH-OQA was a Rolls-Royce Trent 972-84 variant, with the engine serial number (ESN) 91045. The engine's rated maximum take-off thrust was 341.4 kN (76,752 lb).

Figure 25: Trent 900 main rotating assemblies

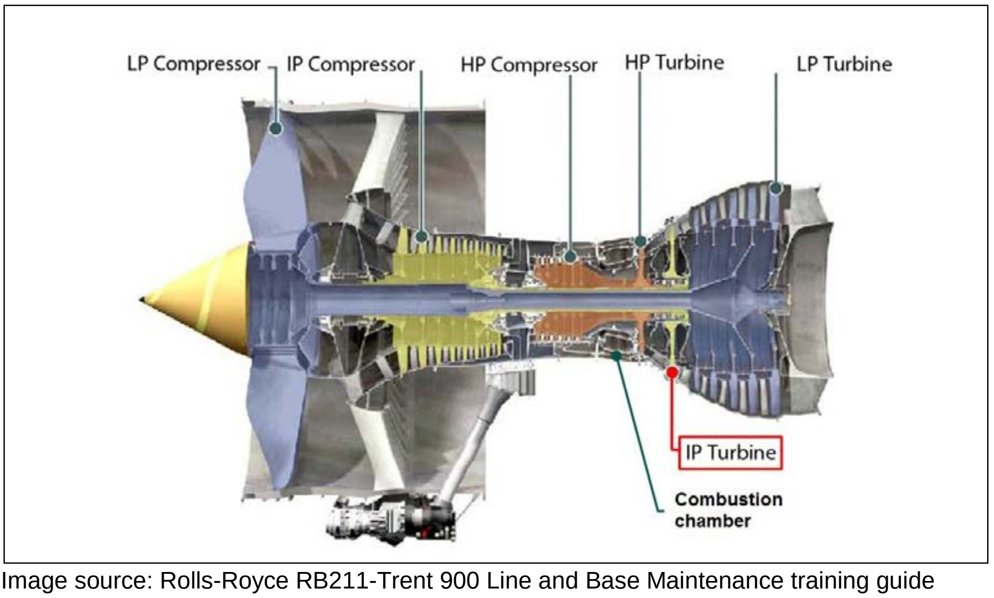

Image source: Rolls-Royce RB211-Trent 900 Line and Base Maintenance training guide

The Trent 900 is of modular construction and includes a separate module for the IP turbine, referred to as module 05 (Figure 26).

Figure 26: Trent 900 modular breakdown (IP turbine highlighted in red)

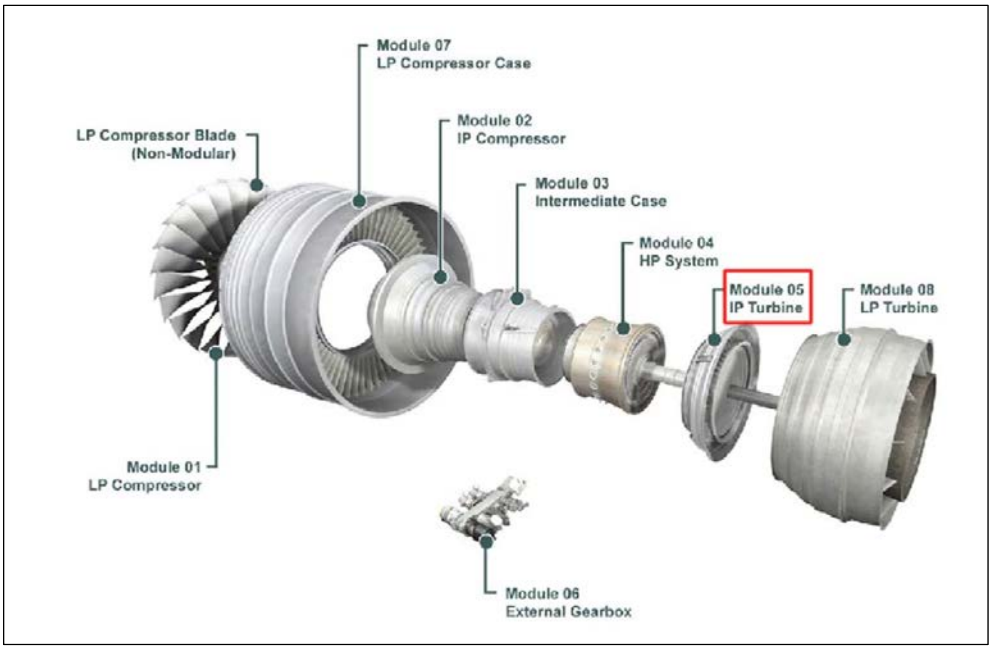

Image source: Rolls-Royce RB211-Trent 900 Line and Base Maintenance training guide

The IP turbine module consists of the IP shaft, IP turbine disc and blades, IP nozzle guide vanes (NGVs),[53] LP stage 1 NGVs, IP turbine case, LP turbine front panel, HP and IP bearings, and the HP/IP bearing support structure (Figure 27).

Figure 27: IP turbine module

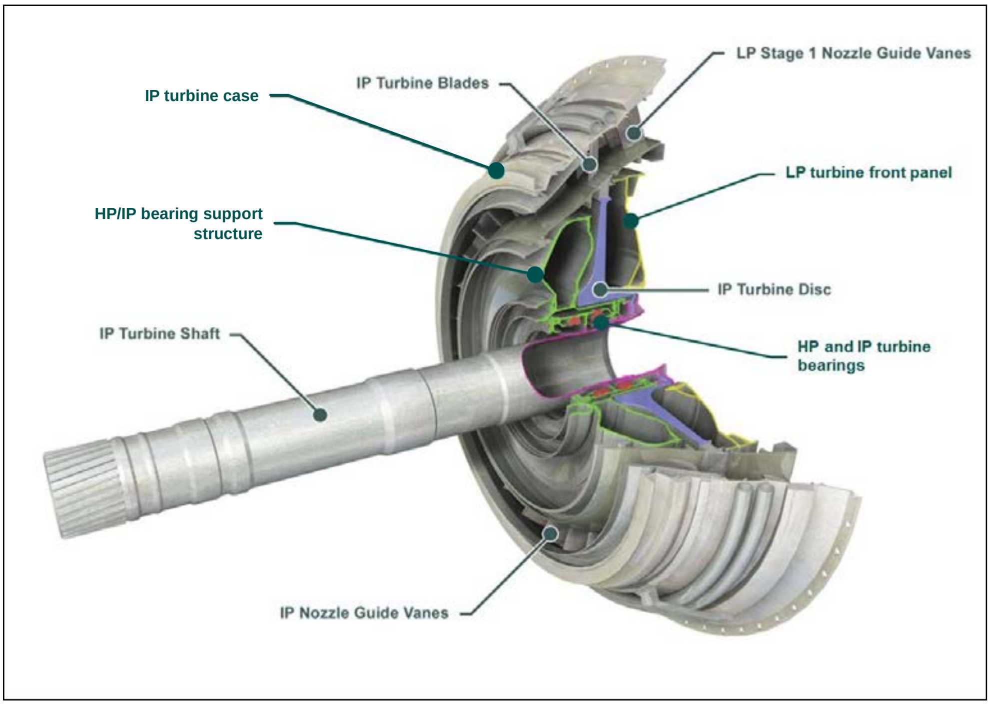

Image source: Rolls-Royce RB211-Trent 900 Line and Base Maintenance training guide

53 Nozzle guide vanes are fixed structures that direct the airflow into the turbine at the correct angle.

The HP/IP bearing support structure provides radial and axial location support for the HP and IP bearings. The bearings are located in a hub structure that encloses an oil fed bearing chamber. That hub is connected to the IP turbine case through a strutted ring structure that has a front and rear panel. The support struts pass through the IP turbine NGVs.

Spaces within the IP turbine module are supplied with compressed air from various engine compression stages to provide cooling or sealing functions (see the different coloured regions in Figure 28). For example, the HP stage 3 (HP3) air[54] in the space around the IP turbine disc (coloured pink in Figure 28) provides cooling for the IP turbine disc. The air/oil mix that is contained within the HP/IP bearing chamber (coloured light green) is separated from the relatively hot HP3 air by a buffer space (coloured dark yellow). The buffer space surrounding the HP/IP bearing chamber is supplied with cooler IP stage 8 (IP8)[55] air and under normal circumstances is free from oil (Figure 28).

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54 Bleed air taken from the third stage of the HP compressor.

55 Bleed air taken from the eighth stage of the IP compressor.

Figure 28: Side view through the Trent 900 (regions of differing air pressure and temperature indicated by differing colours)

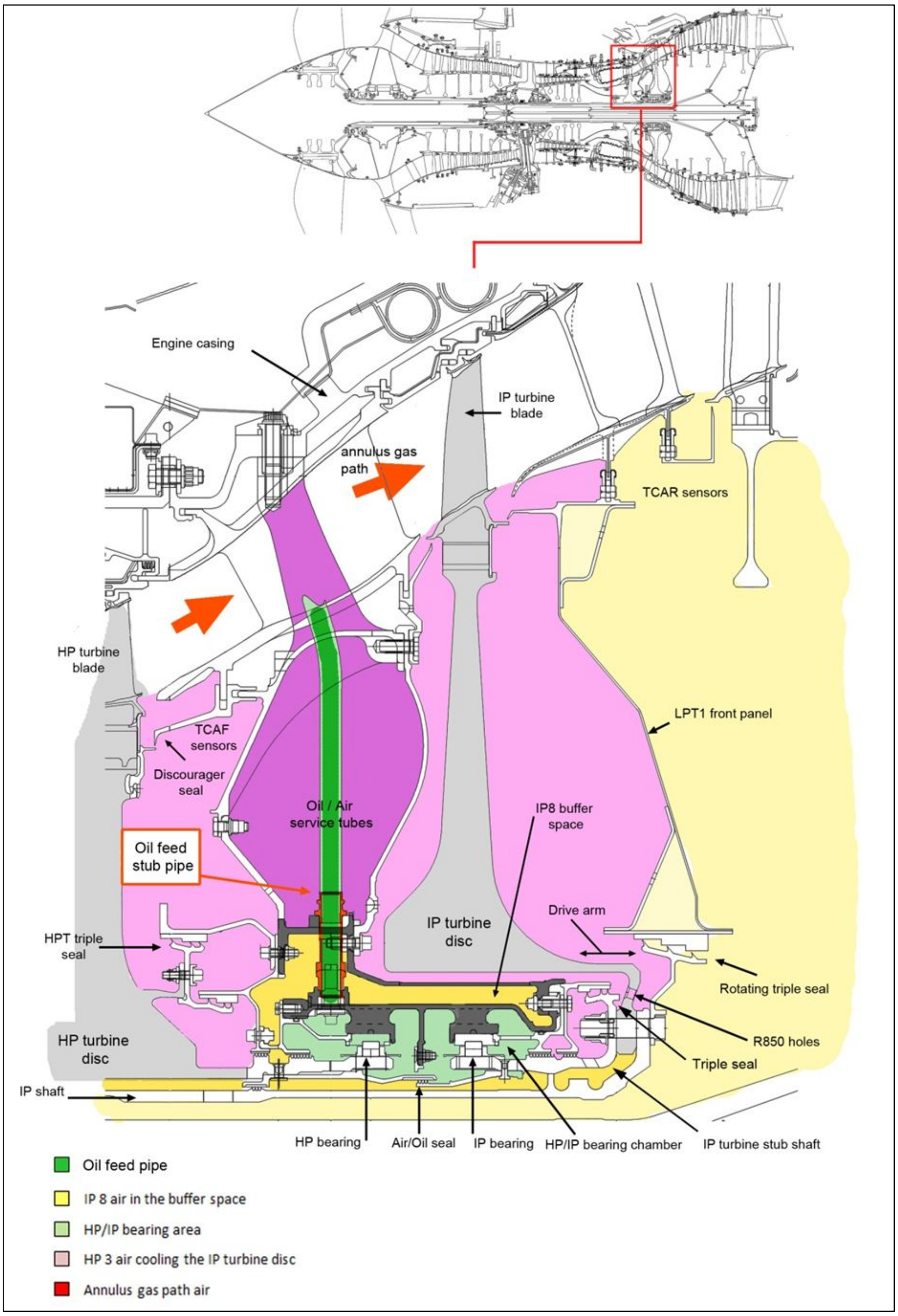
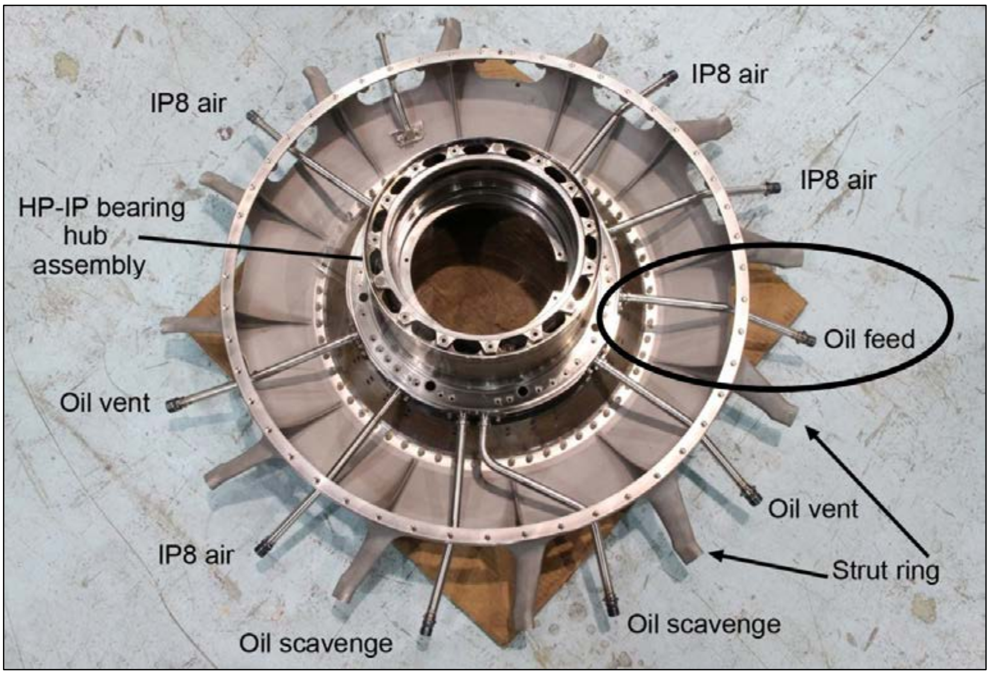


Image modified from a Rolls-Royce plc supplied model

A number of pipes pass through the HP/IP structure to service the bearing chamber. These service pipes pass through the inside of the IP turbine NGVs to variously: supply IP8 air to the buffer space, vent air from the bearing chamber, supply oil to the bearing chamber and scavenge oil from the chamber (Figure 29).

Figure 29: Bearing chamber service pipes in HP/IP support structure



Note: The above view is from the rear of the HP/IP support structure and the rear panel of the structure has been removed for clarity.

The oil feed, scavenge and vent pipes pass through clearance holes in the outer hub, then through the buffer space and were secured to the inner hub by an interference fit[56] and end welded (Figure 30). These pipes differ from the IP8 air pipes as they have an extension, referred to as a stub pipe, to account for the additional length required to reach the inner hub. An integral filter in the oil feed stub pipe was included with the intent of preventing oil contaminants entering the HP/IP bearings.

To accommodate the filter in the oil feed stub pipe, the inner hub end of that pipe has an enlarged inside diameter (Figure 31).

[56] An interference fit is where the outside of the pipe is slightly larger than the inside of the hole. The pipe is forced into the hole during manufacture and the associated friction assists in retaining the pipe in place.

Figure 30: Cross section of a generic HP/IP hub with a service pipe

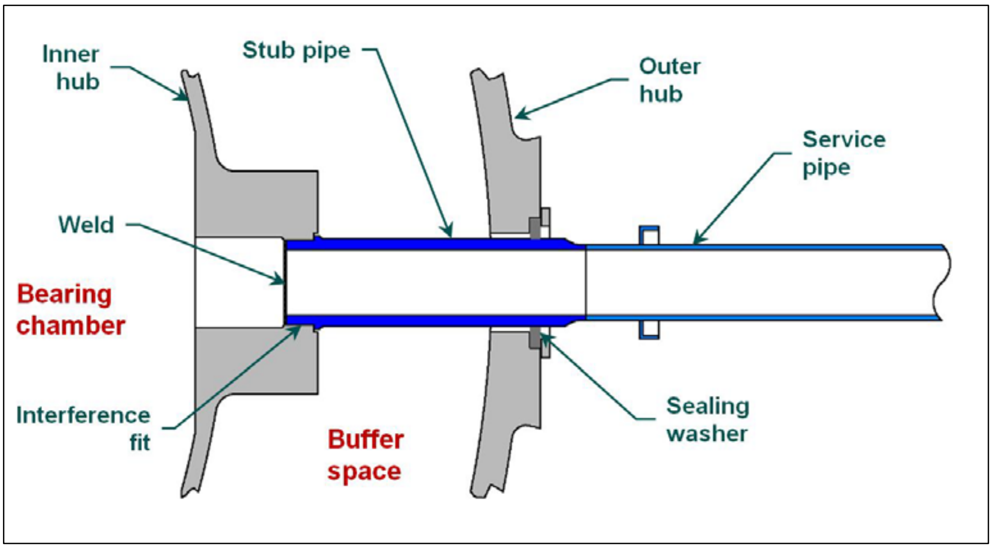
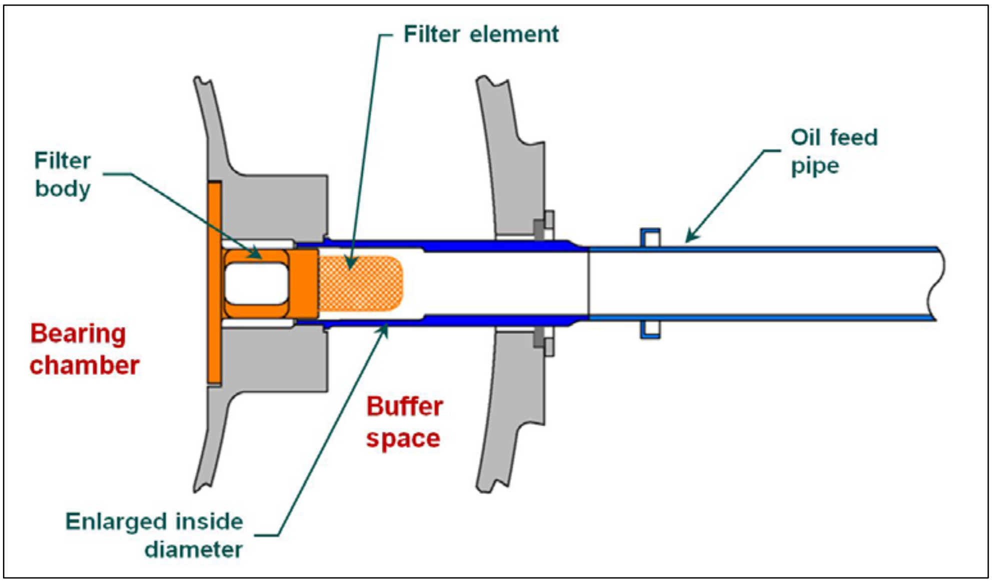


Figure 31: Cross section of a generic HP/IP hub with an oil feed stub pipe



The IP turbine disc was constructed from a high-strength, heat-resistant nickel alloy and was designed to transmit the loads generated by the IP turbine blades to the IP drive shaft. The disc was connected to the IP drive shaft through an extension to the disc, referred to as the 'drive arm' (Figure 28). The drive arm had a number of holes to allow the HP3 cooling air to flow from the front to the back of the disc. Those holes (R850) were located close to the bolted connection to the IP shaft, immediately behind the triple seal at the back of the bearing chamber.

2.1.2 Engine control

The Trent 900 is controlled using a full authority digital engine control (FADEC) that controls the thrust level of the engine as demanded by the aircraft's control systems. The FADEC consisted of an engine electronic controller (EEC), and a number of sensors and controller systems. Based on the inputs from a number of engine and aircraft sensors, the EEC computes and commands a number of controller systems on the engine.

The FADEC performs a number of functions, including:

- control of engine start
- fuel and airflow control in response to thrust demand and the environmental conditions
- scheduling of thrust levels as required by the aircraft systems
- engine protection
- recovery from engine surge
- the provision of information to the aircraft for control, recording and display purposes.

2.1.3 Engine protection functions

The FADEC provides a number of engine protection functions. Of these, the ATSB was particularly interested in the engine overspeed protections and turbine overheat detection. The following discussion examines those functions.

Overspeed protection

The engine's three primary rotating assemblies are not directly connected to each other and operate at different rotational speeds. The speed of each rotating assembly is denoted by N1, N2 and N3 for the LP, IP and HP assemblies respectively. These speeds are represented as a percentage, with 100% representing a defined speed datum value set during the engine design. Each assembly's maximum certified speed (referred to as the *redline speed*) for normal operation has been approved for each of the rotating assemblies, based upon the results of testing.

The engine's rotating assemblies were designed and tested for in-service operational conditions and to meet the certification standards. The rotating assemblies were designed to operate at rotational speeds above the redline speed for very short periods without the forces generated by rotation exceeding the structural capability of the rotating assemblies.[57] Operation at speeds above the redline speed is termed an *overspeed*.

An overspeed that exceeds the structural capability of a rotating assembly can create a hazardous situation in which the rotational forces exceed the strength of the structure, leading to its fracture. If any high-energy debris cannot be contained within the affected engine case, a hazardous event can occur as a result of that ejected debris further damaging the aircraft.

The Trent 900 FADEC provided a number of overspeed protections to minimise the risk of such an event.

The LP overspeed protection system provides two protection functions, against LP rotor assembly overspeed and LP turbine overspeed. LP rotor assembly speed, or N1, is measured at the front and rear ends of the LP shaft. If the speed sensors

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[57] Forces are generated within a rotating object, the higher the rotational speed, the greater the forces generated.

detect an overspeed event, the FADEC commands the fuel flow to the combustion chamber to stop. Combustion ceases and N1 reduces.

If the LP shaft fails, there would be a difference between the speed at the front (compressor end) and rear (turbine end) of the affected shaft. A shaft failure could result in a turbine overspeed because the core airstream continues to apply a load to rotate the turbine, but the load would not be balanced by the load required to turn the LP compressor. To protect the LP turbine from such an event, the FADEC monitors the difference in the speed between the front and rear LP shaft speed sensors. If the FADEC detects a difference between these sensors, the LP turbine overspeed system (LPTOS) activates and shuts off the fuel flow to the combustion chamber. Combustion stops and any overspeed of the unrestrained LP turbine would be limited.

The IP rotor assembly overspeed protection system has speed sensors at the compressor end of the IP turbine shaft. It operates in a similar manner to the LP rotor assembly overspeed protection system, but does not provide for the turbine overspeed protection function in the event of an IP shaft failure. During the design of the engine, it was understood that this protection was not required because the expected engine behaviour would prevent a hazardous overspeed of the IP turbine in the event of a shaft failure. Further detail of this engine behaviour is provided in section 2.4 of this report.

There is no specific HP overspeed protection function to interrupt the fuel supply in a similar manner to the LP and IP systems. The HP shaft system was designed and certified on the basis that failures of the shaft would not lead to a hazardous overspeed.

Turbine overheat detection

The FADEC's turbine overheat detection function monitors the temperature around the IP turbine. The system consists of two temperature sensors located in the spaces to the front of the HP/IP support structure (TCAF)[58] and to the rear of the LP turbine front panel (TCAR)[59] (Figure 28). If either of these sensors detect a temperature in excess of the predetermined overheat limit, the FADEC sends a message to the relevant aircraft system for it to display a 'turbine overheat' warning on the engine/warning display in the cockpit. No change is made to an engine's operation by the FADEC as a result of the identification of a turbine overheat. Any subsequent precautionary actions are the responsibility of the flight crew.

2.1.4 Engine surge

The compressor section of a jet engine increases the pressure of the air delivered to the combustion chamber. The compressors in the Trent 900 are made up of a number of stages of rotating and static aerofoils (or blades and vanes respectively). Steady flow through the stages of a compressor occurs within a relatively narrow band of conditions. If the conditions inside a compressor go outside of this band due to an operating condition or a disturbance, the flow around the blades can break

58 Turbine cooling air front.

59 Turbine cooling air rear.

down in a manner known as a *stall*. In this instance, the blades would no longer effectively compress the air.

A stall could be a local phenomenon, where a small region of stalled flow rotates around the compressor (a rotating stall), or could affect the entire compressor (a locked stall). If the breakdown of flow in a compressor stall is significant enough, the pressure change within the engine could be sufficient to reverse the flow through the compressor in a phenomenon known as a 'surge'. A surge is characterised by a rapid drop in HP compressor delivery pressure (in the Trent 900 engine this is denoted as P30) and is often associated with a loud bang, or series of bangs, that can be heard in the aircraft. Sometimes a flame can be observed at the core engine inlet and exhaust as combustion products are expelled from the combustion chamber. If action is not taken immediately, the surge can cycle between normal and surge flow, potentially resulting in damage to the engine and loss of engine thrust.

In the normal operation of a modern FADEC-controlled engine, surges are not common. The Trent 900 FADEC includes a surge recovery function that includes opening the bleed air valves and reducing the fuel flow.[60]

2.2 No. 2 engine on VH-OQA

Engine serial number (ESN) 91045 was manufactured in the United Kingdom in June 2006 and initially fitted to the No. 4 position on VH-OQA at the Airbus factory in France. The aircraft was subsequently released into service on 18 September 2008.

The aircraft's maintenance documentation was reviewed in relation to the in-service operation of ESN 91045, including its oil usage. This review established that the engine had similar engine operation and oil consumption characteristics compared to the other engines fitted to the aircraft and throughout the operator's Trent 900 powered fleet.

2.2.1 Significant service history

In September 2009, after 3,418 hours and 416 cycles of operation, metal debris was detected in the oil of ESN 91045. The engine was removed from the aircraft and sent to a manufacturer-approved service facility for examination.

The metal in the oil was found to have originated from a bearing in the LP system. The bearing was replaced and, while the engine was at the service facility, a number of other servicing items were addressed. This included the replacement of the HP and IP turbine bearings, inspection of the IP turbine disc, replacement of the HP turbine disc and inspection of the HP/IP bearing support assemblies. The only defects detected during the inspections were excessive wear in a number of the HP/IP bearing chamber service pipes. That wear was at the point where the fittings exit the IP turbine case, and was rectified by replacement of the pipe ends via a manufacturer-approved repair scheme.

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60 The primary purpose of the reduced fuel flow is to move the compressor operating conditions away from stall, restore stable compressor airflow and therefore prevent over-temperature of the gases in the combustion chamber. Any over-temperature of the combustion chamber gases could damage the turbine blades.

The replacement of the bearings and other work was the last significant service activity on the engine prior to the uncontained failure.

Following the rectification work, the engine remained in storage until 24 February 2010 when it was fitted to the No. 2 position on VH-OQA. The engine remained in service in this position until the uncontained engine failure. At the time of the failure, the engine had accumulated a total of 6,314 hours in service and 677 cycles.

Observation:

The investigation did not identify any link between the maintenance performed on the engine and the uncontained engine failure.

2.3 Damage to the engine

The primary damage to the engine occurred in the region of the IP turbine. An initial 'on-wing' visual examination of the engine identified that the entire circumference of the case around the IP turbine was breached and that the IP turbine disc was missing (Figure 32). The thrust reverser cowl and cold airstream[61] duct fairings sustained significant damage and the section of the thrust reverser cowl and cold airstream fairings to the rear of the primary damage had separated from the engine.

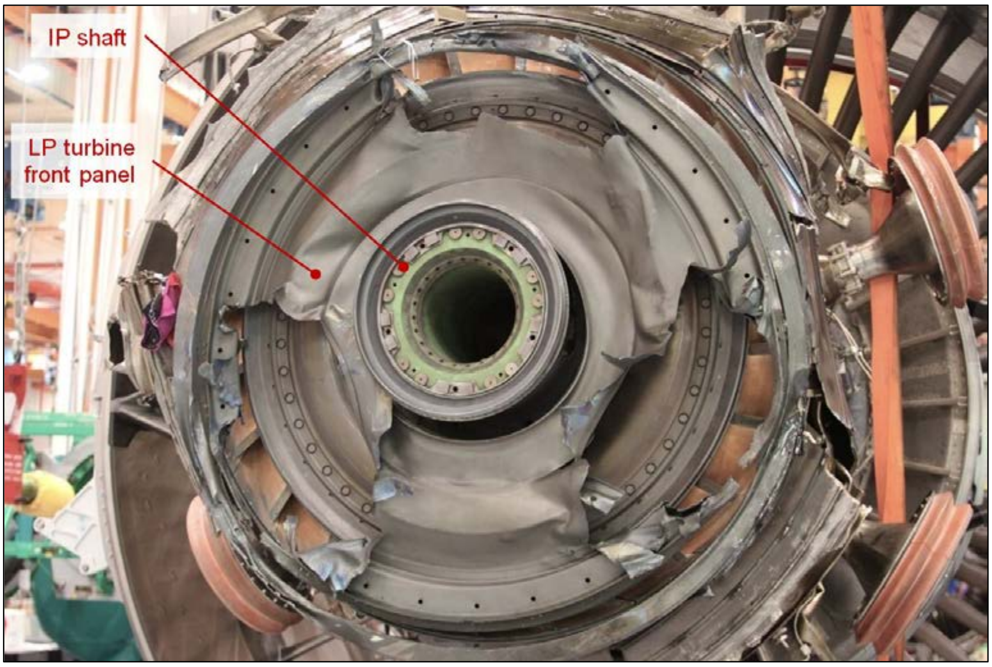
Figure 32: General damage to the engine (looking outboard)

The engine was disassembled in a manufacturer approved service facility under the supervision of the ATSB.

[61] There are two primary air streams through the engine: the 'cold' and 'hot airstreams'. The air that passes through the fan only, and is ducted around the core of the engine is termed the cold airstream. The hot or core airstream passes through the fan and engine core, which consists of the intermediate and high pressure systems, the combustion chamber and the low pressure turbine.

Further damage to the engine was identified during that disassembly including three large perforations to the LP turbine front panel that were approximately equally distributed around the circumference of the panel (Figure 33). The first stage LP turbine blades, in three equally distributed locations, were also damaged.

Figure 33: LP turbine front panel damage



Note: The above view is from the rear of the engine, looking forward.

Damage was also identified between the HP/IP bearing support structure and the HP turbine. This included scoring of the HP turbine disc and significant damage to the triple seal. The rotating and static portions of the HP turbine triple seal had separated from their support structures (Figure 34), fractured and then entwined within each other (Figure 35). There was a rough, grey deposit on the surfaces within module 05 near the HP turbine shaft that was identified as melted and resolidified metal.

Figure 34: Location of the HP turbine triple seal

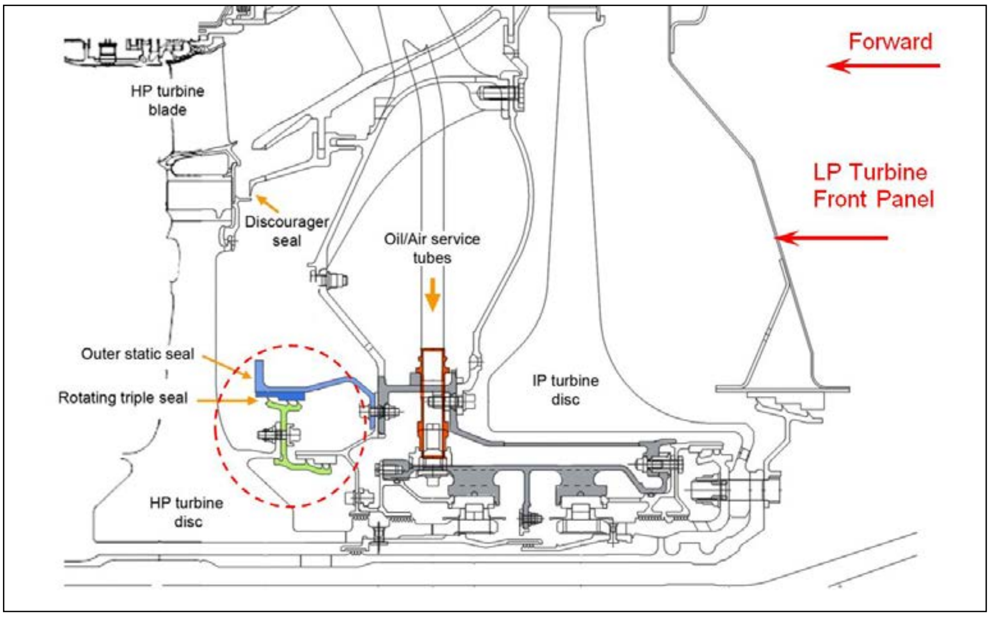
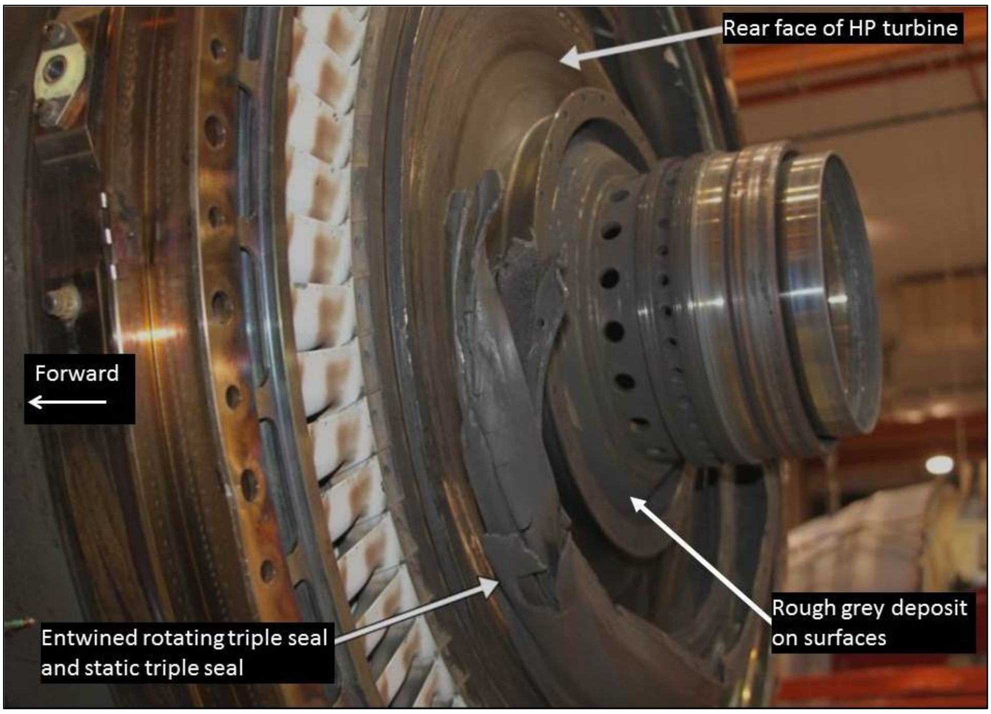


Image modified from a Rolls-Royce plc supplied model

Figure 35: Damage between the HP/IP structure and HP turbine



Examination of the IP shaft found that a remnant of the IP turbine drive arm bolting flange was still attached to the drive shaft (Figure 36). The IP turbine drive arm had separated from the bolting flange at the R850 cooling holes. There was a significant amount of heat damage and erosion in and around these cooling holes and the surfaces were coated in rough melted/resolidified material. Although much of the surface was damaged by metal-to-metal contact, sections of the original fracture surface were examined and found to be consistent with tensile and shear overstress.

The HP/IP bearing chamber triple seal and the rear rotating triple seal between the IP and LP turbines showed a significant amount of erosion and heat damage (Figure 37). The flange of the rear rotating triple seal also contained erosion pits immediately behind each of the R850 cooling holes.

Figure 36: Damage to IP drive shaft

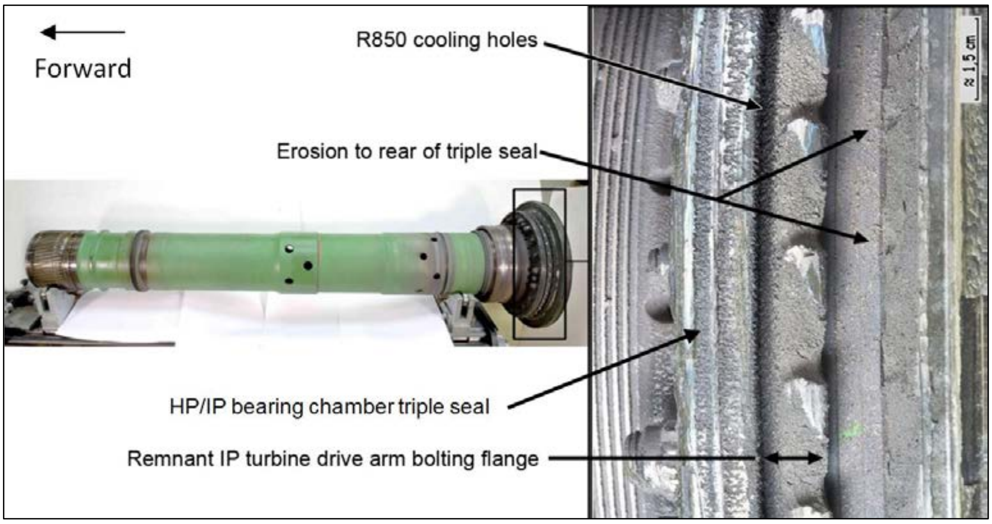


Figure 37: Location of the triple seals

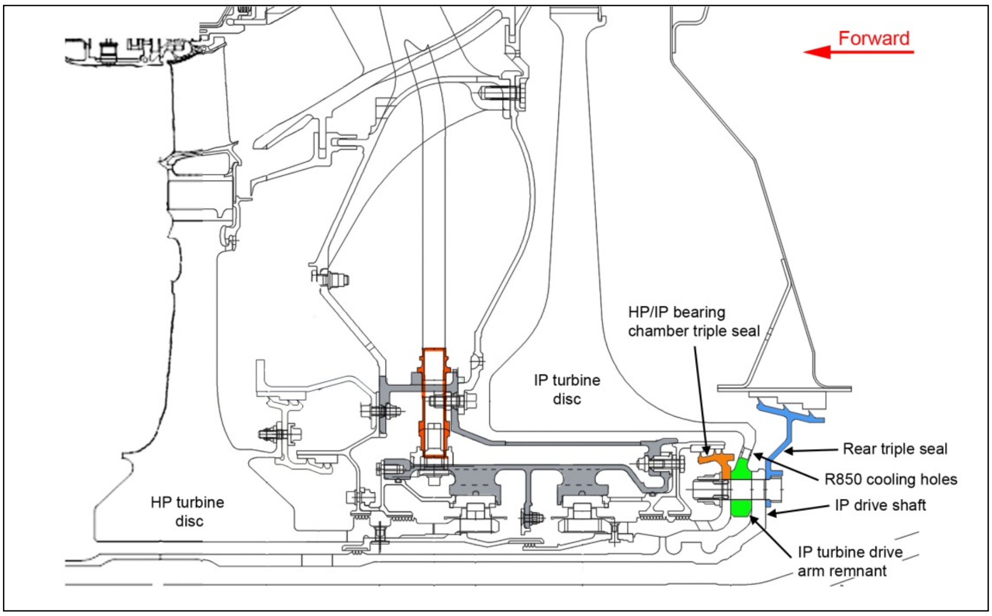
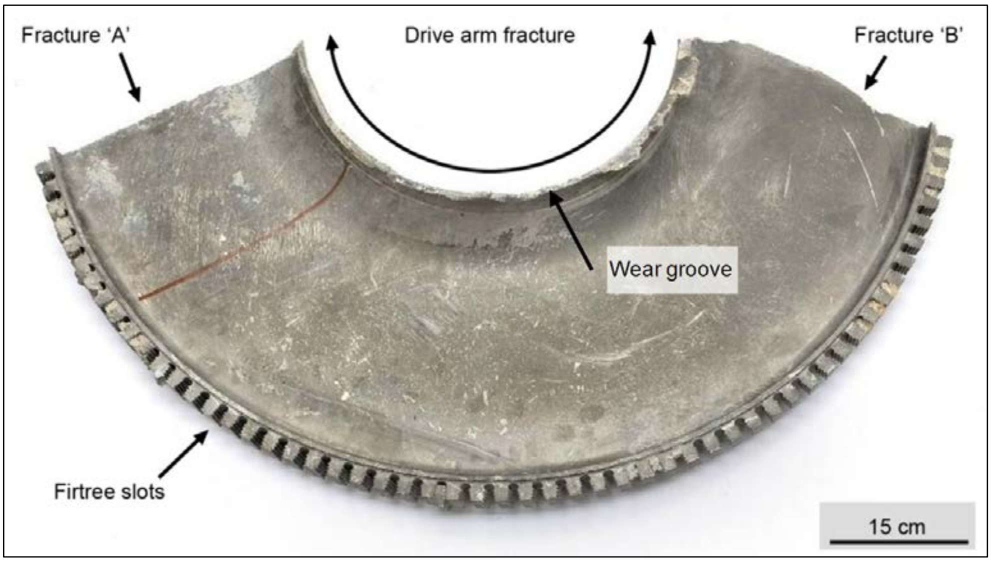


Image modified from a Rolls-Royce plc supplied model

The section of the IP turbine disc that was recovered from Batam Island, Indonesia (Figure 12) was examined in detail by the engine manufacturer under the supervision of the ATSB. This section of disc consisted of about 43% of the total disc and weighed about 70 kg. In addition, two radial fractures in the disc and a circumferential fracture in the drive arm were identified (Figure 38). The examination found that all of the fractures were consistent with an overstress of the material. There was no evidence of any pre-existing material defect or damage that contributed to the fractures.

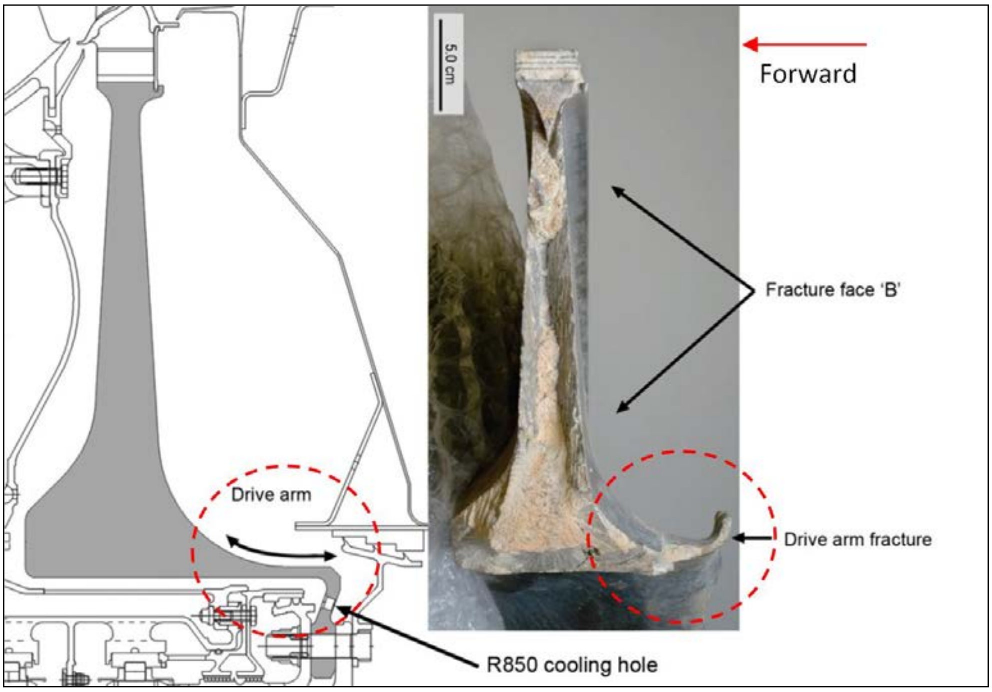
The rear face of the disc contained a circumferential wear groove in the radius where the disc transitions to the drive arm. This wear groove was consistent in size and location with the forward flange on the LP turbine front panel.

Figure 38: Recovered section of IP turbine disc – looking forward (radial fractures labelled 'A' and 'B')



The drive arm was bent outward by approximately 90° (Figure 39). Metallurgical examination of the drive arm found changes in the material that were consistent with its exposure to high temperatures (in excess of 1,125 °C). The portion of the drive arm remaining on the disc did not contain any remnants of the R850 cooling holes. The separated portion of the drive arm, which would likely have contained those remnants, was not recovered.

Figure 39: Comparison of a diagrammatic representation of the IP turbine disc with the recovered segment of the disc

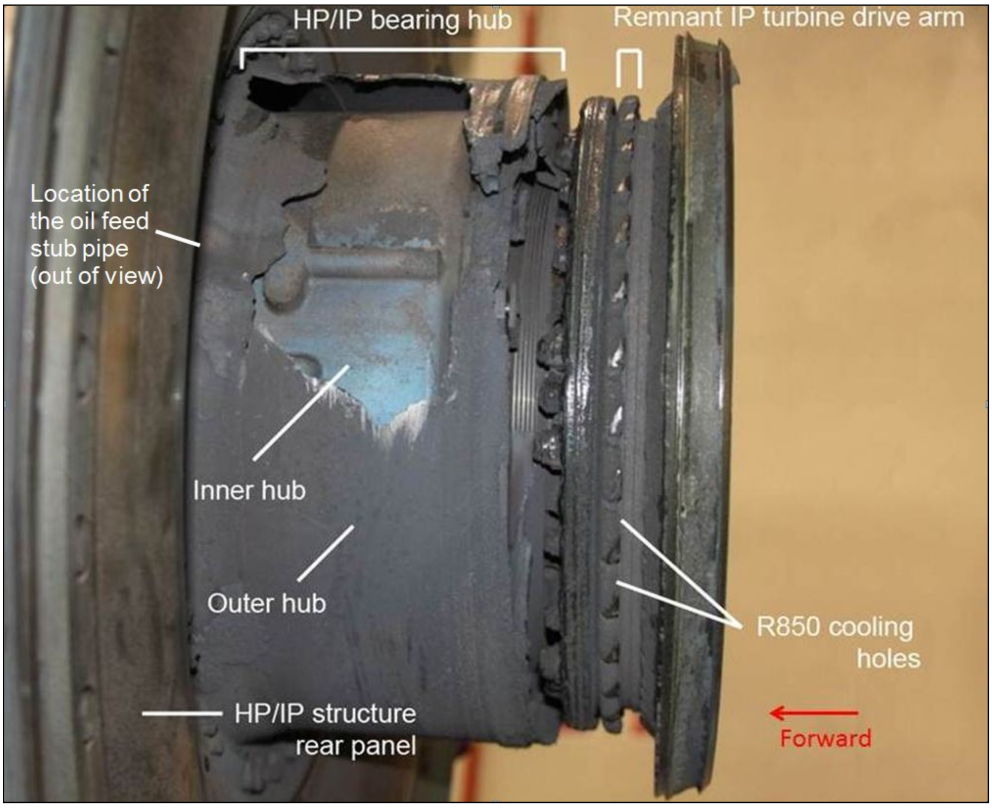


The disc segment exhibited a significant amount of overall deformation. There was an average radial growth of about 19 mm and the blade restraint (firtree) slots had opened by 0.3 to 0.4 mm when compared to the manufacturer's specifications.

There were no complete turbine blades in the recovered section of the disc. However, a number of blade root remnants were recovered from that disc section, as well as from Batam Island and various locations in the aircraft structure. No evidence of any pre-existing defects or damage was identified on any of these blade root remnants. All damage was consistent with having resulted from the uncontained failure.

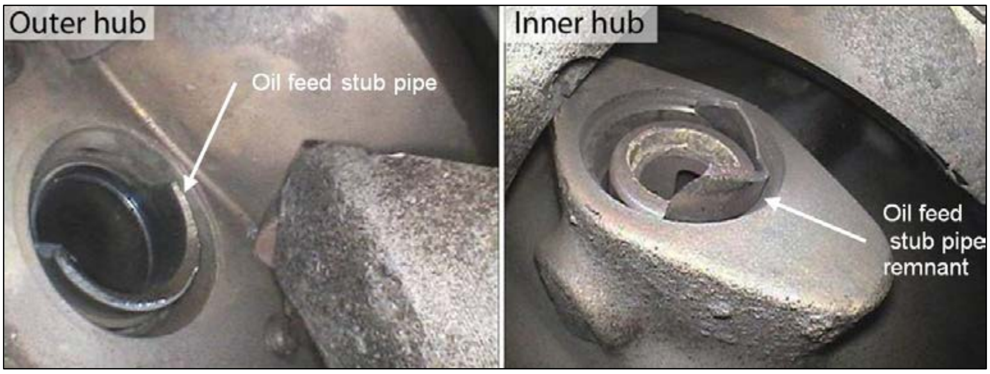
The HP/IP bearing hub and structure exhibited a significant breach of the outer bearing hub (Figure 40), which was also associated with a bulging of the area. Portions of the hub had a coating of the rough melted/resolidified material observed in other areas. The outer hub was breached in the upper-right quadrant, extending down to the vicinity of the oil feed stub pipe (not visible in Figure 40).

Figure 40: Damage to the HP/IP hub structure



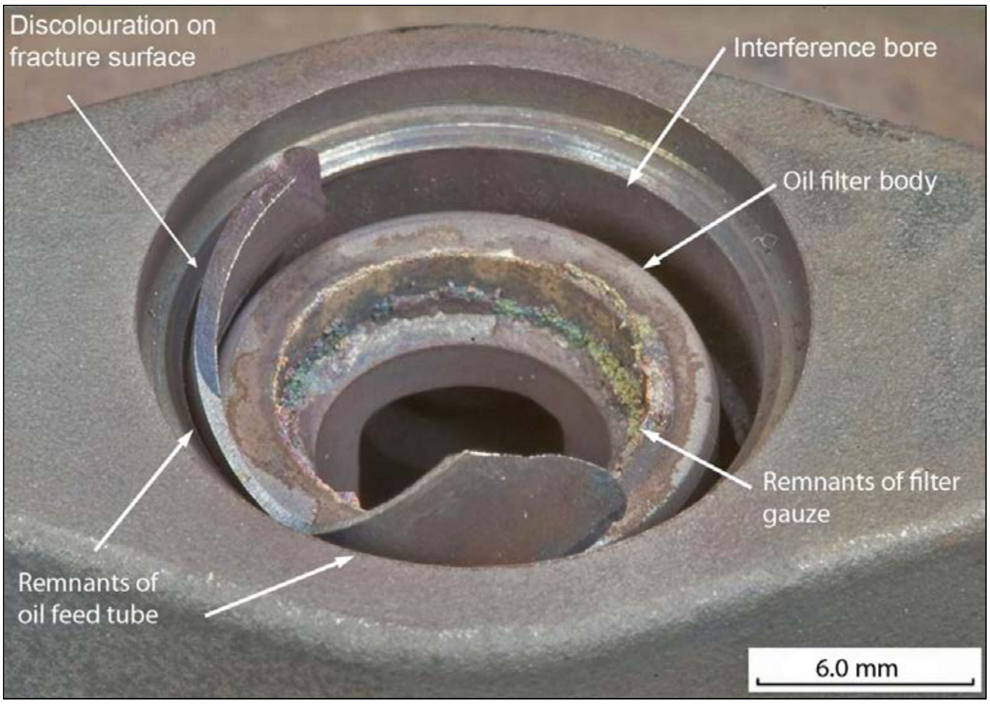
The oil feed stub pipe had detached and displaced outwards about 30 mm from the inner hub. A remnant of the oil feed stub pipe remained in the inner hub (Figure 41).

Figure 41: Inner and outer HP/IP bearing hubs at the oil feed pipe location



Although the filter body was still in place, the wire gauze filter element was missing (Figure 42). Upon inspection, it was apparent that the filter body was misaligned with respect to the interference fit bore in the inner hub. There was also an additional misalignment of the filter body with respect to the oil feed stub pipe as the remnant of that pipe was trapped between the HP/IP inner hub bearing (Figure 42).

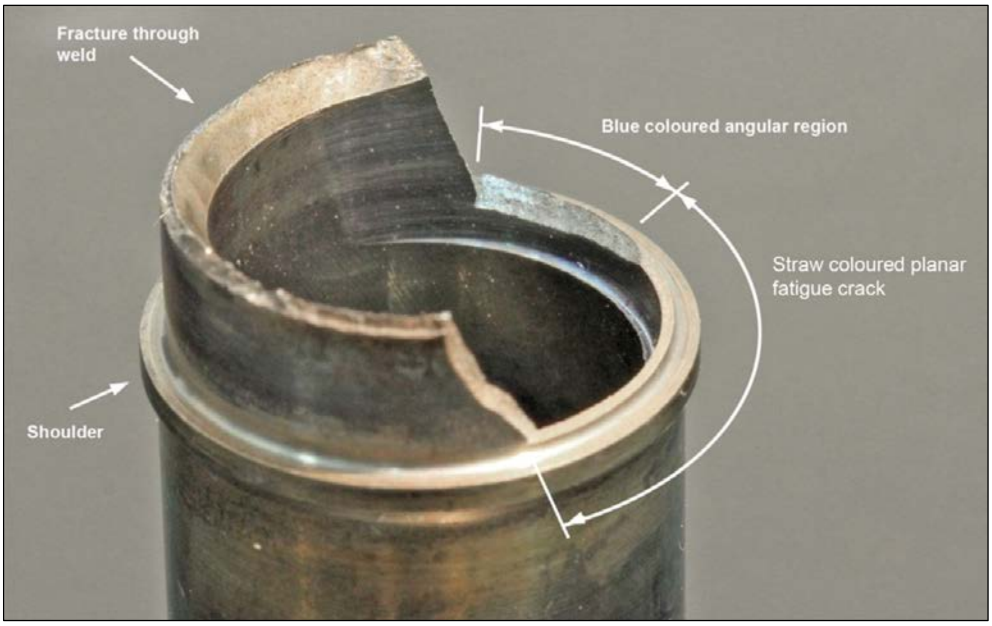
Figure 42: HP/IP inner hub oil feed pipe



The oil feed stub pipe was fractured in two planes, one around the weld that attaches the pipe to the inner hub, and the other further outboard, close to the shoulder where the pipe presses into the inner hub. Three distinct fracture regions were identified on the oil feed stub pipe:

- one through the stub pipe-to-hub weld
- a straw coloured planar region
- a blue coloured angular region (Figure 43).

Figure 43: Oil feed stub pipe fracture



Detailed laboratory examination of the fracture regions found that the majority of the cracking, including the blue coloured region, was typical of an overstress fracture. The planar region covered an arc of about 90° of the oil feed stub pipe and was identified to be a pre-existing fatigue crack. The origin of this crack could not be precisely determined, however, the general shape of the crack indicated its origin on the outer surface of the pipe before growing inwards.

A band of fretting was identified on the outer diameter of the stub pipe at a position coincident with the fatigue cracking (Figure 44). The fretting was along the area that formed an interference fit with the inner bearing hub, indicating relative movement between the stub pipe and the inner hub interference bore. The engine manufacturer reported observing similar fretting bands on some of the engine's other HP/IP stub pipes.

Figure 44: Oil feed stub pipe fretting

Of particular note, the wall thickness of the fractured end of the oil feed stub pipe was not uniform. A number of measurements were taken around the stub pipe wall showing that the wall thickness varied from 1.42 mm down to 0.35 mm (Figure 45).

Figure 45: Fractured oil feed stub pipe wall measurements

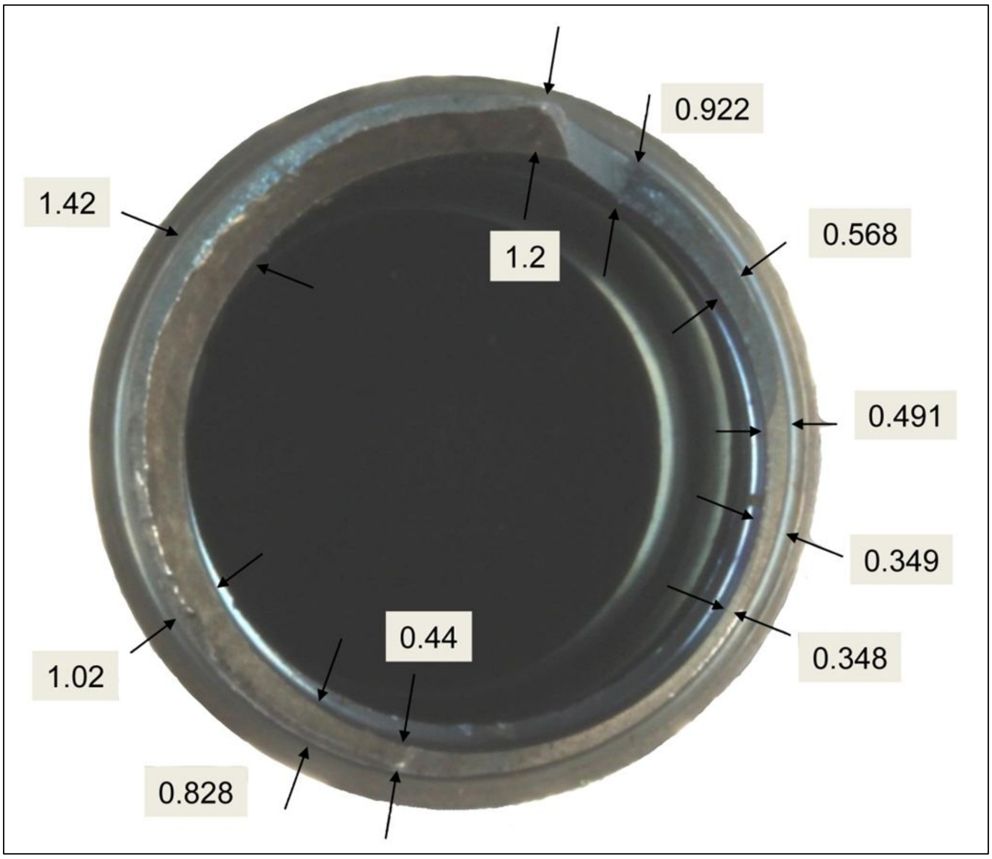


Image modified from a Rolls-Royce plc supplied mode

The fatigue crack coincided with the thinnest wall section and it was apparent that the counter bore[62] used during manufacture to enlarge the inner diameter of the oil feed stub pipe to accept the oil filter was offset from the centre of the pipe (Figure 46). The manufacturer’s counter boring operation was a two-stage process that initially involved drilling into the pipe end. The pipe end was then reamed[63] to a shallower depth than the drilled hole to form a close-tolerance fit for the filter body.[64] The drilled and reamed sections of the counter bore were misaligned with respect to the centreline of the pipe and each other (Figure 47).

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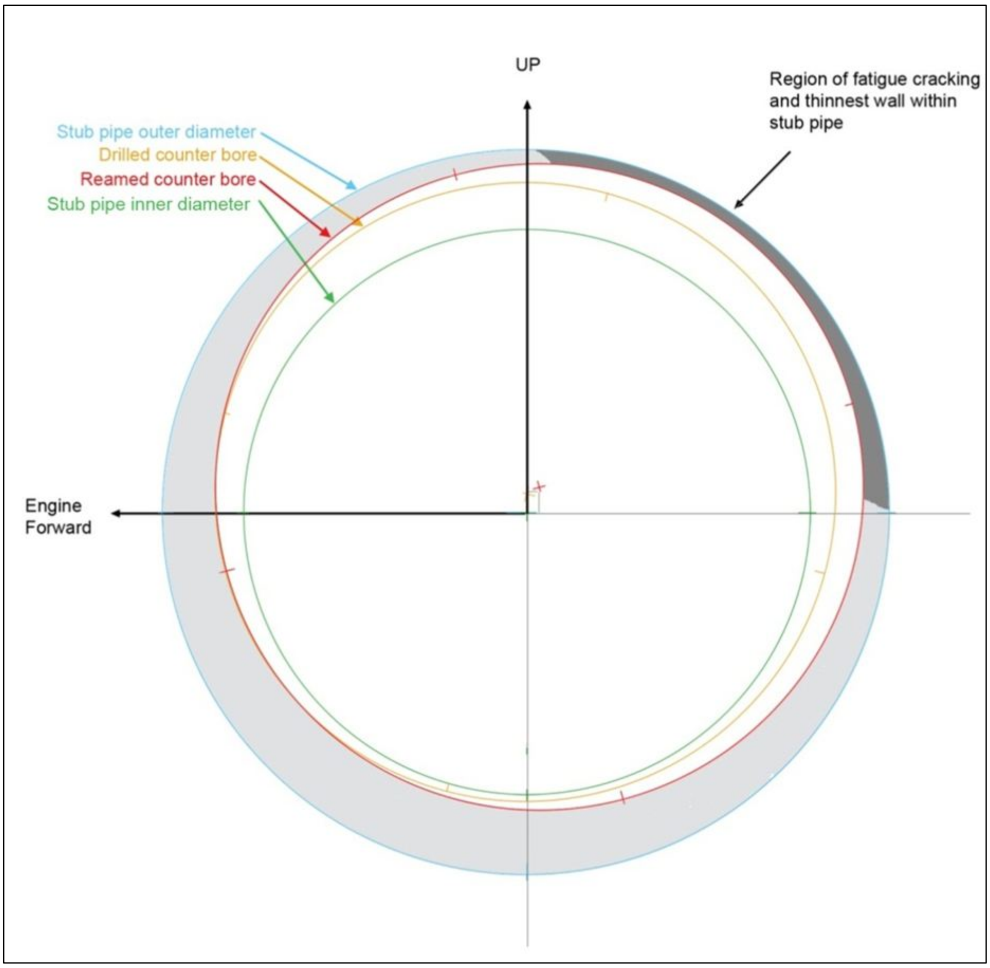
[62] A counter bore is a hole of defined depth that is used to locally enlarge another hole.

[63] The use of a rotating tool to enlarge and finish a hole or opening.

[64] Further detail on the manufacturing processes involved in the counter bore is provided in Part 4.

Figure 46: Offset oil feed stub pipe counter bore

Figure 47: Diagrammatic representation of the offsets produced during manufacturing operations



2.4 Intermediate pressure turbine behaviour

2.4.1 Disc burst

During the normal operation of a gas turbine engine, the load applied by the hot airstream as it passes through a turbine is transferred through the interconnecting shaft to the compressor. That load is used to drive the compressor to increase the pressure of the air passing through the compressor. A turbine will rotate at a constant speed when there is a balance between the load applied to the turbine by the hot airstream and the compressor load as it compresses the air.

When a drive shaft (or drive arm) fails, the compressor load is removed from the turbine and the system is no longer balanced. This condition, known as a 'loss of load' on the turbine, results in a decrease in compressor speed and an increase in turbine speed, as the hot gas stream is still applying a load on the now unloaded turbine.

When a disc is rotating, there is a radial expansion of the disc as a result of centrifugal forces generated by the rotation. The greater the rotational speed, the greater the forces that are generated. These centrifugal forces are resisted by the stiffness of the disc[65] and result in stresses within the disc. When the speed is such that those stresses exceed the yield strength of the disc, permanent deformation results and the disc 'grows' larger than its original size. When the stresses exceed the ultimate strength of the disc, the disc fails. That failure is typically in tension, and results in radial (that is, from the centre of the disc outward to the rim) overstress factures. Thus, a disc that failed in overspeed (disc burst) will exhibit permanent growth and radial overstress fractures.

Because a disc burst generally occurs at a high rotational speed, the released debris has sufficient energy to pass through the engine case at great speed and can impact the aircraft structure. The non-containment of significant high-energy debris is defined as a 'hazardous effect'[66] and the prevention of a disc burst is paramount for designers and manufacturers. It is therefore critical that the speed of a turbine disc under any condition does not exceed its burst speed.

2.4.2 Design and certification – predicted IP turbine behaviour

Although a turbine disc will accelerate when it loses its load, the turbine's speed will not increase indefinitely. The disc's maximum, or terminal, speed is limited by a number of factors, depending on where in the system the failure occurs and how the engine behaves following the failure. The designer needs to show that the terminal speed is below the burst speed of the disc.

To determine the Trent 900 IP turbine disc's terminal speed, the engine manufacturer assessed how the engine would behave following an IP shaft failure at a range of locations. This analysis identified a number of factors that combine to limit the terminal speed.

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65 The stiffness of the disc is a combination of the material properties and its geometric configuration (size and shape).

66 As defined by Joint Aviation Requirements – Engines (JAR-E), Section 2 *Acceptable means of compliance and interpretations*, ACJ E 510.

The engine manufacturer predicted that on failure of the IP shaft, the IP turbine disc would move rearward under gas loads and immediately engage with the hardware behind the disc (the LP turbine front panel). This contact would result in frictional loads that resisted the rotation of the IP disc. The rearward movement of the disc would also partly move the turbine blades out of the gas flow, reducing their efficiency and hence reducing the load being applied to the disc. Because the IP compressor would no longer be driven by the IP turbine, it would rapidly decelerate. The deceleration rate was predicted to be so great that the engine control system could not properly maintain the airflow through the compressor and a compressor stall would result. The resulting stall would affect the whole IP compressor, which would no longer be capable of holding back the already compressed air in the engine. In consequence, it was believed that the flow through the compressor would reverse, causing the engine to surge. During the surge, the air pressure within the engine and the air flow through the engine would rapidly reduce, further reducing the load applied to the turbine.

During testing of the Trent 700, a geometrically and aerodynamically similar predecessor to the Trent 900, one of the development engines experienced a shaft over-torque at very high thrust levels. On another test engine, a shaft failure occurred at mid-thrust settings due to an internal oil fire. In both cases the engine surged, the HP compressor locked in stall and the engine ran down without a disc burst.

During the design and certification[67] of the Trent 900, the manufacturer's modelling, testing and experience from the above Trent 700 shaft over-torque and shaft failure events indicated that the effect of a surge would include the HP system. It was expected that the HP compressor would remain locked in stall and would run down. The terminal speed calculations for the IP turbine 'loss of load' case in the Trent 900 were based on this expected behaviour and the resulting pressure profile through the IP turbine.

The Trent 900 also included a number of design changes over the Trent 700 that were expected to further improve the post-shaft failure behaviour of the engine.

2.4.3 Certification testing

The design standard required that an engine's fan, compressor and turbine would not burst when operated at the engine's most critical speed. An engine's most critical speed was considered to be the highest of:

- 120% of its maximum operating speed
- 105% of the maximum speed resulting from a component or system failure
- 105% of the highest overspeed resulting from a loss of load on a turbine.

The critical speed for the IP turbine in the Trent 900 was determined to be 105% of the maximum overspeed from a loss of load on the turbine.

The fan, compressors and turbines in the Trent 900 were all individually tested during certification. The test speeds were increased until the disc under test burst and showed that the IP turbine burst speed was in excess of the requirement. The

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67 Certification is the showing of compliance with an accepted design standard. In the case of the Trent 900, the design standard was JAR-E at Amendment 11.

test IP disc burst into four main fragments, the largest of which was similar in size to the one found on Batam Island from the No.2 engine in VH-OQA (Figure 48).

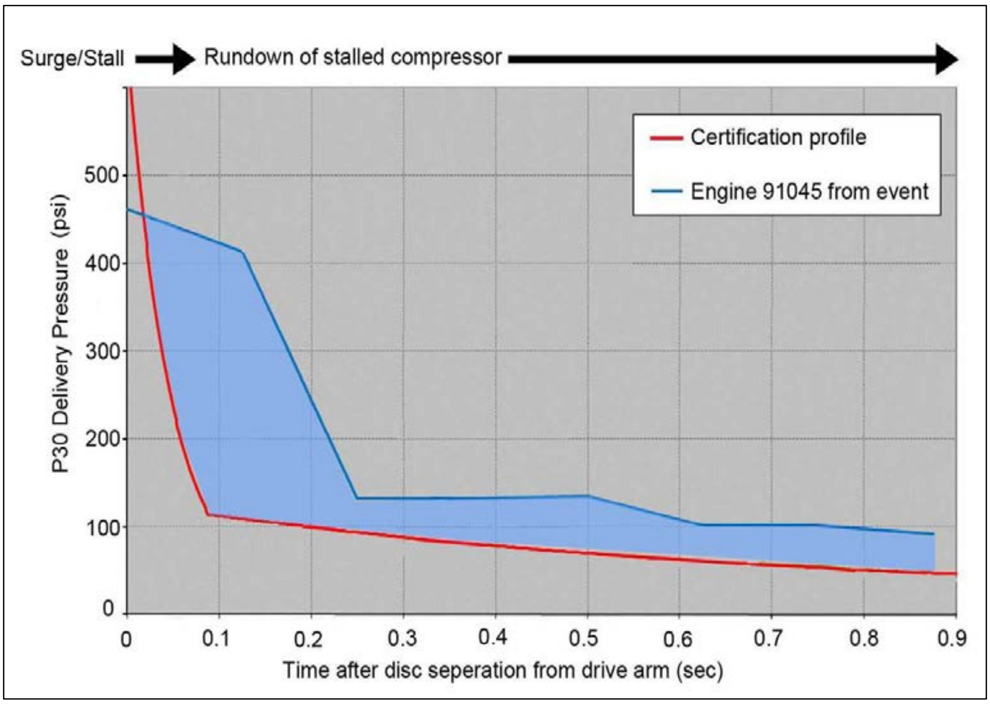
Figure 48: Trent 900 IP turbine disc after overspeed burst test

Image source: Rolls Royce plc.

2.4.4 Actual engine behaviour

The recorded data from the uncontained failure of ESN 91045 identified an engine behaviour that differed from that predicted following a drive shaft failure. The data showed that when the drive arm failed at the R850 holes, the engine surged as expected. However, the air pressure in the engine did not decay to the level predicted from the manufacturer's modelling. A comparison between the expected 'certification' P30 delivery pressure profile and that recorded during the uncontained failure in ESN 91045 is presented in Figure 49.

Figure 49: Post-surge engine behaviour – a comparison of the HP compressor pressure at certification and in ESN 91045



Source: Rolls Royce plc.
Note: The blue region depicts the differences between the Trent 900 certification profile and that recorded during the uncontained failure of ESN 91045.

The engine manufacturer determined that the higher than expected pressure recorded during the failure was due to a partial recovery of the HP compressor. That recovery was unexpected, and according to the engine manufacturer had not been previously observed. The additional pressure provided by the HP system following the surge (indicated by the blue shaded area in Figure 49), accelerated the now unloaded IP turbine beyond the predicted terminal and burst speeds.

The partial recovery of the HP system was not completely understood by the engine manufacturer. It was not known if the observed behaviour was limited to the particular set of conditions encountered, or whether the behaviour could occur in other conditions.

Observation:

After the accident, the engine manufacturer developed a Trent 900 intermediate pressure turbine overspeed protection system (IPTOS). It was enabled through a modification to the FADEC software and was designed to limit the available energy and prevent the IP turbine from reaching burst speed following an internal engine oil fire.

Part 7 of this report discusses this modification further.

2.4.5 Certification failure analysis

To show compliance with the design standard, the engine manufacturer was required to carry out a failure analysis and show that the probability of a hazardous effect would not occur at a rate in excess of that defined as 'extremely remote'.[68] The Trent 900 failure analysis was contained in the certification documentation in the form of a *failure modes, effects and criticality analysis* (FMECA).

The Trent 900 FMECA consisted of a large number of reports. A summary FMECA was prepared and accepted by the certifying authority[69] as part of the certification of the engine. The FMECA was based on the assumption that all manufactured parts conformed to the design specifications.

The failure modes identified in the FMECA that resulted in a shaft, or drive arm failure, were categorised as either having a major effect[70] or a hazardous effect. A number of those failure modes included a description of the surge and engine rundown behaviour preventing an overspeed failure.

The consideration of a failure mode very similar to the oil feed stub pipe failure in ESN 91045 was recorded in the Trent 900 FMECA. Although the description of the propagation of the failure was not exactly the same as the actual failure, the end result (failure of the IP turbine drive and overspeed above the critical burst speed) was predicted and assessed. It was classified as a hazardous effect and was determined to have the acceptable probability of extremely remote.

Post-event review

Following the uncontained failure of ESN 91045, the engine manufacturer carried out a complete revision of the Trent 900 FMECA and applied the knowledge gained from the actual engine behaviour. The revised summary FMECA was also reviewed by the ATSB and it was found that the probability of all of the hazardous failure modes was still considered by the manufacturer to be extremely remote. Those failure probabilities were based on the probability of failure of the applicable component (for example, the oil feed stub pipe), and were not reliant on the post-shaft failure behaviour of the engine.

2.4.6 Classification of the HP/IP structure

In order to ensure, amongst other things, that the appropriate levels of process control were applied to manufactured parts, the engine manufacturer had a system that classified the parts in terms of the criticality of their failure. There were three classification levels in the manufacturer's procedures as follows:

Critical part

> For all applications critical parts are those whose primary failure is shown by the failure analysis as likely to have hazardous effects, and which consequentially require special controls in order to achieve an acceptably low probability of occurrence.

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68 JAR-E defined extremely remote as 'not greater than 10^{-8} per engine flying hour'. That is, a failure effect that is unlikely to occur during the total operational life of all engines of that type, but nevertheless has to be regarded as being possible (one failure in not less than 100,000,000 flying hours).

69 For the Trent 900, the certifying authority was the European Aviation Safety Agency (EASA).

70 There was no specific definition of a major effect in JAR-E, but it was considered to be one that fell between a minor effect (one in which the only consequence was a partial or complete loss of thrust from one engine) and a hazardous effect.

Reliability sensitive part

Reliability sensitive parts are those that are not critical but which meet one of the following criteria a) or b) below:

a) Their failure is likely to have a significant effect on reliability or cost of operation and which require special controls to achieve an acceptably low rate of failure. Significant reliability effects include In flight shut down (or in the case of a single-engine aircraft insufficient power to sustain flight), Mission abort/diversion/turn back. (note: this list may not be exhaustive)

b) They have a significant impact on the achievement of the specification performance and require special controls to ensure that their performance characteristics are consistently achieved.

and

A component is regarded as sensitive to source or method of manufacture where changes to source or method, even though correctly applied, may prevent the design intent being consistently achieved.

Unclassified part

Unclassified parts are those parts that are neither Critical nor Reliability sensitive.

When parts were classified as critical or reliability sensitive, the design definition drawings were annotated to indicate that status.

The FMECA that was carried out during certification of the Trent 900 identified the HP/IP bearing support assembly as being ‘Unclassified’.

Observation:

The engine manufacturer’s post-event review of the FMECA identified that the HP/IP bearing support structure had been inappropriately classified and has been reclassified as a Reliability Sensitive part.

2.5 Oil feed stub pipe stress analysis

2.5.1 Stress analysis of the oil feed stub pipe

The engine manufacturer carried out a stress analysis of the oil feed stub pipe to determine the distribution of the stresses during normal engine operation, in particular the location of the maximum or peak stress. The analysis included a detailed computer model of the oil feed stub pipe from ESN 91045 that included the actual wall thickness distribution.

The stresses in the oil feed stub pipe resulted from the bending in the pipe from the relative movement between the engine case and the HP/IP bearing hub. Other analysis by the engine manufacturer identified that the magnitude of that movement was directly related to the HP compressor delivery pressure (P30). The greatest movement of the hub, and therefore stress on the oil feed stub pipe, during normal operation was at the engine’s maximum thrust setting and measured about 6 mm. This movement was used in the computer stress model.

The stress analysis identified that the peak stress in the oil feed stub pipe occurred on the outer surface of the pipe, between the shoulder and the interference fit in the inner hub. That peak stress was highly localised, and was significantly greater than the other stresses in the pipe (Figure 50). The location of the peak stress corresponded with the location of the fatigue crack in the oil feed stub pipe in the occurrence engine.

Figure 50: Oil feed stub pipe stress distribution

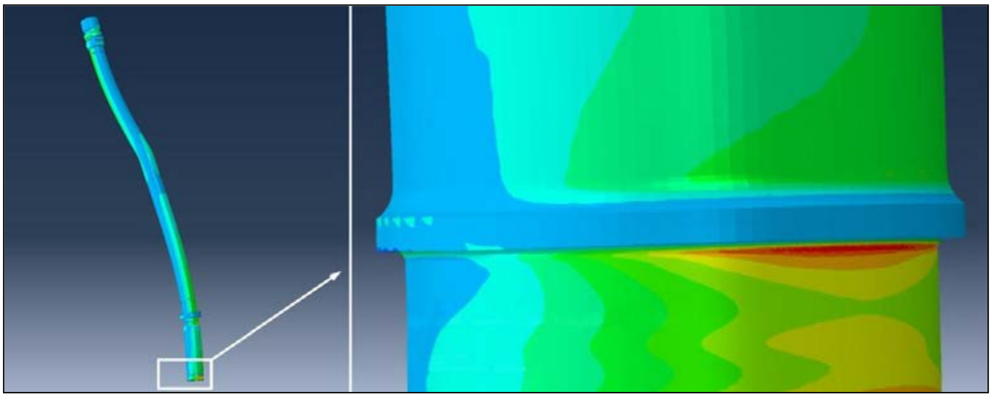


Image source: Rolls Royce plc.

The blue represents minimum stress, graduating through green and yellow up to red, which represents the maximum stress.

2.5.2 Effect of wall thickness

To assess the potential for similar fatigue cracking on the remainder of the Trent 900 engine fleet, the manufacturer analysed a number of other misaligned counter bore scenarios. The movement of the hub was directed along the length of the engine and moved rearward as thrust increased. This would generate bending stresses that would be greatest if the minimum wall thickness was towards the rear of the engine. Thus, in order to be conservative, the manufacturer's modelling was based on the minimum wall thickness being located in that position. The minimum wall thicknesses assessed were 0.50, 0.56, 0.70 and 0.91 mm. The resulting stress distributions reflected that shown in Figure 50. The peak stresses recorded below the shoulder in the manufacturer's testing are presented in Table 8 and Figure 51.

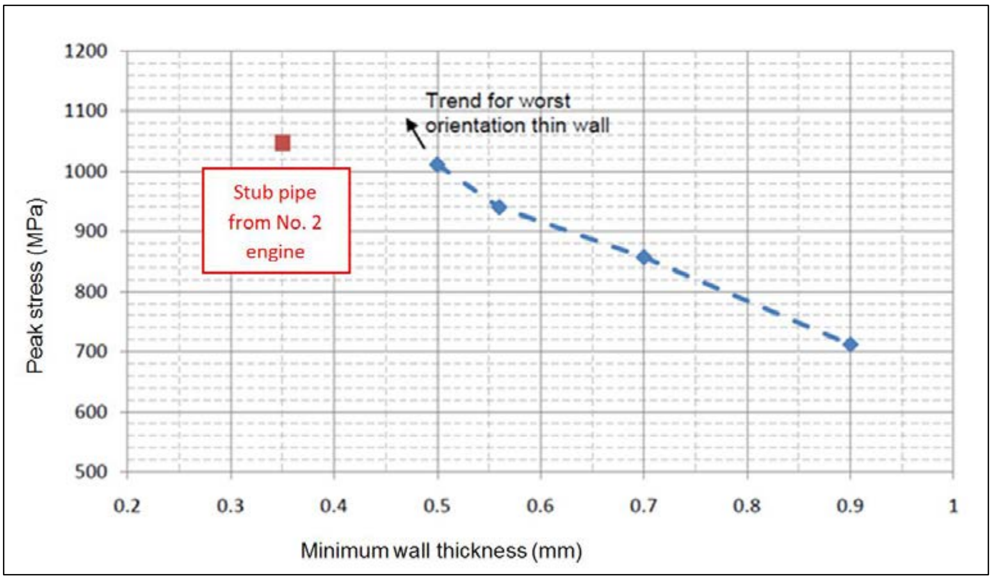
Table 8: Effect of wall thickness on peak stress

Minimum wall thickness (mm)	0.35[71]	0.50	0.56	0.70	0.91
Peak Stress (MPa)[72]	1047	1011	940	857	712

[71] The 0.35 mm wall thickness represented ESN 91045 and the stress presented here is for the actual orientation, not the worst case scenario.

[72] Megapascal.

Figure 51: Effect of wall thickness on peak stress



Source: Rolls Royce plc.

A detailed vibration survey was also carried out by the manufacturer and it was determined that the number of engine cycles (low-cycle fatigue) was the primary contributor to the fatigue in the oil feed stub pipe, rather than the effects of a high frequency vibration (high-cycle fatigue). Based on these findings, the manufacturer carried out a fatigue life analysis of the oil feed stub pipe. This analysis conservatively assumed that the engine developed full thrust for each of its operating cycles, effectively representing a worst case scenario in an engine's life.[73] The resulting worst case fatigue lives are presented in Table 9 and Figure 52, along with ESN 91045's actual life of 677 cycles at the time of the uncontained failure.

Table 9: Calculated fatigue lives

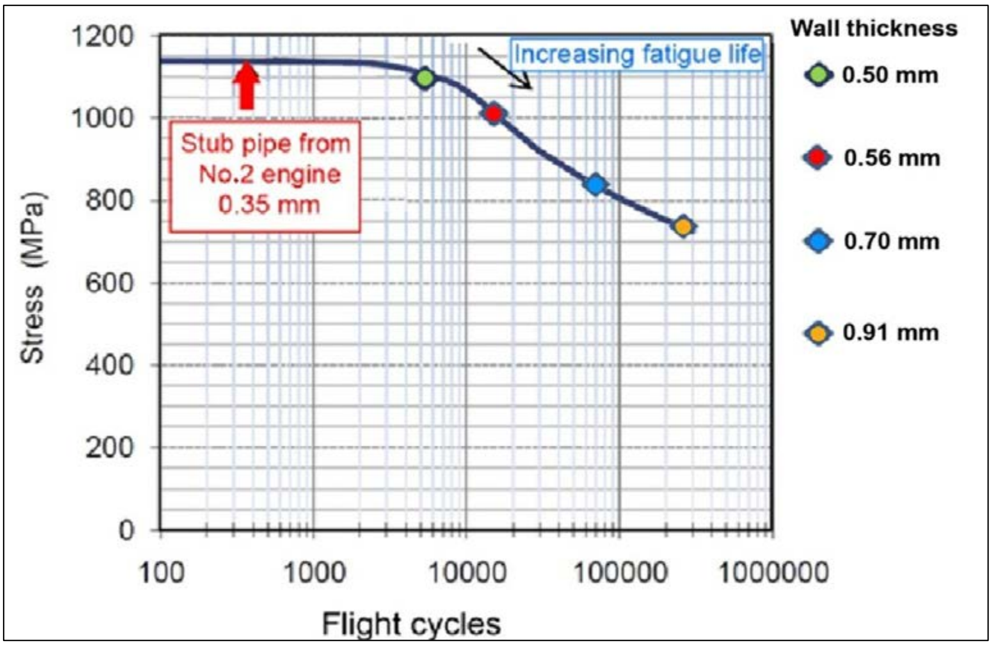
Minimum wall thickness (mm)	0.35[74]	0.50	0.56	0.70	0.91
Fatigue life (flight cycles)	677	5,445	13,892	60,000[75]	159,378

[73] The majority of commercial jet aircraft takeoffs are reduced thrust takeoffs, where less than the engine's maximum thrust is used.

[74] This is the number of cycles accumulated by ESN 91045 at the time of failure. The fatigue lives presented here have been calibrated to this number of cycles.

[75] This value was not provided by the manufacturer and has been estimated by the ATSB from Figure 52.

Figure 52: Calculated fatigue lives



Source: Rolls Royce plc.

2.6 Recorded engine data

The following information was assembled from a number of data sources including the FDR, EEC, EMU and other aircraft-based recorders. The recorded engine data sequence of events is presented in Table 10 and Figure 53 to Figure 56. Further detail on the recorded data is contained in Appendix C.

Table 10: Sequence of events relevant to the No. 2 engine from the recorded data

Actual time (UTC) hh:mm:ss	Time sequence	Comment
01:45:30	-15 min 37 sec	All four engines started and stabilised at idle and with an N1 of 17%.
01:55:00	-6 min 07 sec	Aircraft commences takeoff.
01:56:47	-4 min 20 sec	Aircraft airborne, No. 2 engine N1 at 79%.
01:59:07	-2 min 0 sec	Crew select CLIMB thrust setting. No. 2 engine N1 increases to about 86% within 13 seconds.
02:00:07	-60 sec	No. 2 engine turbine cooling air rear (TCAR) begins to rise in relation to the other engines (TCAR not displayed to crew)
02:00:15	-52 sec	No. 2 engine oil temperature and pressure values begin to diverge from the recorded values for the other engines.
02:00:58	-9 sec	No. 2 engine N3 starts to fluctuate and the N1 in that engine shows 87%.

Actual time (UTC) hh:mm:ss	Time sequence	Comment
02:01:01	-6 sec	No. 2 engine HP turbine vibrations peak in amplitude[76] and the N1 and N2 drop by about 1%. N3 in the No. 2 engine increases by about 3% and there is a large spike in turbine cooling air forward (TCAF) and TCAR.
02:01:03	-4 sec	No. 2 engine HP turbine vibrations rapidly decrease.
02:01:06	-1 sec	No. 2 engine N1 at 87%.
02:01:07	0 sec	Rapid and large reduction in the No. 2 engine N1 and N2 and a sudden and rapid drop in P30 in engine No. 2. Turbine disc separates from drive arm.
02:01:08	+1 sec	No. 2 engine oil pressure drops, TCAF peaks in temperature, No. 2 turbine overheat ECAM warning activates and N3 stabilises at 96%.
02:01:09	+2 sec	Master Warning and Caution activate in the cockpit.
02:01:11	+4 sec	No. 2 engine TCAR loss of signal, second drop in P30 (less than at t=0), N3. Disc burst.
02:01:17	+10 sec	No. 2 engine oil pressure decreases.
02:01:18	+11 sec	No. 2 engine oil quantity begins to decrease.
02:01:38	+31 sec	No. 2 engine thrust lever reduced to idle.
02:01:53	+46 sec	No. 2 engine thrust lever progressively advanced to half range.
02:02:08	+1 min 01 sec	No. 2 engine P30 increases.
02:02:20	+1 min 13 sec	No. 2 engine N3 and P30 peak then rapidly drops.
02:02:25	+1 min 18 sec	No. 2 engine fire ECAM warning.
02:02:39	+1 min 32 sec	No. 2 engine thrust lever reduced to idle.
02:03:21	+2 min 14 sec	No. 2 engine high pressure shut-off valve closed and engine shutdown.

76 The vibrations were below the triggering threshold for ECAM notification.

Figure 53: Graphical representation of Table 10

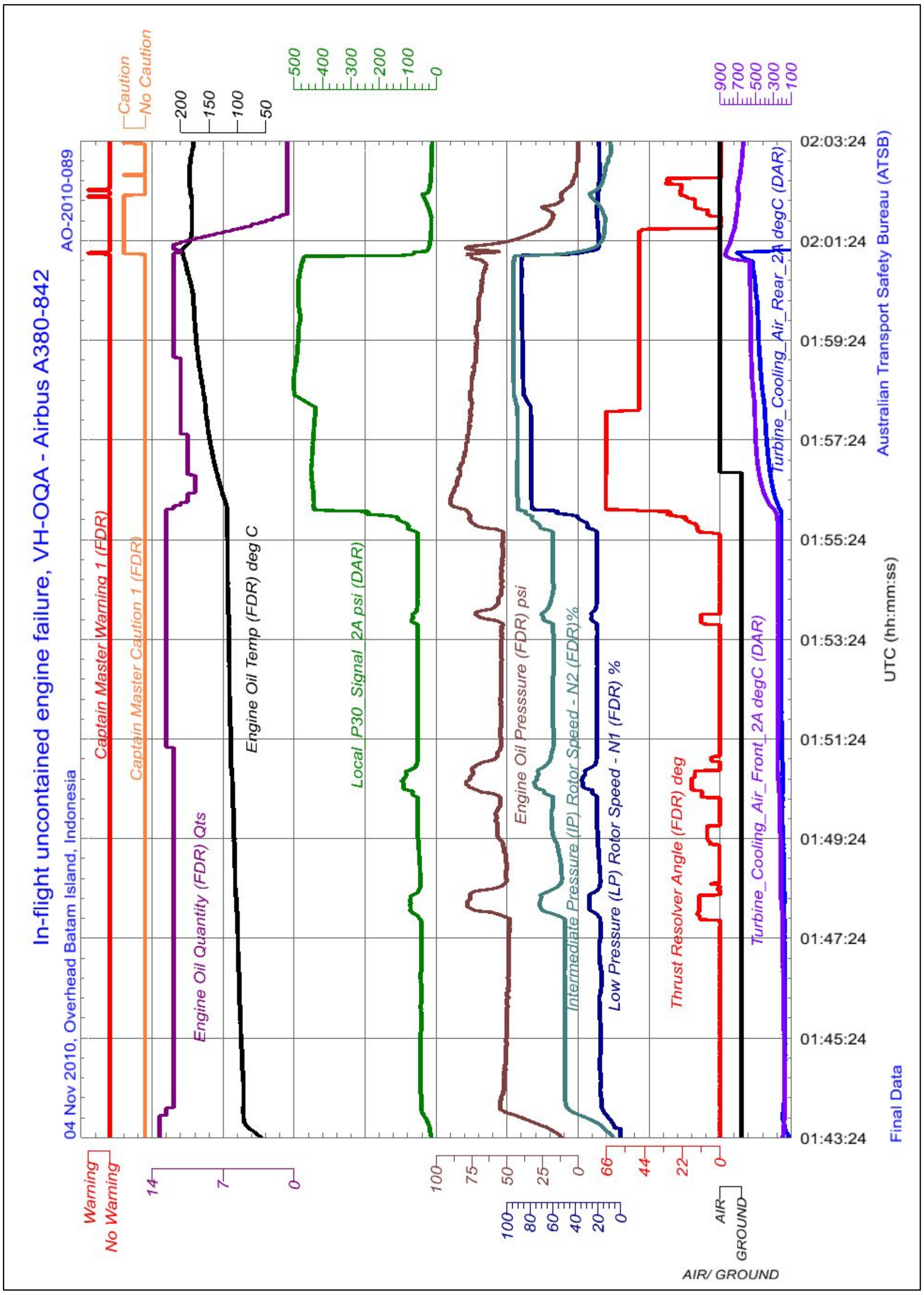


Figure 54: Graphical representation of engine data from 02:00:00 to 02:01:30

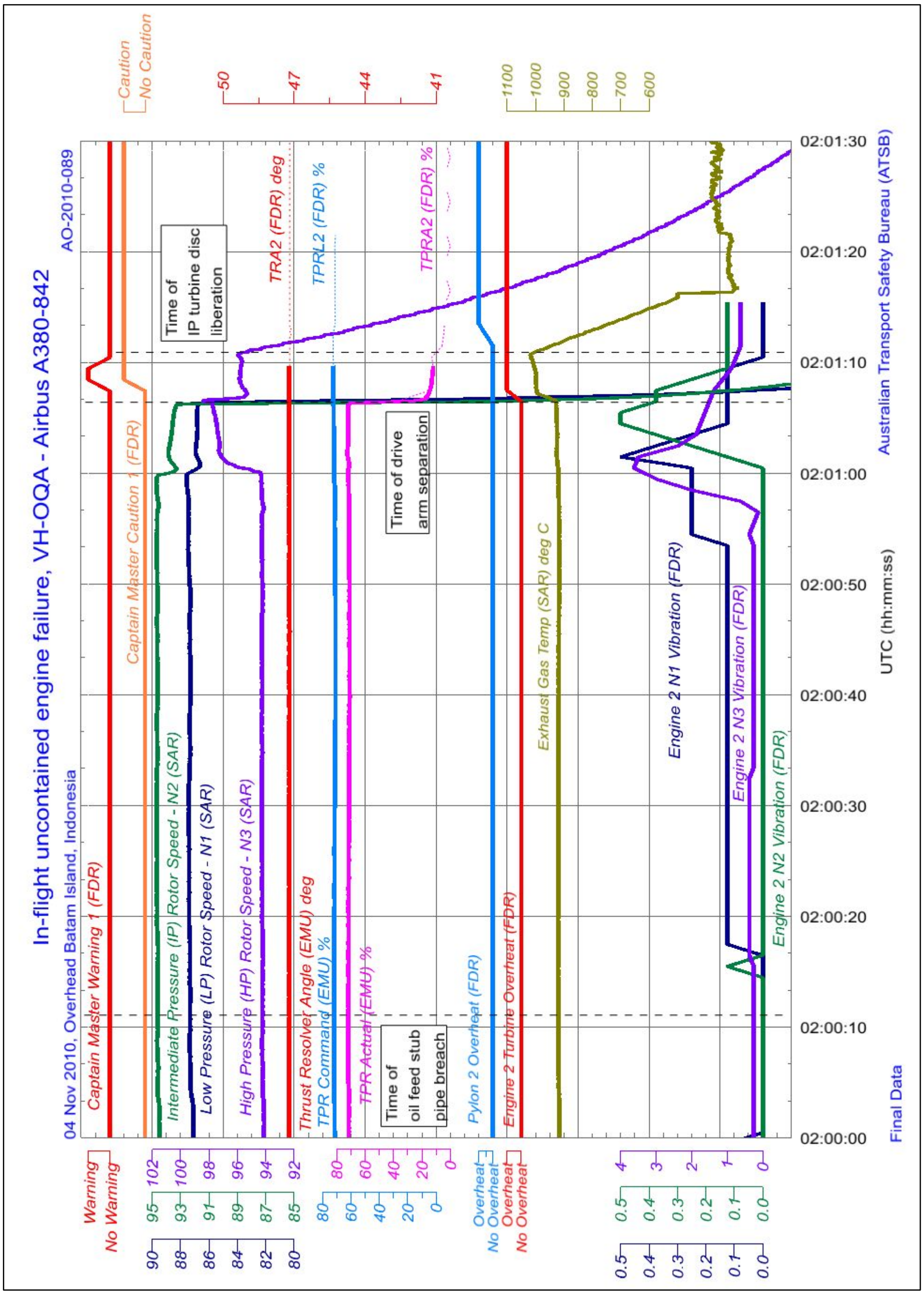


Figure 55: Graphical representation of engine data from 02:00:00 to 02:01:30

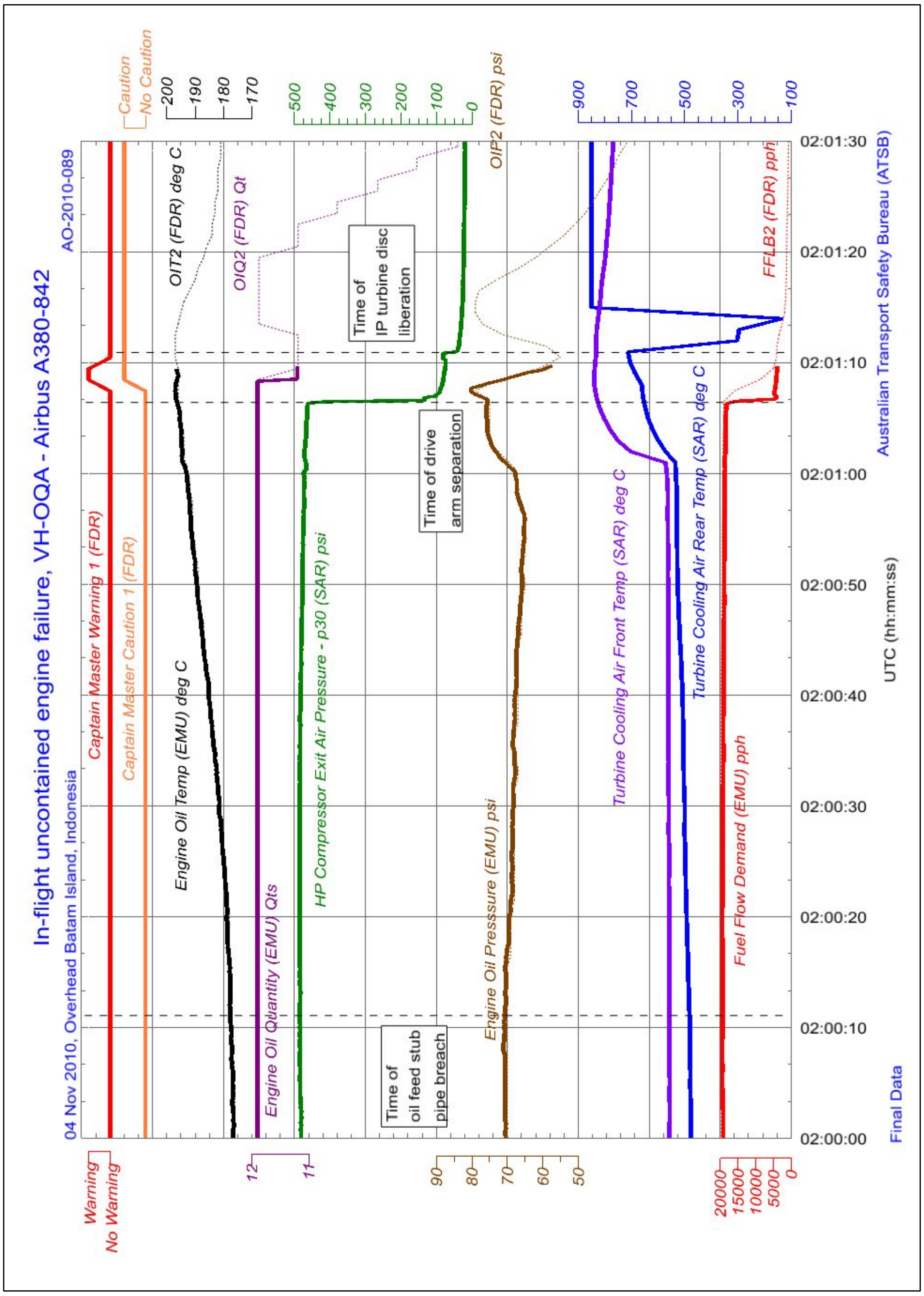
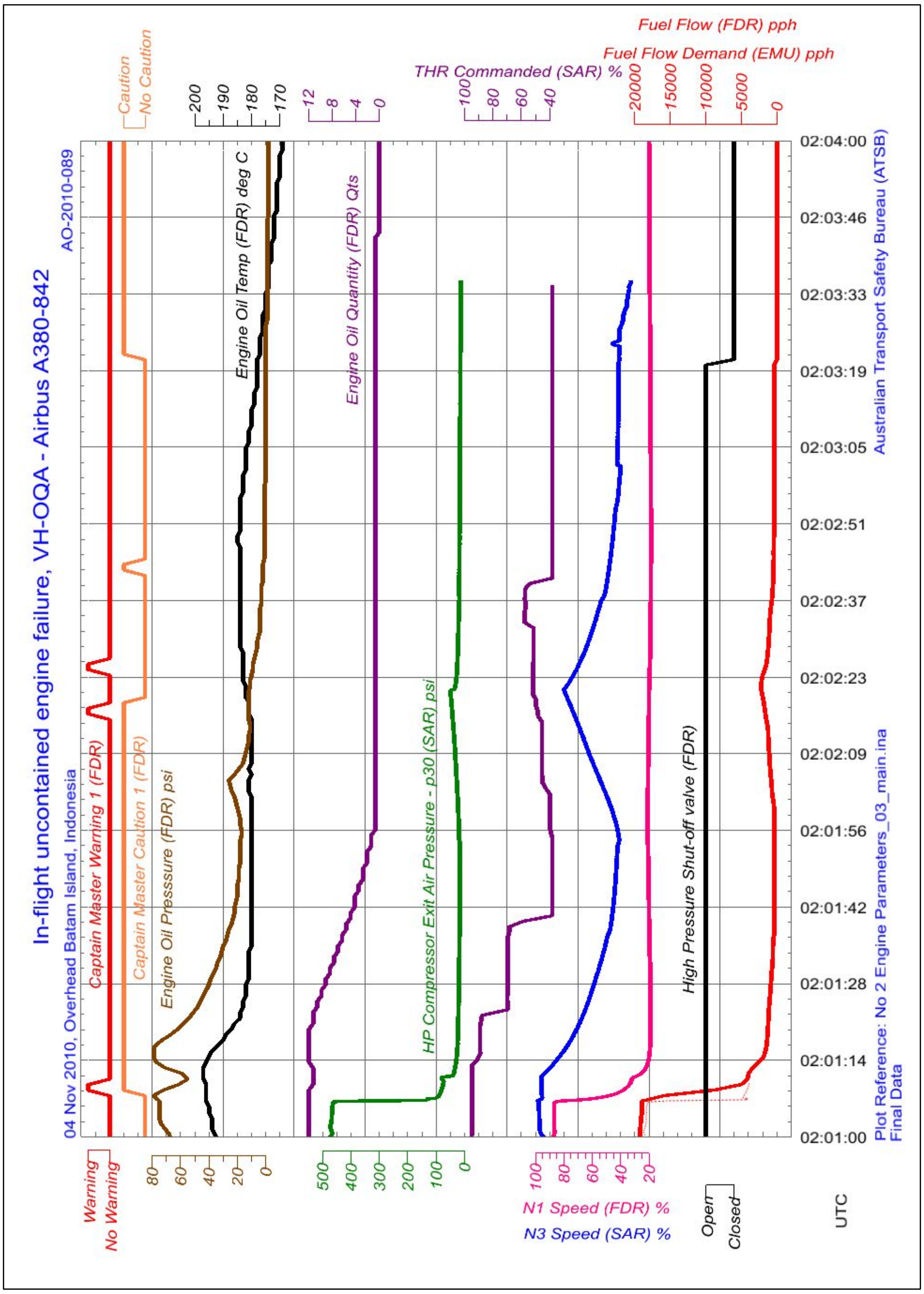


Figure 56: Graphical representation of engine data from 02:01:00 to 02:04:00



3 FACTUAL INFORMATION: ENGINE FAILURE SEQUENCE

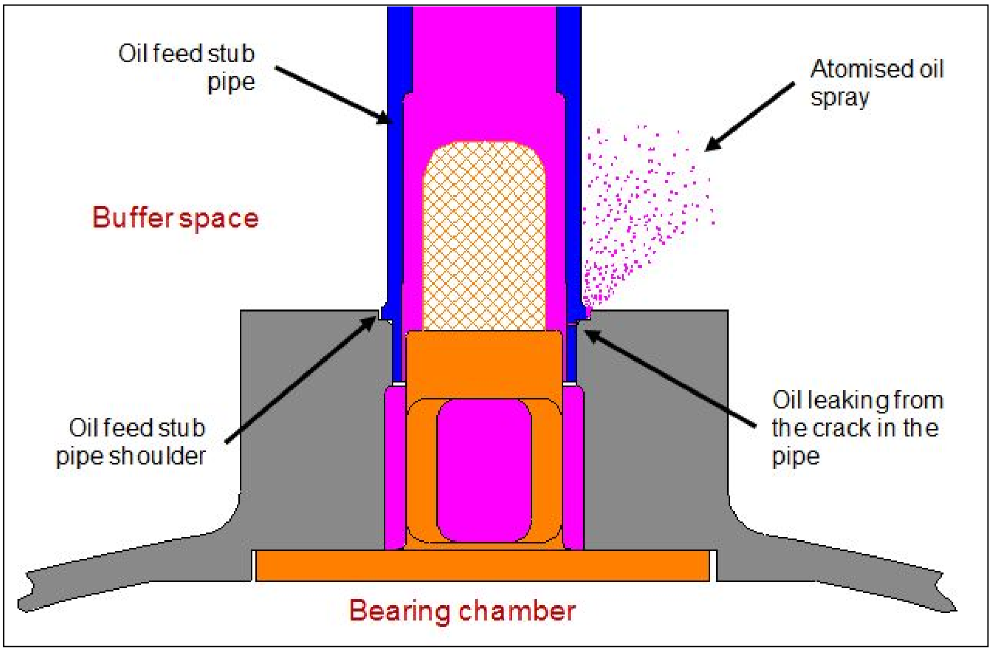
This part of the report details the sequence that led to the uncontained failure of the aircraft's No. 2 engine. The failure sequence presented is based on the physical and recorded evidence.

The failure sequence occurred in a continuous manner, but contained discrete events that broke the sequence down into five key phases. The following discussion examines the failure sequence in terms of those phases.

3.1 Phase 1: Oil feed stub pipe failure and oil fire

As described in Part 2 of this report, a fatigue crack initiated and grew in the HP/IP oil feed stub pipe below the shoulder where it fits into the HP/IP bearing chamber inner hub. It was not determined when the crack started; however, during the occurrence flight the crack grew to a size that allowed oil to leak out of the pipe and into the buffer space. The leak was at a low rate but at a pressure that likely resulted in an atomised oil spray (Figure 57).[77]

Figure 57: Oil leak into the buffer space



The temperature in the buffer space was reported by the manufacturer to be in the range of 365 to 375 °C. The synthetic oils used in the engine had auto-ignition temperatures as low as 280 °C. A comparison of these temperatures indicates the potential for auto-ignition and, given that there were no other ignition sources identified, it was likely the leaking oil auto-ignited when it atomised on entry into the buffer space.

[77] The buffer space was designed to be an oil free environment.

The direction of the airflow was from the buffer space, along the IP drive shaft and into the low pressure (LP) turbine cavity (represented as red arrows in Figure 58), where the TCAR sensor was located. That airflow directed a portion of the fire forward from the leaking oil pipe and onto the front of the HP/IP buffer space.

Given that the air temperature, as measured by the TCAR sensor, started to increase at about 02:00:07, 60 seconds before the engine failure, it is likely that the fire started shortly before that time.

Figure 58: Oil leakage and fire

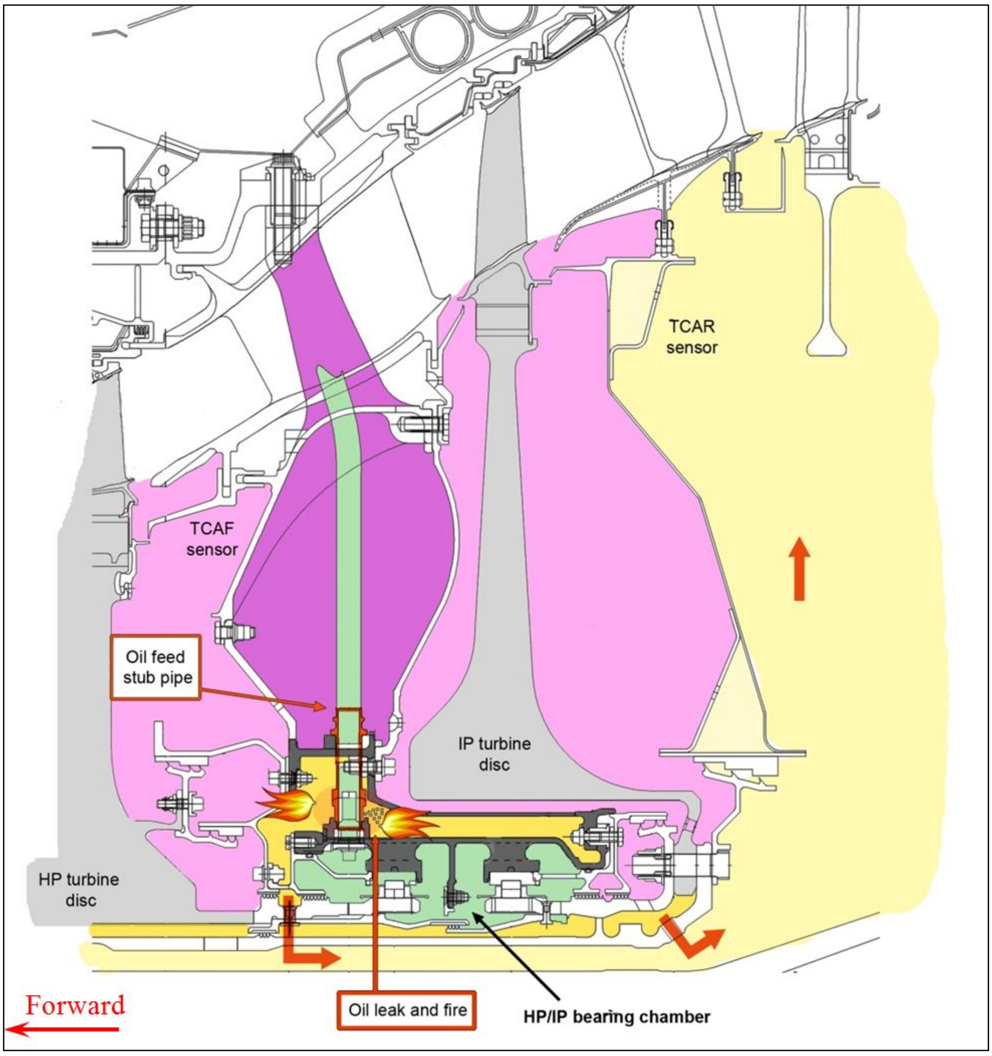


Image modified from a Rolls-Royce supplied model

3.2 Phase 2: HP turbine triple seal failure

About 10 seconds before the engine failure, the fire breached the front face of the HP/IP bearing chamber buffer space and impinged directly on the HP/IP triple seals. The intense heating distorted the seals, causing them to come into contact with each other.

The distortion and contact significantly increased the vibration levels in the HP section. That vibration increased, until about 4 seconds later, it rapidly decreased when the triple seal separated from the structure and the static and rotating portions became entwined, scoring the rear face of the HP turbine disc (Figure 59).

The design of the engine was such that the pressure within the buffer space and behind the IP turbine was lower than the hot (annulus gas) airstream exiting the HP turbine. This meant that when the triple seals were breached, those hot gases were drawn into the space behind the HP turbine disc, where the TCAF sensor was located (Figure 59).

Figure 59: HP turbine triple seal failure

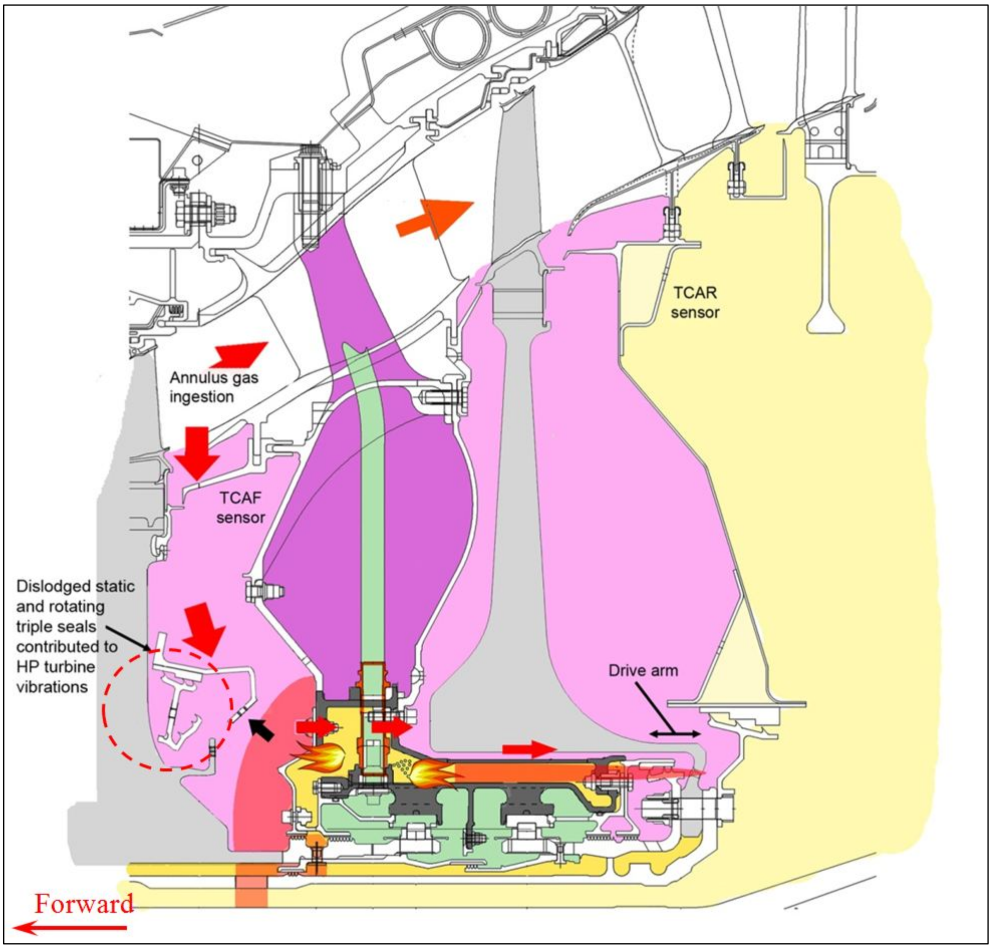


Image modified from a Rolls-Royce supplied model

When the hot gases were drawn out of the annulus gas stream, the performance of the engine changed, with the HP assembly speed (N3) increasing and the IP and LP assembly speeds (N2 and N1) decreasing. This ‘rematching’[78] of the assemblies resulted in the engine control system mode changing to limit the HP system speed, and a slight decrease in fuel flow was observed.

78 The turbines operate independently and find a steady state for the current conditions. When the conditions change, the turbines ‘rematch’ to a new steady state.

3.3 Phase 3: Drive arm heating and disc separation from the drive shaft

The change in the pressure distribution inside the engine resulted in the primary airflow within the buffer space becoming disrupted. Openings in the hub at the rear of the buffer space directed intense radiant heat and flame onto the triple seal at the rear of HP/IP bearing chamber. The triple seal failed and the flame was then directed onto the IP turbine drive arm at a point corresponding to the location of the R850 holes.

The IP turbine drive arm rapidly heated in the area of the R850 holes, reducing the strength of the material. The flame also eroded areas of the drive arm around the holes, further reducing its strength and, at 02:01:07 the drive arm failed in overstress, separating the IP turbine from the IP drive shaft. As a result of the air pressure loads across the turbine, the unrestrained turbine moved rearwards and contacted the LP turbine front panel (Figure 60).

Figure 60: Drive arm heating and disc separation from the drive shaft

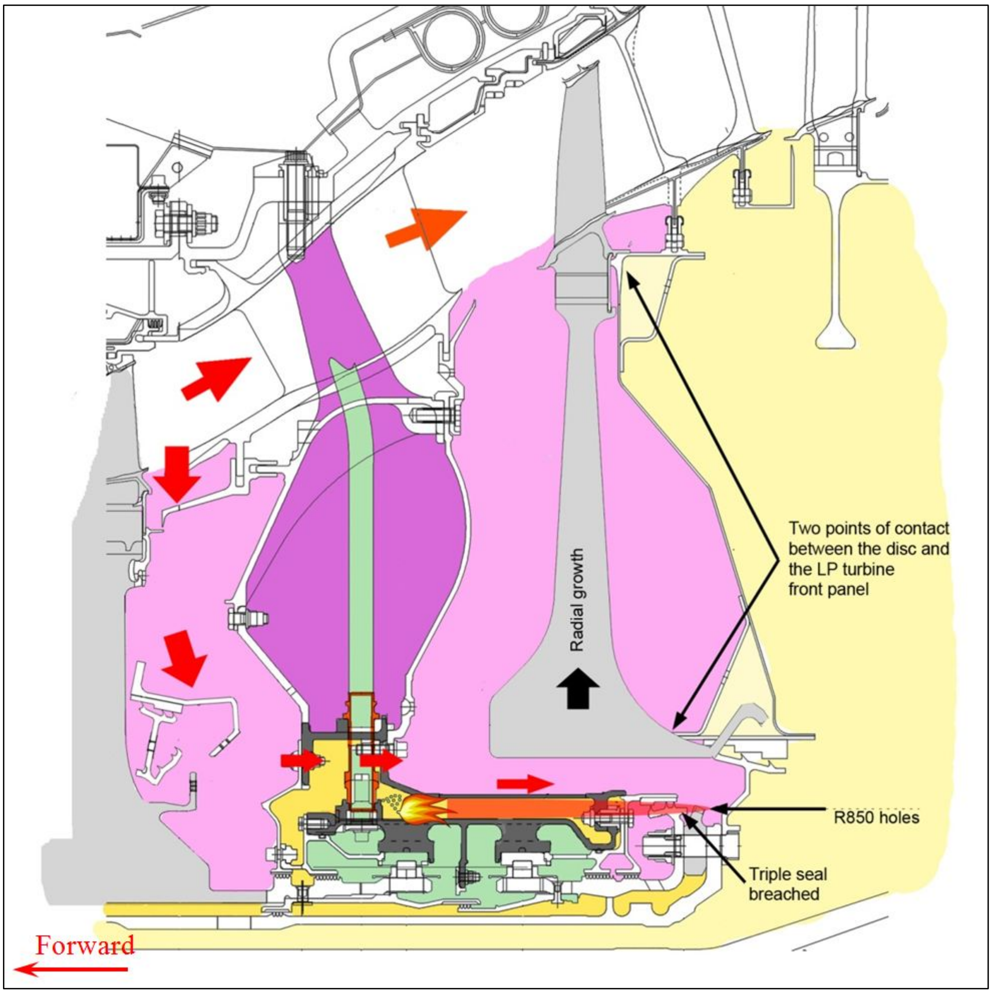


Image modified from a Rolls-Royce supplied model

When the IP turbine disc separated from the drive shaft, the IP compressor speed rapidly decreased and the engine surged, leading to a rapid reduction in the HP compressor delivery pressure (P30). The engine control system detected the surge and significantly reduced the fuel flow to the engine.

3.4 Phase 4: Disc acceleration and burst

Immediately after the IP disc separated from the drive shaft there was a rapid reduction in the IP and LP compressor speeds, but only a small reduction in the HP assembly speed, which stabilised at about 96%. Together with the recorded parameters for fuel flow and the higher than expected HP compressor delivery pressure following the surge, this indicated that the HP system had at least partially recovered from the surge.

Although the pressure in the engine rapidly decreased, there was still sufficient air flow to accelerate the unrestrained IP turbine disc. With the partial recovery of the HP system, the pressure of the air passing through the IP turbine was greater than expected. The friction between the disc and the LP front panel, and the loss in efficiency from the IP turbine blades as the disc moved rearwards out of the normal airflow, was insufficient to prevent the disc from accelerating past its maximum design rotational speed. As the rotational speed increased, the stresses within the IP disc increased, it expanded radially and at 02:01:11, 4 seconds after its separation from the drive arm, the disc material failed in overstress and the disc burst (Figure 61).

Figure 61: Unrestrained IP turbine disc acceleration and burst

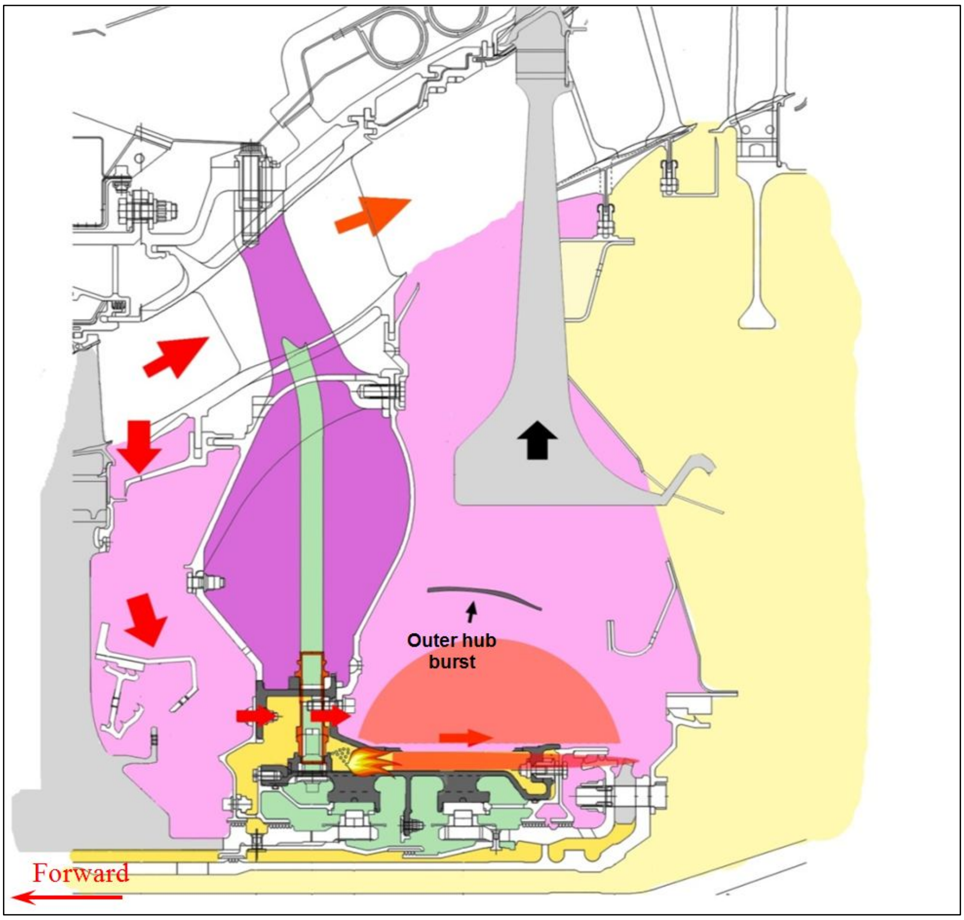


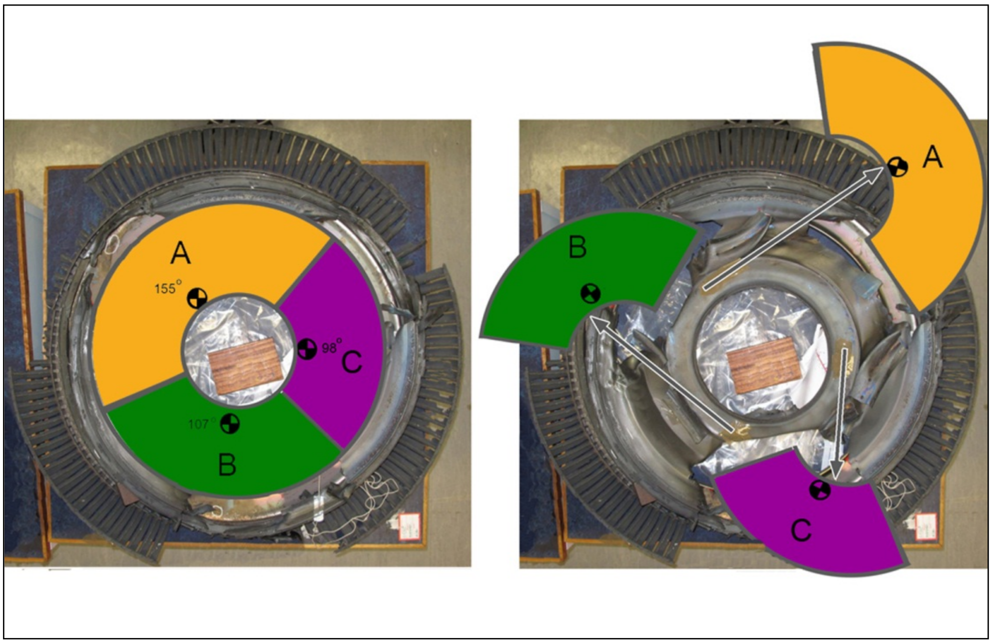
Image modified from a Rolls-Royce supplied model

The disc fractured into three primary segments that were projected outwards with a force sufficient to breach the engine casing and damage the aircraft. The damage to the LP turbine front panel and LP turbine blades indicated that these segments were initially projected outwards in different directions (Figure 62).

When the turbine case was breached the pressure within the IP disc space dropped to atmospheric pressure. That drop in pressure likely resulted in the heat-weakened HP/IP bearing chamber outer hub bursting (Figure 40 and Figure 61).

Associated with the uncontained failure was a further drop in P30 and the speed of the HP system decreased to about 40%. The TCAR reading rapidly dropped when the wire loom was damaged and the engine oil pressure and quantity began to decrease when the supply lines were severed.

Figure 62: Initial disc segment trajectories, looking rearward



Note: Segment A was recovered from Batam Island. Segments B and C were not recovered. The size of segments B and C were estimated from turbine blade fragments that were recovered from the aircraft, Batam Island and from the damage to the LP front panel.

3.5 Phase 5: Post-uncontained failure

At 02:01:38 (29 seconds after the disc separated from the drive shaft), in response to the ECAM turbine overheat indication, the flight crew reduced the thrust commanded on the No. 2 engine to idle. From 02:01:53, the flight crew incrementally advanced the thrust lever. The recorded data showed that the engine responded to that command with an increase in P30 and HP speed, indicating that the engine continued to operate following the disc burst.

The HP speed and P30 continued to increase, until at 02:02:20 the engine surged again. At 02:03:21, the flight crew shut down the No. 2 engine.

4 FACTUAL INFORMATION: MANUFACTURING

This Part details the design and manufacture of the Trent 900 High Pressure/Intermediate Pressure (HP/IP) structure and its role in the development of the uncontained engine failure. The information is presented in a semi-chronological order starting at the design definition, working through the development of the manufacturing processes and ending with the production of engines with non-conforming oil feed stub pipes. The production of the component that was directly involved in the uncontained engine failure is examined, together with the opportunities to detect and manage the affected engines. Finally, an indication of the extent of non-conforming oil feed stub pipes within the Trent 900 engine fleet is discussed.

In this context, the applicable procedures used by the manufacturer, for example 'non-conformance management', are presented to assist with the reader's understanding. Because the entire sequence leading to the uncontained engine failure occurred over a number of years, a timeline of key events discussed in this report is contained in Appendix E.

4.1 Company structure

Rolls-Royce plc, a large design and manufacturing organisation, produced powerplants and associated services to the civil and defence aerospace, marine and energy industries. The manufacturer was divided into a number of business divisions reflecting the industries they serviced.

The Trent 900 engine program was part of the manufacturer's Civil Aerospace Division and, while the program team was primarily based at Derby in the United Kingdom (UK), the manufacture of engine components was conducted in operating business units distributed across the globe. During the development of the Trent 900 and at the time of the manufacture of the event engine (engine serial number (ESN) 91045), the HP/IP support structure was manufactured in its plant at Hucknall in the UK.

In March 2007, the engine manufacturer reorganised, replacing operating business units with supply chain business units. These units were responsible for the design, manufacture and purchase of gas turbine components and sub-systems, differing from the previous operating business units that only manufactured parts. As a consequence of the reorganisation, the part of the Hucknall plant that manufactured the HP/IP support structure (Hucknall Casings and Structures) was transferred to the Transmissions, Structures and Drives supply chain business unit. The design function for this business unit remained at the Derby facility at that time.

4.2 Product development

The Trent 900 was a variant of the RB211 Trent family and although similar in many ways to other engines within family, had numerous differences in design detail. From the early stages of the design, a team was drawn together comprised of engineers from a number of disciplines, including structures, performance and systems, and was located at Derby.

An iterative design process was carried out primarily by a team of design engineers, but included input from all functions that had a role in the design and manufacture of the parts that comprised the engine. The process culminated in a set of drawings that defined the engine (design definition) being formally accepted by manufacturing.[79] This acceptance confirmed that any manufacturing issues associated with the design had been reviewed and, where necessary, corrected. The Trent 900 design definition was frozen when the engine was certified in 2004.

Concurrently during the manufacturing acceptance of the design definition, a number of manufacturing engineers commenced work on the manufacturing process specification and instructions. This included the production of a set of manufacturing stage drawings and manufacturing instructions that were completed and approved after the design definition had been approved. The process specification and instructions were developed with the intention of producing the components specified in the design definition. The manufacturing stage drawings broke the components down into a number of manufacturing process steps detailing their manufacture.

Before the production of items for testing and service, the manufacturing process specification and instructions were verified by a *first article inspection*. The intent of this inspection was to check the first item produced to the process specification for conformance with the design definition. In the case of the HP/IP bearing support structure, the manufacturing acceptance, production of the manufacturing process specification and instructions, and the first article inspection were carried out at Hucknall.

4.3 Design definition

The design definition for the prototype HP/IP bearing support structure was completed in September 2004. The design definition drawings contained the detail of the entire support structure, including the bearing hub, strut ring and service pipes. Due to the complexity of the structure, there were around two to three thousand individual features[80] specified in the design definition. A small number of those features related to the fitting of the oil feed stub pipe. The stub pipe, which was common to all the service pipes that penetrated the outer hub, was detailed in a separate drawing.

The inside diameter of the oil feed stub pipe was enlarged to accommodate an integral filter, as described in section 2.1.1 of this report. This enlargement, referred to as the oil feed stub pipe counter bore, was detailed in the design definition drawings. A number of features relating to the oil feed stub pipe are referred to in this report. This includes the outer hub clearance hole, the interference bore, the inner hub counter bore and the oil feed stub pipe counter bore, all of which are presented in Figure 63. The oil feed stub pipe counter bore consisted of two primary features; a standard tolerance and a high-tolerance (fine limit) counter bore.

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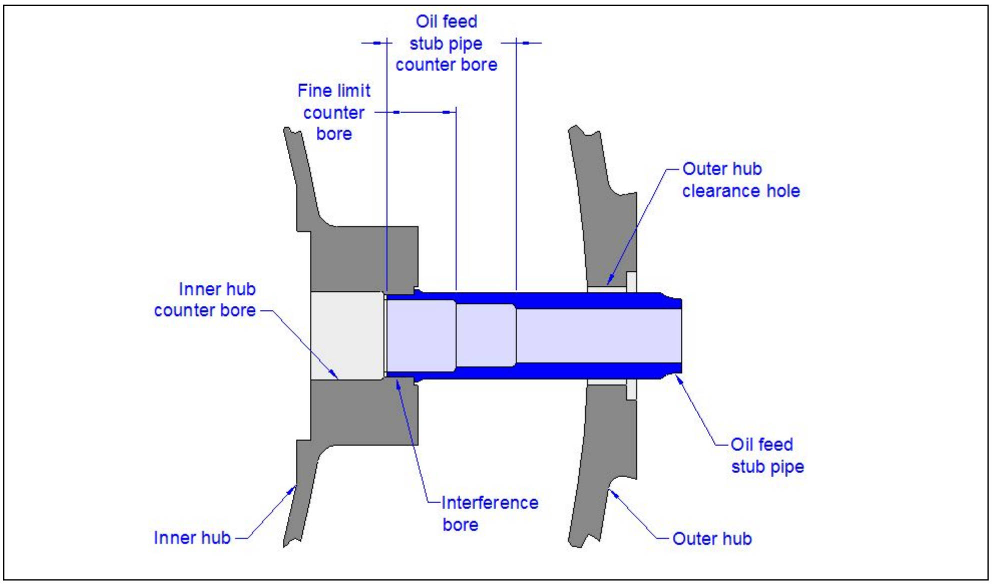
79 Prior to final approval and release of the design definition a check was required to verify that the design was optimised for robustness against all technical requirements, including the ability to manufacture and inspect.

80 Specific items, such as hole diameters and positions.

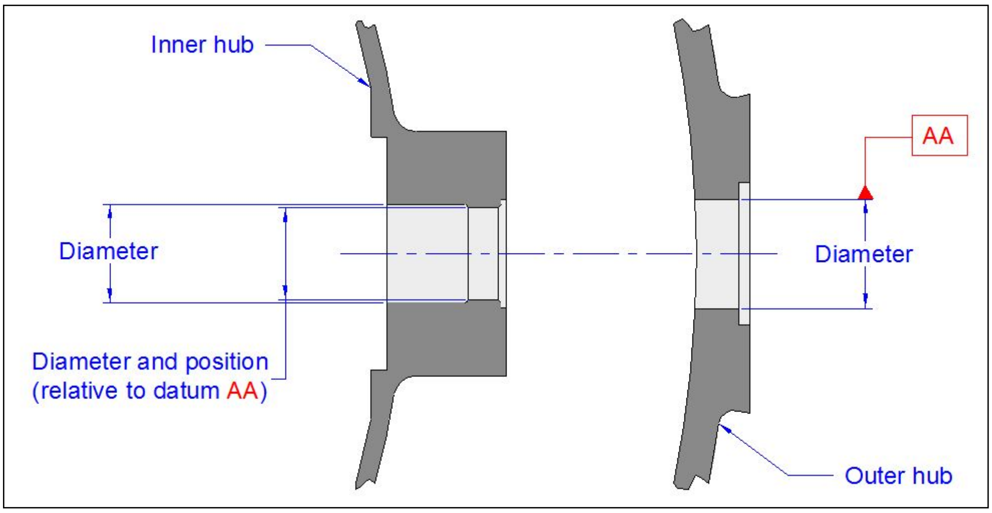
Figure 63: Oil feed stub pipe feature terminology



The oil feed stub pipe features were distributed across two sheets of the design definition drawings. The first of those sheets contained the detail of the outer hub clearance hole, the interference bore and the inner hub counter bore. The second provided detail of the oil feed stub pipe counter bore.

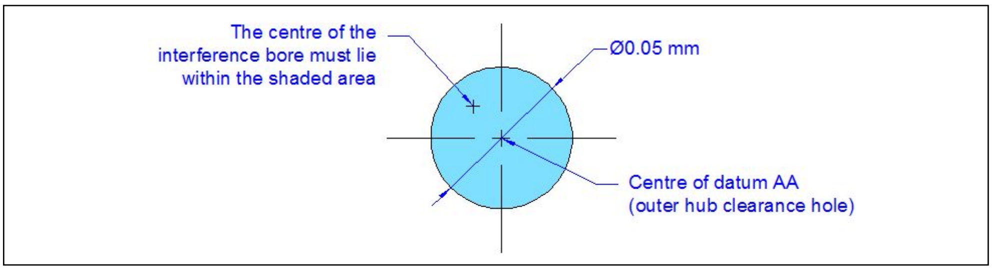
To define the relative locations of related features, a set of datums was used on the design definition drawings. The primary datum for features relating to the oil feed stub pipe was the outer hub clearance hole, which was identified on the design definition drawings as AA (Figure 64).

Figure 64: Representation of the design definition drawing that identified datum AA



The dimensional information for the oil feed stub pipe interference bore included a geometric positional tolerance that required it to be positioned with respect to datum AA to within a diameter (∅) 0.05 mm circle. This meant that the centre of the interference bore was required to lie within a 0.05 mm diameter circle centred on the centre of datum AA (Figure 65).

Figure 65: Physical interpretation of ‘geometric positional tolerance’ using the interference bore as an example



The size of the inner hub counter bore was defined and was shown on a common axis with the interference bore and datum AA, but there was no positional tolerance applied to the inner hub counter bore.

The design definition drawing sheet that detailed the oil feed stub pipe counter bore included the following information (Figure 66):

- the diameter of the fine limit counter bore
- a positional tolerance of ∅0.10 mm for the fine limit counter bore with respect to datum AA
- the overall depth of the counter bore (including the standard tolerance) – shown as a distance from the axis of the hub
- the minimum depth of 16 mm for the fine limit counter bore
- the maximum step of 0.25 mm between the surface of the fine limit counter bore and the standard tolerance counter bore (‘max step’ feature).

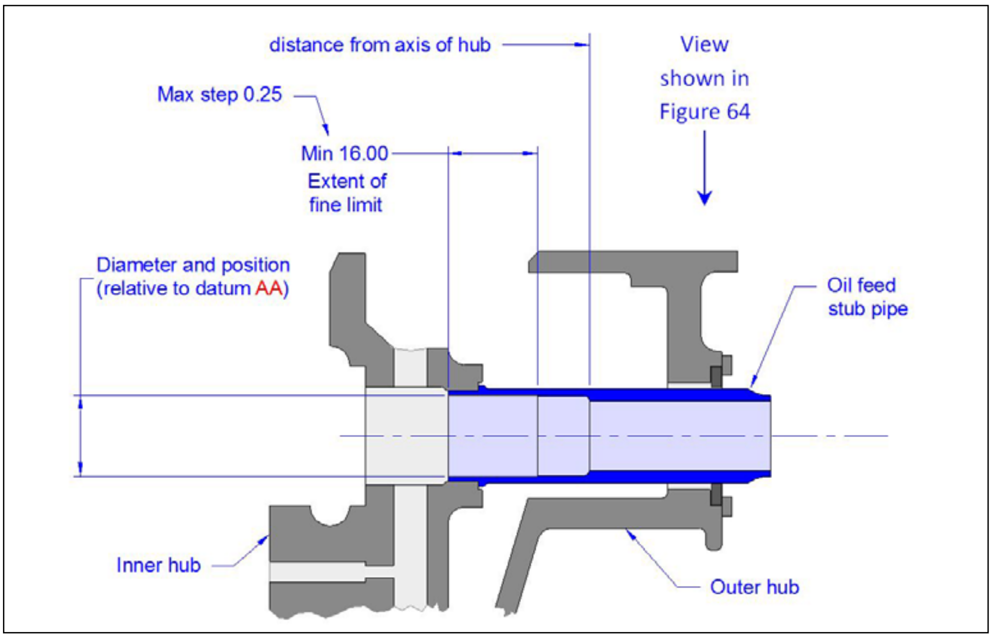
The diameter of the standard tolerance counter bore was not specified on the drawing.

Observations:

The wall thickness of the oil feed stub pipe at the counter bore was not an explicit feature in the design definition. The design intent was that the interference bore and oil feed stub pipe counter bore would be concentric with the outer hub clearance hole (datum AA), thereby providing the pipe wall thickness. Accepting that in any normal manufacturing environment such features could not be expected to be made in perfect alignment, the designers applied both dimensional and positional tolerances to the interference bore and oil feed stub pipe counter bore to ensure that an acceptable wall thickness would be attained.

The ATSB determined that the nominal wall thickness of the oil feed stub pipe within the counter bore was 0.91 mm. Using the ‘worst case’ tolerance combination of the interference bore at its smallest size, the oil feed pipe counter bore at its maximum size and the positional tolerances at their extreme values in opposite directions, the minimum permissible wall thickness was calculated to be 0.82 mm.

Figure 66: Representation of the design definition drawing that defined the oil feed stub pipe counter bore



Note: the above view is from the side of that shown in Figure 64, consistent with how it was depicted on the design definition drawings.

Manufacturing acceptance of the design definition drawings

The engine manufacturer reported that, when the design definition drawings for the HP/IP support structure were accepted by manufacturing in October 2004, the reviewing ME identified a number of issues that required changes to the design definition. However, none of those changes related to the oil feed stub pipe counter bore features.

4.4 Manufacturing specification and instructions

Following the manufacturing acceptance of the design definition and during the preparation of the manufacturing specification and instructions, the MEs identified that the oil feed stub pipe needed to be fitted and welded in place before the oil feed stub pipe counter bore could be machined. Because the pipe was in place, the inner surface of the clearance hole in the outer hub was not accessible to the machining or inspection probes. As a result, datum AA could not be used as a reference for further machining operations (Figure 67).

Observation:

There was no indication that the design or manufacturing personnel identified, either before or during the manufacturing acceptance of the design definition drawings, that datum AA would be inaccessible following fitment of the oil feed stub pipe into the hub assembly.

Figure 67: Inaccessibility of Datum AA with the oil feed stub pipe installed

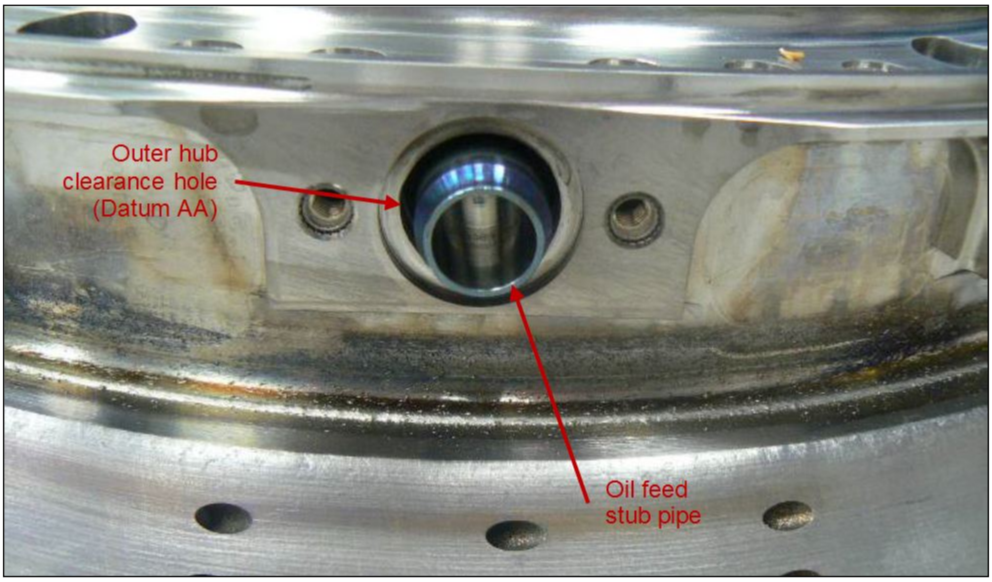


Image source: UK AAIB

During preparation of the manufacturing specification and instructions, a manufacturing datum was introduced that was intended to replace datum AA as the reference for the oil feed stub pipe features. The inner hub counter bore was selected as the manufacturing datum and was designated as datum M.

The ATSB found no evidence of consultation between the design and manufacturing engineers regarding the inaccessibility of datum AA and their development and use of manufacturing datum M. The manufacturer reported that at the time the Trent 900 manufacturing stage drawings were produced, there was no requirement in their procedures for such consultation if the use of the manufacturing datum maintained the design intent.

4.4.1 Manufacturing stage drawings

Similar to the design definition drawings, the oil feed stub pipe features were presented on the manufacturing stage drawings over two sheets. The first of those sheets detailed the outer hub clearance hole, the interference bore and the inner hub counter bore (Figure 68). The second of those sheets provided the detail on the oil feed stub pipe counter bore (Figure 69).

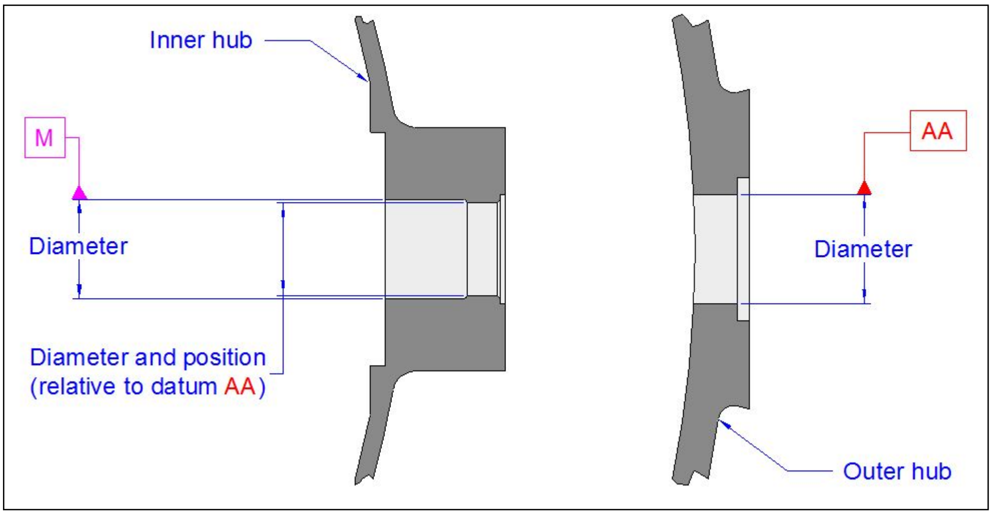
The positional tolerance for the interference bore differed from the design definition only in the magnitude of the tolerance, ∅0.5 mm rather than ∅0.05 mm. Of note, the tolerance was still identified as being with respect to datum AA.

The inner hub counter bore was defined, but there was no common axis with other oil feed stub pipe features presented on the drawing and, like the design definition drawing, there was no positional tolerance applied to the feature. On the same drawing sheet, the inner hub counter bore was defined as datum M. No other references were made to datum M on that sheet.

Observation:

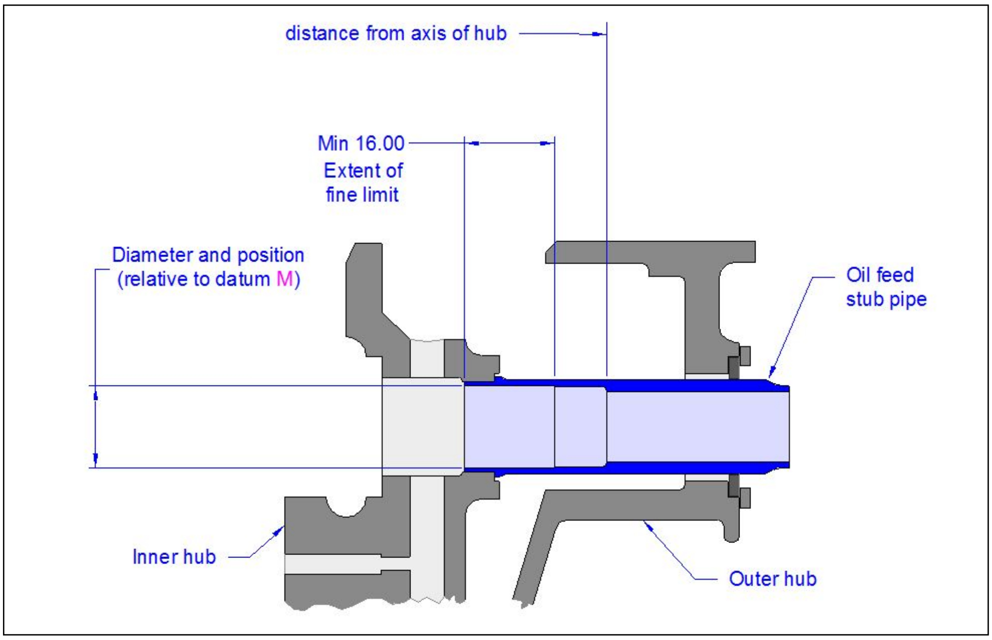
Because no positional tolerance was specified for the inner hub counter bore, datum M was not geometrically constrained. The result was that, according to the manufacturing stage drawings, there was no positive geometric connection between datum M and the interference bore.

Figure 68: Representation of the manufacturing stage drawing that identified datum M



The oil feed stub pipe counter bore detail on the manufacturing stage drawing differed from the design definition in that the positional tolerance for the feature was referenced to datum M, rather than datum AA. The positional tolerance on the drawing was specified as ∅0.2 mm, rather than the 0.1 mm specified on the design definition.

Figure 69: Representation of the manufacturing stage drawing that defined the oil feed stub pipe counter bore



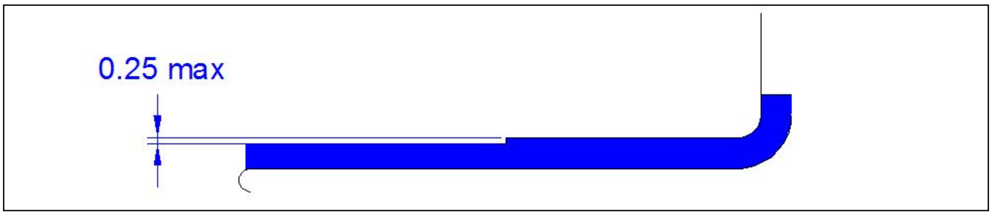
Observations:

There were significant differences between the sizes of the positional tolerances specified on the design definition and manufacturing stage drawings. The ATSB did not identify an explanation to account for those differences.

Because datum M had no geometric constraint with respect to the interference bore, and hence the location of the oil feed stub pipe, the minimum wall thickness for the oil feed stub pipe could not be determined using the specifications presented on the manufacturing stage drawings.

Similar to the design definition drawing, the oil feed stub pipe counter bore feature on the manufacturing stage drawings only had the diameter of the fine limit counter bore specified. The counter bore max step feature was also specified in a different manner to that in the design definition drawing. Rather than an annotation to the 'min 16.00 extent' dimension on the feature on the design definition drawing (Figure 66), the step was depicted in a separate enlarged view on the same drawing sheet (Figure 70).

Figure 70: Depiction of the max step feature on the manufacturing stage drawing



4.4.2 Manufacturing process

The determination of the manufacturing process started during the design stage, was concurrent with the production of the manufacturing stage drawings and was completed following the final approval of the design definition.

The manufacturing process for the HP/IP hub assembly was a combination of assembling parts, machining (milling and grinding), welding and inspection/test. The process was specified in a set of manufacturing instructions that broke the manufacture down into a series of operations (OPs) that were identified by their sequence number.

The manufacture of the HP/IP hub assembly was made up of 23 operations that ranged in sequence number from OP 10 to OP 230. Aspects of those operations involved in the misalignment of the oil feed stub pipe counter bore are described in the following.

OP 10 Join inner and outer hub castings

The inner and outer hubs were supplied as preformed items. In this first operation, the inner and outer hubs were bolted together using 3 of the 15 attachment lugs in the rear of the hubs (Figure 71).

Figure 71: Joining of the inner and outer hub castings

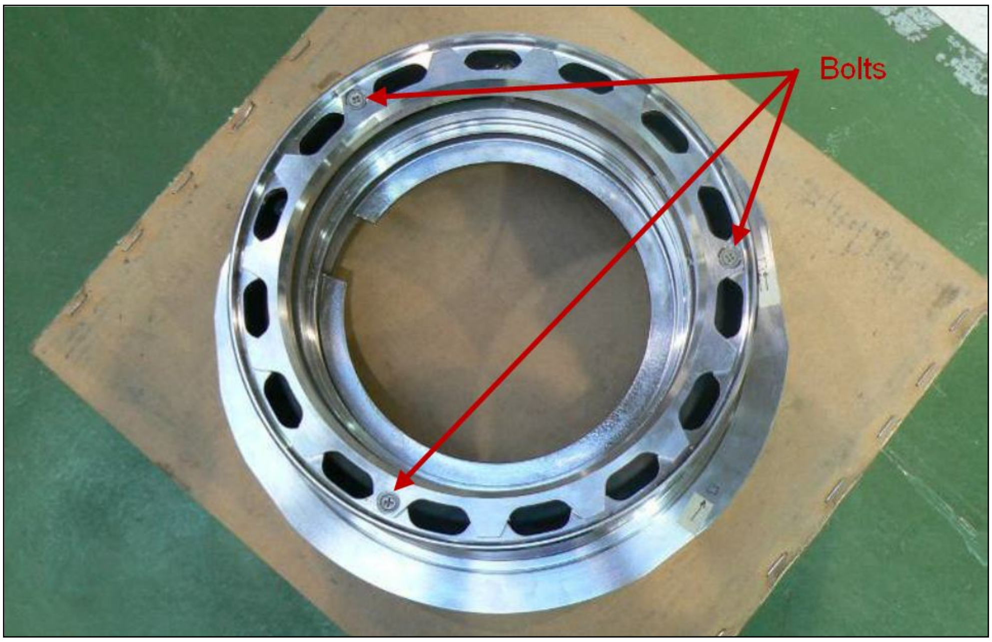


Image source: UK AAIB
View looking forward from the rear of the assembly.

OP 15 Machine air and oil pipe location bosses around the outer hub

The assembly was placed into a machining fixture that secured the assembly in place and provided a known reference to the computer numerical control milling machine. Although the majority of the features were yet to be machined into the hub, there were a number of features preformed into the hub that required the correct alignment of the hub within the fixture to ensure the features were machined in the correct location. To achieve this, the inner hub was supplied with a pre-drilled and reamed high-tolerance hole where the oil feed stub pipe was to be located. A pin was fitted into that hole and the assembly was oriented in the machining fixture to align the pin with the fixture. This pin was referred to as the 'timing pin' as it allowed the milling machine to orient itself with the direction of the assembly, very much like finding the 'twelve o'clock' position on a clock.

The manufacturer reported that the intention was for all of the oil feed stub pipe features, including the outer hub clearance hole, interference bore and inner hub counter bore, to be referenced from the axis of the timing pin, thus replacing datum AA in the manufacturing process.

Observations:

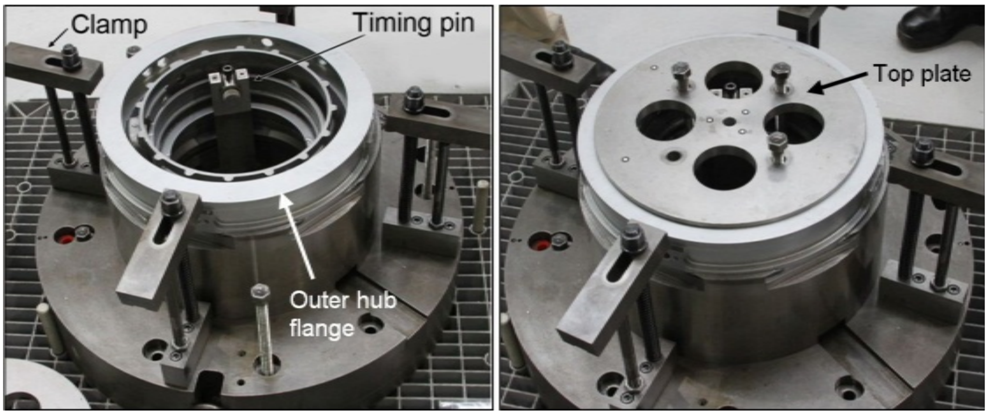
Although the intention was that datum M would replace datum AA, the inner hub counter bore had not been formed at this point and the timing pin effectively became the datum until the inner hub counter bore was formed. The datum formed by the timing pin was not listed on either the design definition or manufacturing stage drawings.

By using the timing pin as the reference for locating the interference bore, rather than referencing datum AA, the manufacturing process did not match the manufacturing stage drawings.

The assembly was then secured in the machining fixture using a set of clamps that secured the outer hub flange (Figure 72, left). The milling machine probed the component and the timing pin to orient itself with the assembly.

Before the milling operation started, the assembly was clamped using a top plate that clamped down on the inner hub (Figure 72, right) and the outer clamps removed. The outer surface of the hub was then machined by the mill to form the outside shape around the air and oil pipe positions (bosses). The milling operation included drilling and reaming of the clearance holes in the outer hub and the interference bores in the inner hub. Those operations were carried out from the outside of the hub.

Figure 72: Machining fixture clamping arrangement



The machining of the clearance and interference holes was a seven step process, but was essentially performed in four main stages as illustrated in Figure 73. Those stages consisted of:

1 drilling and reaming the clearance hole in the outer hub

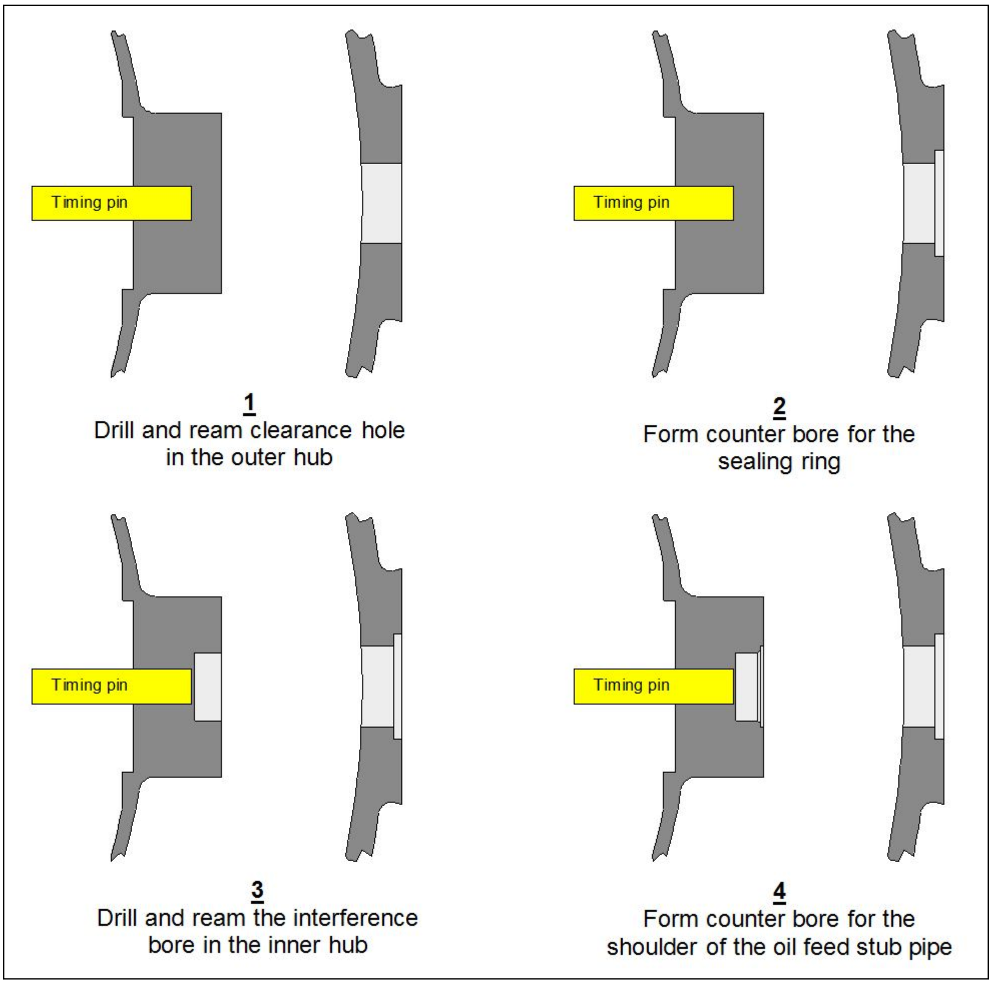
2 forming a shallow counter bore in the outer hub for the sealing washer

3 drilling and reaming the interference bore in the inner hub

4 forming a shallow counter bore in the inner hub to seat the shoulder of the oil feed stub pipe.

The machining fixture was then unloaded from the milling machine, leaving the HP/IP hub assembly in situ.

Figure 73: OP 15 formation of the oil feed stub pipe holes



OP 20 Machine oil pipe bosses inside the inner hub casting

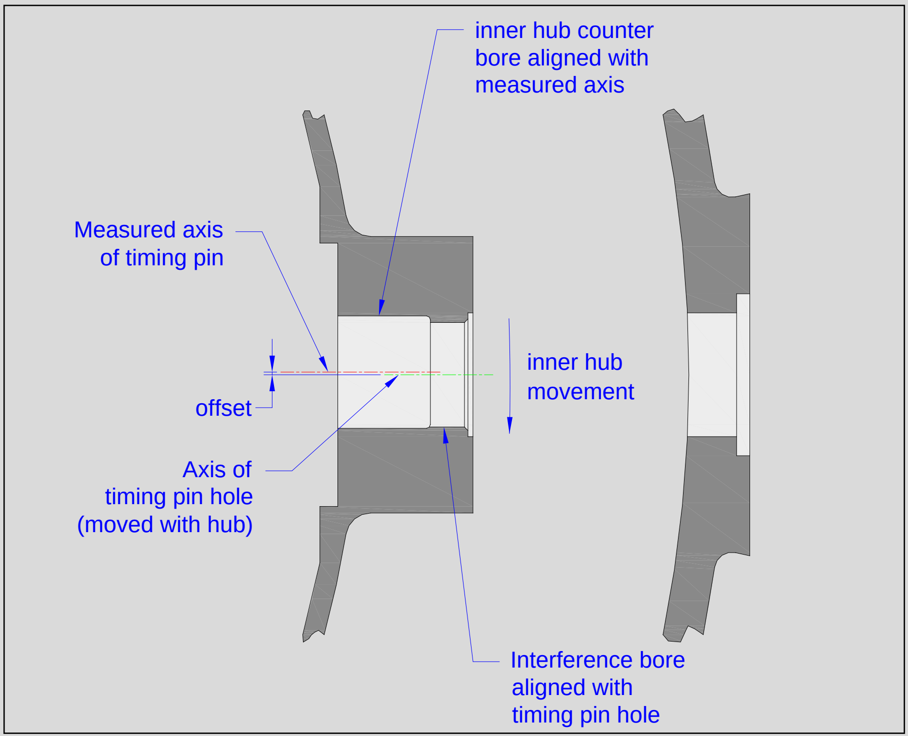
The machining fixture was then loaded into a different computer numerical control milling machine. The outer hub was clamped using the external clamps and the top plate clamp removed. A number of features, including the timing pin, were probed by the mill and the timing pin removed to provide access to the inside of the hub.

During operation OP 20, all of the machining was carried out from inside the hub. A number of features were milled into the inner hub over a number of hours before the inner hub counter bore was milled. The inner hub counter bore was formed by the mill's cutting tool transcribing a circle about the previously probed timing pin axis, and then the bore was reamed to the final size. This was intended to increase the size of the existing hole in the inner hub. It was expected that the location of the timing pin axis would remain aligned with the axis of the hole into which the pin had been inserted throughout the OP 20 process.

Observation:

During the investigation, it was found that after the change in the clamping arrangement and during the machining processes prior to forming the inner hub counter bore, there was movement of the inner hub. Because the machining of the inner hub counter bore was carried out several hours after the timing pin had been probed, any movement within the hub resulted in the inner hub being in a different position to what it was when the timing pin was probed. The result of any movement was that the inner hub counter bore was not machined on the axis of the hole in which the timing pin had been inserted and hence it was not concentric (the axis was offset) with the interference bore. The interference bore had been machined with the timing pin in place during OP 15 (Figure 74).

Figure 74: Example of inner hub counter bore/interference bore offset



At the completion of the machining, the HP/IP hub assembly was unloaded from the fixture.

OP 70 Dimensional inspection

After a visual inspection for burrs, damage and other obvious defects, the HP/IP hub assembly was loaded into a coordinate measuring machine (CMM).[81] This machine contained a computer-controlled program that automatically measured the hub's features and produced a report for the inspector to review.

As described in more detail in section 4.4.3 of this report, this inspection included measurement of the size and position of the inner hub counter bore, the interference bore and the outer hub counter bore. The position of the interference bore was measured and reported relative to the inner hub counter bore (datum M), rather than the outer hub clearance hole (datum AA) as listed on both the design definition and manufacturing stage drawings.

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[81] A machine used to measure the three-dimensional geometric properties of an object that may be either manually or computer controlled.

OP 110 and OP 120 Fit the stub pipes and weld

The stub pipes were inserted into the oil feed, vent and scavenge positions. The pipes were an interference fit with the inner hub and had to be tapped into position. After being checked to ensure their correct installation, the inner end of each pipe was welded into the hub.

> **Observation**:
>
> Ready access to datum AA was lost when the oil feed stub pipe was fitted during these actions.

A number of other operations were carried out, which included heat treatment and non-destructive testing.

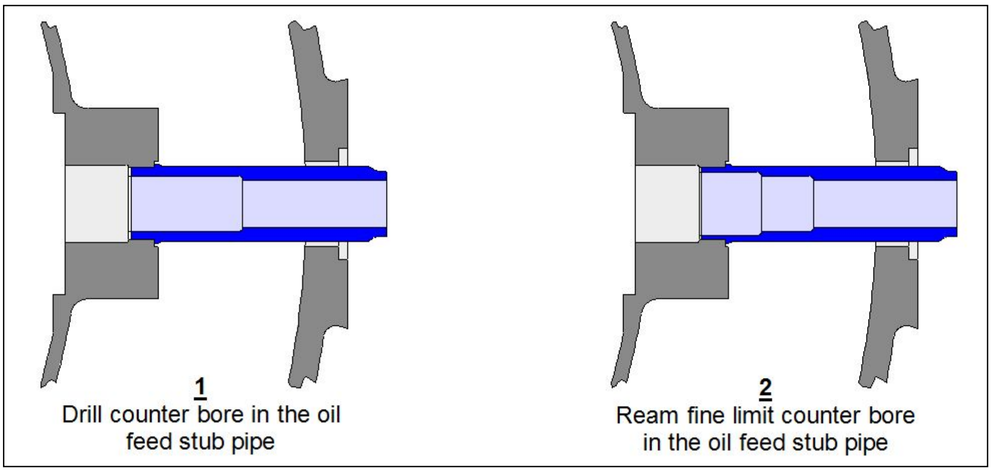
OP 190 Milling of the oil feed stub pipe counter bores

The HP/IP hub assembly was fitted into the machining fixture and the outer hub flange secured using the external clamps. A timing pin[82] was inserted into the inner hub counter bore to orient the hub in the fixture.

The hub and fixture were then loaded into a computer numerical control milling machine, which probed the hub and timing pin to orient itself with the assembly. Probing of the timing pin defined the position of datum M and aligned all subsequent machining to that axis. The timing pin was then removed to provide access to the oil feed stub pipe from the inside of the hub assembly.

During the milling operation, the oil feed stub pipe counter bore was drilled and reamed (see 1 and 2 respectively in Figure 75).

Figure 75: OP 190 oil feed stub pipe counter bore



Note: The wall thickness of the oil feed stub pipe is not shown to scale and has been exaggerated for clarity.

> **Observations**:
>
> Since the oil feed stub pipe has an interference fit in the inner hub, the outside diameter of the pipe is in the same position as the interference bore. Because the oil feed stub pipe counter bore was positioned relative to datum M, any misalignment of datum M from the interference bore created during OP 20 could result in a corresponding

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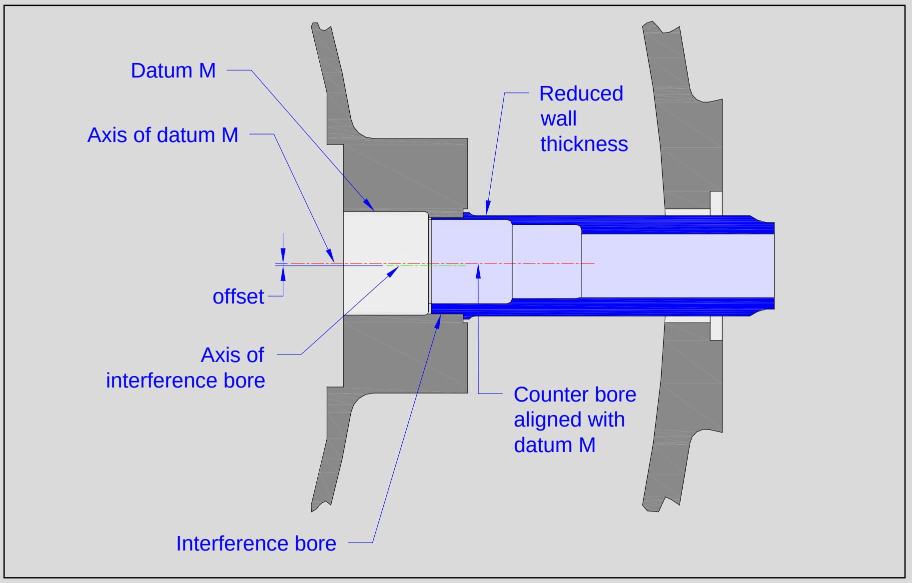
82 This was a different timing pin to that used in OP15 because the diameter of the hole had been enlarged.

misalignment of the oil feed stub pipe counter bore with respect to the outside diameter of the stub pipe. This would in turn result in a reduced wall thickness within the stub pipe (Figure 76).

Figure 76: Reduced oil feed stub pipe wall thickness due to misalignment



The engine manufacturer was unaware of the potential for a reduced oil feed stub pipe wall thickness until identified in March 2009. Detail regarding the identification of this and how it was managed is provided in sections 4.12 and 4.13 of this report.

OP 230 Final inspection

Finally, the HP/IP hub assembly was cleaned and visually examined before being loaded into a CMM for a dimensional inspection. That inspection included measurement of the oil feed stub pipe counter bore size and position relative to datum M. This is discussed in more detail in section 4.4.3 of this report.

The visual examination was only aimed at checking for swarf,[83] surface finish and other obvious defects. There was no requirement for the inspector to inspect the oil feed stub pipe counter bore for concentricity of the machining, as the process relied on the CMM measurements to check for conformity.

Observation:

When the end of a stub pipe was welded into place, the interface between the interference bore and the stub pipe was obscured by the weld material. Because the oil feed stub pipe counter bore was machined after the stub pipe had been welded into position, the wall thickness of the oil feed stub pipe could not be seen by the inspector. As such, even a significant reduction in the wall thickness would likely not be visually detected.

4.4.3 CMM dimensional inspections

The manufacturer's CMM was a computer-controlled precision measuring device consisting of a fixed table over which a moveable bridge was situated. A moveable arm, upon which a calibrated probe could be fitted, was supported from the bridge

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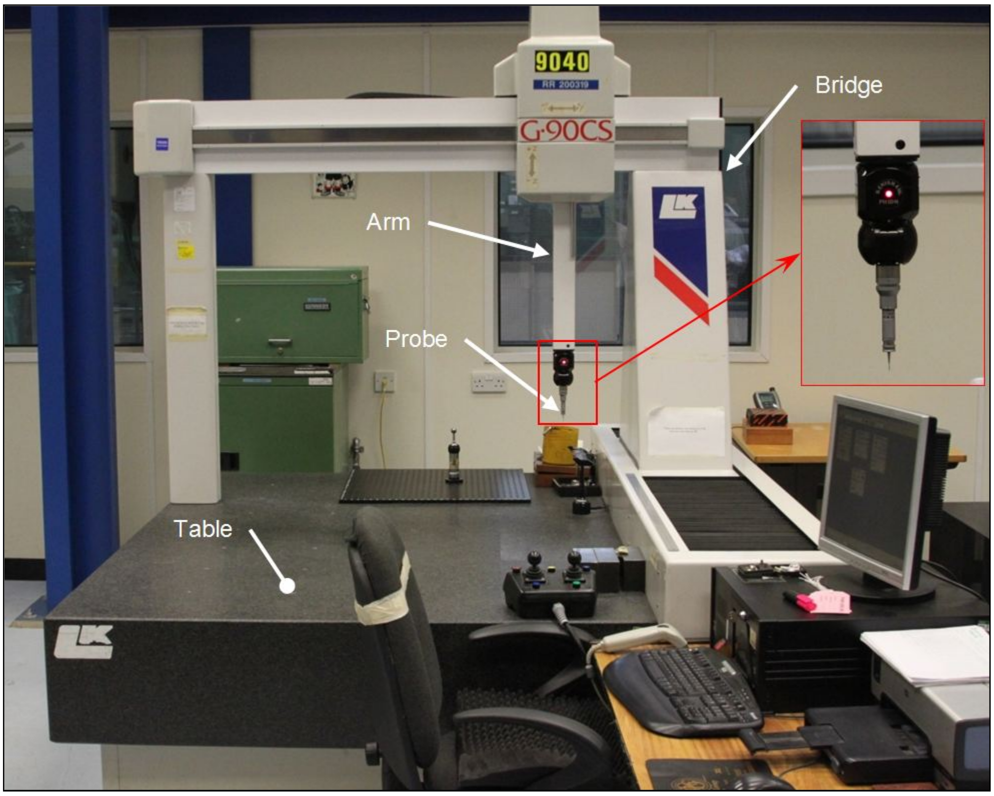
83 Chips of metal removed during the machining process.

(Figure 77). The probe could be either manually driven by an operator, or automatically driven by a pre-programmed set of movements through a computer.

The component to be measured was placed on the CMM table and specific features were measured by touching them with the calibrated probe. When the probe touched the surface, the location of the point in three-dimensional space was determined. Shapes could be measured by the collection of a series of points and their geometric properties, such as the diameter of a circle, provided to the operator.

Figure 77: Coordinate measuring machine



The CMM was used for two inspections during the manufacture of the HP/IP hub assembly, at OP 70 and OP 230. Due to the number of measurements to be taken, and to ensure consistency, the manufacturer used computer programs to control the machine for those inspections. Those programs also produced a report for review by an inspector.

The CMM report detailed the measurements taken of each feature during the inspection. This included their measured (actual) value, the nominal value, the difference between the actual and nominal values, the high and low tolerances related to the nominal value (if applicable) and an error statement (if applicable). An error statement was produced if the actual value was outside of the allowable tolerance from the nominal value. If there was no error, then the field was left blank. An example of a feature measurement is shown in Figure 78 with the error column highlighted.

The inspectors could print the report in either colour or black and white. When printed in colour, any error statements were printed in red. The printed report was reviewed by an inspector to identify any errors that could indicate a non-conforming feature.

Figure 78: Example feature from a CMM report (error statement highlighted in blue)

Feature - Circle21 DIM23

	Actual	Nominal	Difference	Hi-tol	Lo-tol	MMC	Error
AXIS Y	.066	.000	.066				
AXIS Z	-.061	.000	-.061				
DIAMETER	24.541	24.565	-.024	.065	-.065	--*+---	
T.P.RADIUS			.089	.025		.020	.044
T.P.ANGLE			317.282				
T.P.DIAM.			.179	.050		.041	.088

Note: In the above CMM report, T.P. DIAM is the 'true position' of the circle with respect to the current datum. In the case of a circle, its true position is twice the direct distance between its centre and the datum.

At the time that the Trent 900 went into production, and engine ESN 91045 was manufactured in June 2006, the CMM programs were written, checked and implemented by the inspectors within Hucknall Casings and Structures. There was no formal process for the creation and validation of CMM programs, but it was reported that an informal process was used that was based on the process for the creation and validation of the computer numerical control programs. That informal process was carried out at the inspector level and did not involve the manufacturing engineers.

In August 2007, a formal process for the creation and validation of CMM programs was introduced at Hucknall Casings and Structures. The process involved inspectors and manufacturing engineers, with the engineers responsible for, amongst other things, which features were to be measured.

The ATSB obtained printed extracts from the CMM programs and reports from the inspection of a number of Trent 900 HP/IP hub assemblies. The following discussion is based on a review of those programs and reports. A number of features were measured during the respective inspections, but only those associated with measurement of the oil feed stub pipe features are discussed.

OP 70 Inspection

During the OP 70 inspection, the CMM measured the inner hub counter bore and set it as the datum (datum M). All the following measurements were with reference to this datum.[84] The program then reported the diameter of the feature and compared it to the nominal size and allowable tolerance.

The next feature to be measured was the interference bore, the diameter and the true position of which were measured and reported. The positional tolerance in the program code at that time was ∅0.50 mm.[85]

The CMM then moved on to measuring the clearance hole in the outer hub, measuring the diameter and reporting its true position. Both the program and example report showed that the CMM was programmed with a positional tolerance of ∅0.20 mm.

84 The CMM maintained a datum until it was redefined.

85 In December 2007, to better reflect the intent of the design definition, the CMM program was revised to reduce the positional tolerance to ∅0.05 mm.

Observations:

The datum used by the CMM did not reflect the datum referenced on either the design definition or manufacturing stage drawings, but was consistent with the intention for a manufacturing datum that replaced datum AA.

The true position of the inner hub counter bore (datum M) was not specified relative to any datum on either the design definition or manufacturing stage drawings. It appeared that the positional tolerances that were programmed into the CMM were obtained from the manufacturing stage drawings, because the values corresponded to the positional tolerances specified for those features, but the reference datum changed to datum M.

The position of the interference bore was neither measured, nor calculated, relative to datum AA as required by the design definition and manufacturing stage drawings. Although this would not have assisted with the determination of the resulting oil feed stub pipe wall thickness, it showed that the inspection method did not provide verification, on an ongoing basis, of compliance with the design definition.

OP 230 Final inspection

During the OP 230 final inspection, datum M was defined as the reference datum for the measurement of features in the oil feed stub pipe. The fine limit counter bore in the oil feed stub pipe was measured and the diameter and true position reported. The positional tolerance for the counter bore in the CMM program was ∅0.2 mm, which was consistent with the manufacturing stage drawing, but not the design definition.

The CMM program did not include the measurement of a number of design definition features relating to the oil feed stub pipe counter bore. Those features were the:

- depth of the fine limit counter bore
- overall depth of the standard tolerance counter bore
- diameter of the standard tolerance counter bore
- max step feature.

Observations:

Detailed stress analysis carried out as part of the ATSB’s investigation identified that the critical oil feed stub pipe characteristic was the wall thickness in the region of the fine limit counter bore (refer to section 2.5 of this investigation report). Although the oil feed stub pipe features measured by the CMM program did not reflect either the design definition or manufacturing stage drawings, the features that were best suited for calculating the wall thickness were measured.

Because the true position of both the interference bore and the oil feed stub pipe counter bore were measured from datum M (at OP 70 and OP 230, respectively), a relationship between them could be determined. Had the CMM program reflected either the design definition or manufacturing stage drawings at OP 70, relative movement between the inner and outer hubs and the use of different datums for each measurement would have precluded such a relationship. Thus, the CMM program reflected the design intent better than the manufacturing stage drawings.

Although these measurements were taken, they were not used by the engine manufacturer to determine the oil feed stub pipe wall thickness during the production of the HP/IP hub assemblies.

Using the relationship generated by the CMM program measurements, the ATSB calculated that the minimum oil feed stub pipe wall thickness resulting from the manufacturing process, which could pass the CMM inspections, was 0.545 mm.

4.5 First article inspection

The first article inspection of an assembly, or component, seeks to ensure that the manufacturing specifications and processes are capable of producing an item that conforms to the design definition. This inspection process typically involves the measurement of all of the features specified on the design definition drawings and their comparison with what was specified.

4.5.1 Manufacturer's first article inspection procedures

The engine manufacturer's procedure for the conduct of first article inspections stated that:

> The purpose of the First Article Inspection is to provide objective evidence that all engineering design and specification requirements are properly understood, accounted for, verified and documented.

and defined the first article inspection as:

> A complete, independent, and documented physical and functional inspection process to verify that the prescribed production methods have produced an acceptable item as specified by engineering definition, planning, purchase order, engineering specifications, electronic model and/or other applicable documents e.g. manufacturing definitions as agreed by the IPT.[86]

The procedure also noted that the first article inspection will provide confirmation that:

> - all the necessary documentation, equipment and controls are available and adequate for ongoing series production of the part,
> - the design and manufacturing procedures and processes are adequate and they have been fully utilised to produce the part, and
> - 'closed loop' verification that the part, as produced by the method examined during this activity, fully meets the design intent.

There were no specific instructions in the procedure to describe how the inspector was to determine the 'design intent'. The procedure was aimed at showing that design intent was achieved by confirming that the 'design characteristics' of the part conformed to the design definition. The procedure defined design characteristics as:

> Those dimensional, visual, functional, mechanical, and material features or properties, which describe and constitute the design of the article and can be measured, inspected, tested, or verified to determine conformance to the design requirements…

Associated with the manufacturer's procedure was a first article inspection report, which recorded the first article inspection and contained a checklist to facilitate the inspection. One item on that checklist required the inspector to answer (emphasis added):

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[86] IPT – 'Integrated program team'. A team that 'consists of design, manufacturing engineering, manufacturing laboratory, process operators, manufacturing management and other relevant people.'

> Have all final engineering **definition requirements** been satisfied? (Dimensional, notes specifications etc.)

The procedure defined a 'definition requirement' as:

> Requirements of the definition (including parts lists), specifications, electronic models or purchasing documents to which the article is made, including notes, specification, and lower-level definitions thereby invoked.

The procedure also noted that ‘Observations, such as lack of drawing clarity, should be included’ as an unsatisfactory item in the report.

4.5.2 HP/IP hub assembly first article inspection

The pre-production first article inspection of the HP/IP hub assembly was carried out by personnel at Hucknall Casings and Structures on 26 May 2005. The signatures on the first article inspection report cover page showed that the integrated program team lead was a manufacturing engineer. During the inspection, the inspector annotated a set of drawings with the measurements taken, and attached those drawings to the report. These drawings were predominantly the manufacturing stage drawings.

The manufacturing stage drawing that contained the detail of the interference bore in the inner hub was annotated with the inspector’s measurements. Both the diameter of the interference bore and its position, with respect to datum AA, were measured. As per the drawing, only the diameter of the inner hub counter bore was measured.

The first article inspection report also included the design definition drawing that specified the oil feed stub pipe counter bore. However, there were no measurements on that drawing, only references to the corresponding manufacturing stage drawing. The annotations on the manufacturing stage drawing showed that the size of the fine limit counter bore and its position with respect to datum M were measured and within limits. The max step feature was also measured and found to be within limits.

Observations:

The use of the manufacturing stage drawings in the first article inspection of the HP/IP hub assembly showed that the manufactured part conformed to those drawings. It did not show that the manufacturing processes were capable of producing an item that conformed to the design definition, as was the intent of the first article inspection.

Because the relative location of datum M was not measured with respect to either datum AA or the interference bore, the ATSB was unable to determine the minimum wall thickness for the assembly that was examined in the first article inspection. As such, it could not be determined from the measurements taken if the first article met the design intent.

Features that were missing from the CMM program, such as the ‘max step’ feature, were measured during the first article inspection, indicating that the measurements were taken manually, rather than from the automated CMM program.

4.6 UK CAA Regulatory oversight

The legislation that provided European Union National Aviation Authorities (NAA), such as the United Kingdom Civil Aviation Authority (UK CAA), their model for airworthiness regulation was EU Commission Regulations (EC) 1702-2003. Specifically, in the case of an engine manufacturer within the NAA's jurisdiction, Part 21 Subpart G titled 'Production Organisation Approval' and Subpart J titled 'Design Organisation Approval'. Both subparts establish the procedures for approval issuance through ensuring conformity of products, parts and appliances with the applicable design data and set out the rules governing the rights and obligations of holders of such approvals.

In exercising the privileges of these approvals, the approvals holder must ensure they perform certain functions as set out in the exposition[87] they submit to the NAA. One of the requirements of the exposition is a submission of a description of the approval holder's quality system and procedures. The requirements for the quality system are defined within Part 21 as:

> The production organisation shall demonstrate that it has established and is able to maintain a quality system. The quality system shall be documented. This quality system shall be such as to enable the organisation to ensure that each product, part or appliance produced by the organisation or by its partners, or supplied from or subcontracted to outside parties, conforms to the applicable design data and is in condition for safe operation, and thus exercise the privileges [of the approval].
>
> The quality system shall contain … an independent quality assurance function to monitor compliance with, and adequacy of, the documented procedures of the quality system. This monitoring shall include a feedback system to the person or group of persons nominated by the production organisation to ensure that the organisation is in compliance with the requirements of this Part, and ultimately to the accountable manager to ensure, as necessary, corrective action.

Under this model of regulation, the UK CAA established oversight of the engine manufacturer and confirmed compliance through monitoring of its quality assurance functions. This was achieved through a program of regular audits and surveillance, whereby the engine manufacturer's nominated personnel and accountable managers demonstrated to the UK CAA compliance with the terms of their exposition. Demonstration of compliance was achieved through presentation of evidence from a sample specified by the regulator. This program was designed to verify the effectiveness of the quality assurance function within the quality management system (refer to definition of quality assurance at 4.7.2).

The regulator's audits did not replace an internal auditing program within the production organisation's quality management system, but supplemented and tested that system. The regulator was not responsible for checking that individual parts conformed to the approved design, but rather checked that the quality management system had effective processes for identifying and managing product non-conformity.

The ATSB reviewed the role of the UK CAA regarding regulatory oversight of the manufacturer and determined that the UK CAA had maintained a regular audit and surveillance program which sampled functions and processes within the manufacturer's quality management system.

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87 A document that defines the organisation and procedures upon which the manufacturer's regulatory approval is based.

4.7 Quality management system

A manufacturing organisation's quality management system is a structured and documented system to direct and control the quality of an organisation's activities. A quality management system invariably comprises quality control and quality assurance processes. These are designed to satisfy all customer, regulatory and legal requirements with regard to quality standards in manufacturing.

4.7.1 Quality control

Quality control is that part of quality management focused on fulfilling the quality requirements.[88] It is a system of technical activities whose purpose is to measure and control the quality of a product or service so that it meets all customer, regulatory, legal and business requirements. It includes complete or approved sample inspection of a product prior to dispatch or release to the next stage in the manufacturing line. Quality control can include the use of appropriate statistical methods to ascertain that a product falls within specified quality limits, with a predetermined degree of confidence based on sample size.

4.7.2 Quality assurance

Quality assurance is that part of quality management focused on providing confidence that quality requirements will be fulfilled. It relates to the capacity of an organisation to monitor its own activities, so that it may remain confident that its products fall within specified quality limits, with a predetermined degree of confidence.

Quality assurance relates to the establishment and maintenance of procedures to ensure the consistency of an organisation in the realisation of its product. The quality assurance process can traditionally be seen in the assessment of the documented procedures for completeness and accuracy, and the organisation's compliance with those procedures.

4.7.3 The manufacturer's quality management system

The engine manufacturer had a production organisation approval from EASA and also organisational approvals for ISO 9001:2008[89] and AS/EN 9100.[90] Approval to both these standards was via audit and oversight from an approved, independent third-party organisation. All of these approvals required the manufacturer to maintain a quality management system. Due to the size of the organisation, the manufacturer's quality management system comprised of a large number of individual processes and procedures covering their business divisions and range of

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88 British Standard BS EN ISO 9000:2005 Quality Management Systems – Fundamentals and vocabulary

89 ISO 9000 is a series of standards to assist organisations develop an industry accepted QMS. The set of standards were published by the International Organisation of Standardisation (ISO).

90 This standard was created by the International Aerospace Quality Group (IAQG), with representatives from companies in the Americas, Asia/Pacific and Europe. This standard incorporates the requirements of ISO 9001:2008 and several additional industry requirements and definitions.

products and services. Only those parts of the quality management system that were directly related to the uncontained engine failure were reviewed by the ATSB.

The quality management system processes and procedures of interest were contained within the group quality procedures. The procedures that were directly applicable to the uncontained engine failure were those for the first article inspection and non-conformance management. The applicable details from those processes are contained within this report.

4.7.4 Product quality controls

The engine manufacturer’s quality control for produced parts did not rely on random sampling, instead it was carried out individually on each part produced. This included the inspections and CMM measurements previously described.

To assist in the quality control of each item and in the higher level quality assurance processes, such as internal auditing, the quality management system required detailed record keeping. The primary quality control record for each assembly was the ‘batch card’. These cards included a sequential list of each of the operations required in the production of the part. That list was laid out as a table that included columns for information regarding the completion of each operation (Figure 79).

Figure 79: Excerpt from a sample HP/IP hub batch card

TIMES Op/Suffix/Keywd	Operation	Group/Path Book Op	Op Cert	Resource Type (Dept)	Qty	Oper No	Insp or Op Cert	Date Op Complete	SFDM Step
0001 MARSHL	HCASMARSHL- MARSHALLING OPERATION	Y		HCASMSHL					1
0010 FIT	3536FIT0602- FIT	Y		35360602					10

Any manufacturing documents or records associated with the compliance or non-compliance of the part were required to be retained. The period for which they were to be retained was defined in the group and local quality management system, depending on the document or record type. This could include inspection, correction, non-conformance and rectification work records.

4.7.5 The manufacturer’s quality assurance function

The manufacturer’s quality assurance function included procedures, processes and documentation to ensure the correct and accurate manufacture, release and recording of engines, sub-systems and components in accordance with aviation safety rules and regulations. Ensuring that staff adhered to procedures, conducted processes properly and documented their actions appropriately was also part of the quality assurance system.

The manufacturer had a systems-based auditing program. The program included assessments of documented procedures for compliance with minimum standards, and the comparison of organisational and individual behaviour against the documented procedures.

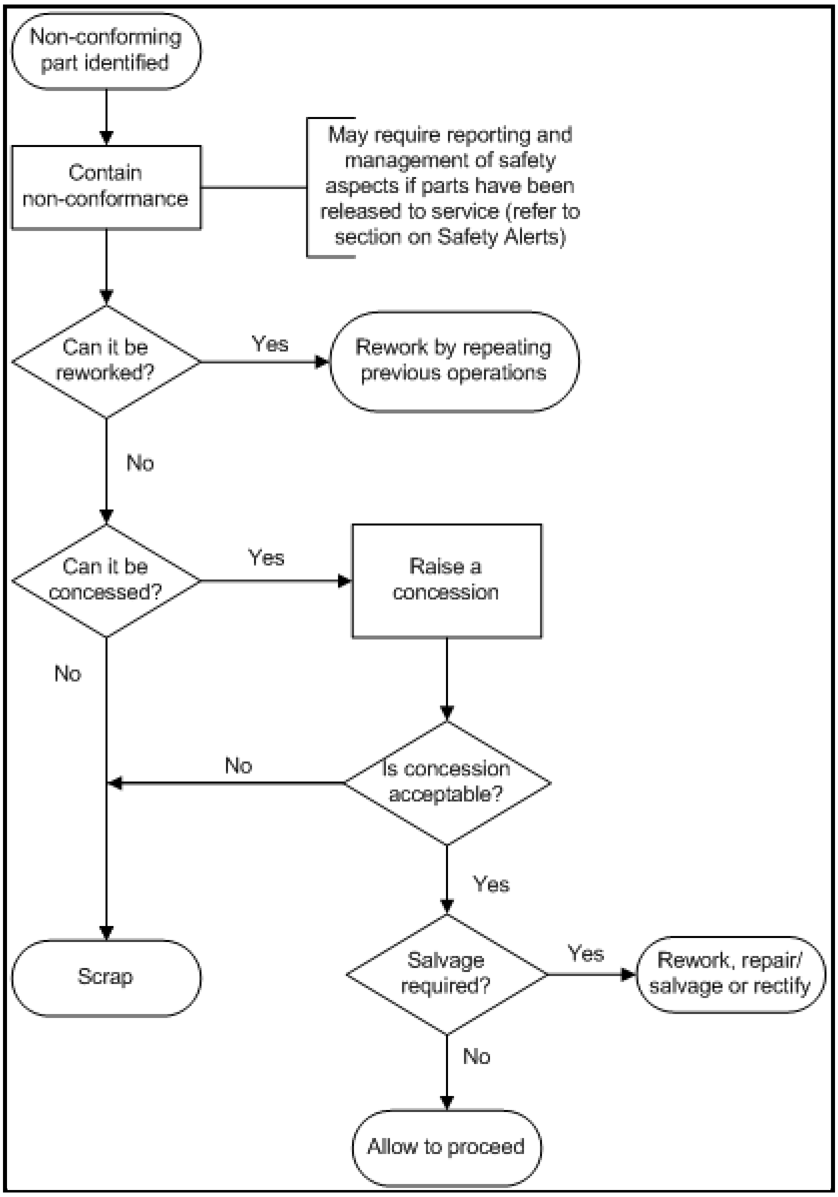
4.8 Non-conformance management

The manufacturer had a process for the management of non-conforming parts within their group quality procedures. A non-conformance was defined in the process as:

> Any aspect of a product which does not conform to the drawings, specifications or customer requirement.

When a non-conformance was identified, the process required that the non-conformance was recorded, contained (by segregation and/or marking to identify it as a non-conforming part) and sentenced.[91] The basic procedure for sentencing a part followed the process presented in Figure 80.

Figure 80: Basic non-conformance management process

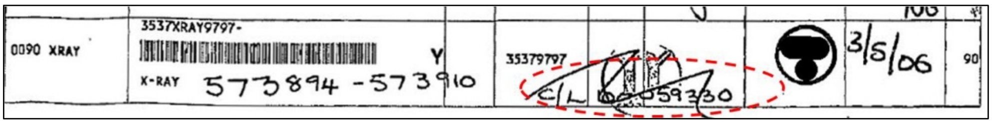


[91] A term used by the manufacturer where the Technical Authority determined that a non-conforming part was acceptable for use or required rectification or scrapping.

4.8.1 Identification of a non-conformance

A non-conformance could be identified at a number of stages during the manufacture of a part, but was most typically identified during one of the inspection operations (for example, OP 230 during the manufacture of the HP/IP bearing hub structure). When a non-conformance was identified, a document referred to as a CT17 was raised to record information relating to the non-conformance. Each CT17 was individually numbered and a reference to that number was required to be recorded on the non-conformance history for the part. The CT17 number was also to be recorded on the batch card in the row corresponding to the operation where the non-conformance was identified (Figure 81).

Figure 81: Example of CT17 reference annotation on batch card (highlighted with red dashes)



4.8.2 Containment of non-conforming parts

The management of non-conforming parts procedure included a section on the containment of parts that are found, or suspected to be, non-conforming. That section referred to the procedure for ‘problem resolution’ and included instructions on how to manage containment of the non-conforming product. The problem resolution process defined containment as:

> A series of activities to restrict the further effect of an identified problem.
>
> Containment involves:
>
> - The location and quarantine of problem product(s) or service(s)
> - A programme of customer protection:
>   - To prevent further delivery of the problem product(s) or service(s)
>
>   and/or
>   - To significantly reduce the likelihood of product(s) in-service impacting the customer
>
> Containment could also involve revised operating instructions for the product or service, where these actions are intended to reduce the effect of the problem until a full solution is introduced.

The manufacturer’s basic process for containment was to:

- define the problem
- prioritise the containment
- agree containment actions and timescales/ownership
- execute the agreed containment actions
- record and monitor those actions

The manufacturer's guidance material for the problem resolution process included the following items under the heading 'Contain the problem and protect the customer':

- Complete Containment Check Sheet ...
- Find and quarantine all affected parts already produced:
    - in transit
    - stores stock, despatch
    - despatched to customer
    - work in progress
    - at sub-suppliers.

Associated with the procedure was a '48hr Containment Check Sheet', which included tick boxes that prompted the required steps in the procedure. The first item on the check sheet was for the containment action owner to 'Identify and locate all suspect material / parts in the supply chain...'. A series of tick boxes was provided to assist in locating the suspect material (Figure 82). There was no tick box for parts that had left the direct control of the manufacturer, such as parts already released to the customer, or agent, for use on in-service engines.

Figure 82: Location tick boxes in the manufacturer's containment check sheet

In Transit		Work in progress	
Despatch / Shipping area		At sub-tiers	
Stores stock			

The problem resolution procedure made reference to the 'Resolve Customer Issues' process where 'external customers have been, or are likely to be, directly impacted by a problem...'. Problems identified by the manufacturer that could affect a customer included the release of non-conforming parts, referred to as 'escaped' parts.

4.8.3 Safety alerts

During the procedure for the control of non-conforming product, the process referred to a separate group quality procedure when '… there is potential Aerospace external impact'. That separate procedure, entitled *The identification/reporting and management of safety events and hazards for aerospace products*, required the manufacturer's departments to:

> ensure that any engine component, material or design condition which might be considered a safety issue is made known to the relevant Airworthiness department via the submission of a completed SAR.

The safety alert report (SAR) was 'fundamentally to identify and communicate safety concerns for sentencing as either an unsafe condition or for rejection.' If it was determined to be an unsafe condition, further action was taken to address it.

There were a number of instances where a safety alert report could be raised. This included in the case of an:

> Unapproved non-conformance that is subsequently identified to have been released via an Authorised release certificate ... and which will not be approved by a retrospective concession application will form the subject of a SAR.

Under the procedure, a safety alert report could be raised by any person within the organisation when they identified a potential 'safety issue'. When such a report was raised, it was submitted to a 'focal point' within the airworthiness department, for consideration. If the focal point considered that there was a potentially unsafe condition, the report would be sent to the Chief Engineer for sentencing. The Chief Engineer would then determine whether the issue constituted an unsafe condition. If an unsafe condition was identified, the Chief Engineer was responsible for initiating the appropriate containment and corrective actions, based on a fleet-wide risk assessment. If it was determined to not be an unsafe condition, the safety alert report would be formally rejected, together with reasons for the rejection.

Observations:

> The safety alert report process provided two levels of consideration as to whether the release of a non-conforming part constituted an unsafe condition before any actions were taken – the focal point and the relevant Chief Engineer. The Chief Engineer, who had a responsibility for the service of the fleet, provided a fleet-wide risk perspective to the decision and response to any identified unsafe conditions. However, this first required somebody within the organisation to identify that there was a potential safety issue and raise a safety alert report.

4.8.4 Concessions

Manufactured parts that did not conform to the product specification, and where rework of the part (by repeating the previous operations) would not result in conformance, were subjected to a 'concession' process. The purpose of the concession process was to determine if the non-conforming part was acceptable for use, or what work was required to either make it conform or to bring it to a state where the non-conformance was acceptable. That work could be in the form of rework, rectification or repair/salvage. The group quality procedure for the concession process defined these actions as follows:

Rework	Use of a sequence of approved operations in an attempt to return non-conforming product to a state that conforms completely to the drawing definition, specifications or customer requirements. This may entail the use of repeat standard router operations.
Rectification	Use of a sequence of approved operations that are not repeat router operations, in an attempt to make non-conforming product conform completely to the drawing definition, specifications or customer requirement.
Repair/salvage	Use of a sequence of operations to reduce, but not necessarily completely eliminate product non-conformance. The purpose of repair/salvage is to return non-conforming product to an acceptable condition, which may not completely conform to the original drawing definition, specification or contract requirement.

The procedure also noted that:

> The process described ensures that a concession application has the correct level of authority at all stages of its progress.

The manufacturer’s Training Guide for the concession process provided guidance on the completion of the concession form and defined a concession as:

> The authorisation to use, release or supply a limited quantity of a completed, or partially completed, product which does not comply with the specified technical requirements.

and stated that:

> Records of non-conformance shall clearly define the deviation to specification such that it will readily permit the authorising signatories to make judgement, allow technical audit and facilitate a reasonable understanding by non-specialists in the field (e.g. a customer). Any supporting data shall be referenced in a manner that permits ready recovery.
>
> All concession applications shall be submitted with sufficient background information (including Corrective Action Plans/Reports, root cause analysis and if the concessed part will be subject to First Article Inspection Report (FAIR)), to allow the authorising signatories to make a judgement.
>
> The concession application must register one part number per concession, except where action is necessary to cover raw material or metallurgical/ chemical processes, the final usage of which may involve a range of finished parts ...
>
> The concession application must never be used to circumvent another part of the [manufacturer’s] Quality Management System.

To ensure that a concession application was fully considered by all of the appropriate people, there were a number of signatures required. Those signatures included the person who raised the concession, the Non Conformance Authority, the Manufacturing Engineering Manager (or authorised nominee) and other functional areas. The Non Conformance Authority was defined as:

> ...individuals granted the authority to review and authorise non conformance in terms of fit, form and function of any level consistent with the scope of the authority granted (i.e. component/engine types/whole engine).

The Non Conformance Authority was responsible for sentencing the concession and determining which functional areas needed to be consulted in support of this judgment. An additional signatories sheet formed part of the concession and included a list of functional areas, with a number of blank lines below the list. The Training Guide noted that the list of functional area signatories was:

> ...not intended to be extensive, so if signatures are required of functions not on the list, then use the blank spaces provided.

The additional signatories sheet included spaces for the consulted functional area to annotate their decision regarding the concession, their categorisation of the concession and their signature.

Concessions were categorised according to the impact of the non-conformance. Appendix 1 to the group quality procedure provided information on the categorisation of concessions and defined three categories for production non-conformances as follows:

- Category 1 (Cat 1) was defined as a major non-conformance that was considered to adversely affect the component and impact on the customer. Such non-conformances could affect, amongst other things, the reliability, life, or performance of the affected part or component and would therefore result in limitations to its in-service use, invoke its non-routine repair, or otherwise cause concern to the customer.
- Category 2 (Cat 2) was defined as a minor non-conformance that required identification to users within the manufacturer. A Cat 2 non-conformance may have associated conditions that require non-scheduled checks, tests or measurements during the remaining manufacturing operations, but would otherwise generally not affect the customer.
- Category 3 (Cat 3) was defined as a minor non-conformance that was not a Cat 1 or Cat 2 non-conformance and would therefore have no limitations or conditions on its use.

Under the heading of 'Reporting', the procedure required that:

> Significant non-conformance and the effective closure of corrective actions shall be reported to, and monitored by, the appropriate Quality review board or equivalent.

There was no information in the group quality procedures, associated appendices or training guide that provided any guidance on what constituted a 'significant' non-conformance.

Retrospective concessions

The concession process permitted the retrospective application of concessions to parts that had already been released into service. The process was the same as for a normal concession except for the following additional requirements:

> With approval of the relevant Chief Engineer and Business Quality Director, or equivalent, to cover exceptional circumstances, where a product is found to be in service and while deviating from specification, remains fit for purpose, a retrospective concession may be raised to permit continued operation.
>
> Consideration must be given as to whether a Safety alert report should be raised.
>
> Businesses must keep a record of occurrences of retrospective concessions for audit purposes.

There was no specific guidance in either the Training Guide or appendices to the group quality procedure on the additional requirements of the retrospective concession. Similarly, neither the concession process nor the Training Guide specified at what point in the process the Chief Engineer or Business Quality Director were to be involved in the retrospective concession approval process. The engine manufacturer reported that their interpretation of the process was that the Chief Engineer was to be the final signature approving the retrospective concession.

The manufacturer also reported that at the time of the uncontained engine failure, Hucknall Casings and Structures did not maintain a separate register of retrospective concessions. Instead, retrospective concessions were recorded as part of the register of all concessions. There was nothing within the concessions register to note that a concession was retrospective.

Observations:

The standard concession form (including the additional signatories sheet) did not contain specific places for the signatures of the Chief Engineer or Business Quality Director. Although these could have been placed in the blank spaces on the additional signatories sheet, there was nothing obvious to prompt the Non Conformance Authority to obtain those signatures. In addition, the form did not require the annotation of a concession as being retrospective.

The retrospective concession process, as used by the engine manufacturer, did not involve the Chief Engineer until the final step in the process. This meant that the concession was treated in the same manner as a routine concession and that a fleet-wide risk assessment was not carried out until all the steps in the routine concession process was complete.

4.8.5 Quality Review Board

The Quality Review Board's function, according to the manufacturer's group quality procedure, was to monitor and ensure effective closure of corrective actions relating to significant non-conformances. The terms of reference for Quality Review Boards (also termed Directors Quality Boards) were contained within the manufacturer's Quality Manual. The manual stated that the terms of reference of a quality review board were:

> Presidents, Managing Directors or functional heads of the businesses chair the (board) meetings with membership comprising nominated line personnel. Meetings are to be held at least three times per year.
>
> The boards are responsible for reviewing:
>
> a) Effective implementation of the manufacturer's quality policy and objectives,
> b) Achievement of product and service quality,
> c) Implementation and effectiveness of the quality system,
> d) Compliance with the quality system,
> e) Continuous improvement initiatives,
> f) Issues from, or for other boards,
> g) Application of best practice.
>
> The input to quality management reviews should include information about:
>
> a) Results of audits,
> b) Customer feedback,
> c) Process performance and product conformity,
> d) Status of preventive and corrective actions,
> e) Follow-up actions from previous management reviews,
> f) Changes, including legal and regulatory items, that could affect the quality management system,
> g) Recommendations for improvement.
>
> The output from the quality management reviews includes any decisions and actions related to:
>
> a) Improvement of the effectiveness of the quality management system and its processes,
> b) Improvement of product related to customer requirements
> c) Resource needs.

The engine manufacturer held a number of quality review boards, each convened at a local supply chain unit level. Such a board was regularly held at Hucknall Casings and Structures. The composition of this board included the Manufacturing Manager, Manufacturing Engineer Manager, a statistical process control expert, Quality Manager, Supply Chain Unit Head of Manufacturing Engineering and Supply Chain Unit Quality Executive.

The manufacturer reported that the board did not perform a quality review of individual quality issues regarding process performance and product conformity. Instead their responsibilities included the monitoring of the progress of activities relating to significant non-conformance when reported to the board. These activities would be discussed in terms of the management of the actions required to progress the issue to closure.

4.9 Engine hub assembly that was involved in the uncontained engine failure

4.9.1 Manufacture

The manufacturing records for the HP/IP hub assembly from ESN 91045 showed that it was manufactured at Hucknall Casings and Structures between April and June 2006. The HP/IP hub assemblies were not allocated a specific serial number, but were identified by a unique number applied to the hub castings. The identification number for the HP/IP hub assembly from ESN 91045 was 0225.

The batch card for hub 0225 was reviewed by the ATSB and it was found that all relevant manufacturing operations (OPs) and the inspections at OP 70 and OP 230 were certified as complete. Each of the inspections was carried out by different inspectors.

The CMM reports for the OP 70 and OP 230 inspections were not attached to the batch card. The manufacturer reported that the CMM reports for hub 0225, and a number of other hub assemblies from around that time, had not been retained.

There were no non-conformance CT17 form references in either the non-conformance history or the individual inspections, to indicate that a non-conformance had been identified with hub 0225 at either OP 70 or OP 230 (Figures 83 and 84). There was one CT17 reference (10-059330) that was listed on both the batch card and non-conformance histories, regarding a non-conforming weld identified at OP 90. The manufacturing records for hub 0225 contained details regarding the rectification of that non-conforming weld and no concession was required.

Figure 83: Excerpts from hub 0225 batch card

0070 INSPCT	3536INSPCT9092- INSPECT	Y		35369092	1			6/4/06	70
0090 XRAY	3537XRAY9797- X-RAY 573894 -573910	Y		35379797	C/L	10-059330		3/5/06	90
0230 FINAL	3536FINAL9092- INSPECT FINAL	Y		35369092	1			3/6/06	230

Figure 84: Excerpt from hub 0225 Non-conformance history

CORRECTION				SCRAP				MANUFACTURING	CHANGE REQUEST
Qty	Document	A[illegible]	Date	Op No	Qty	Document	Serial No	MCR Number	Accepted
1	10-059330		3/5/06						

There were no concessions attached to the batch card and a review of the register of concessions found that none were issued against hub 0225.

4.9.2 Dimensional examination of the oil feed stub pipe

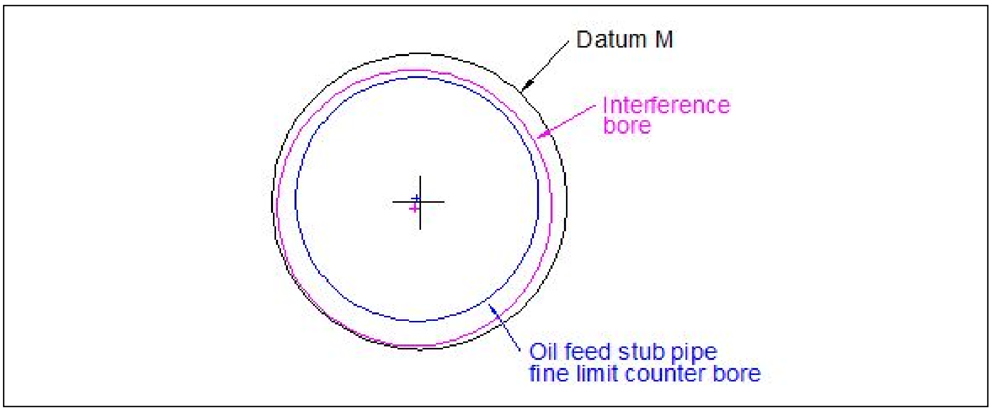
A dimensional examination of the oil feed stub pipe from hub 0225 was carried out by the ATSB. That examination found that the (Figure 85): [92]

- true position of the interference bore with respect to datum M was between ∅0.90 and ∅0.98 mm
- true position of the oil feed stub pipe fine limit counter bore with respect to datum M was between ∅0.00 and ∅0.83 mm
- step between the drilled and reamed counter bores in the oil feed stub pipe varied between 0.0 and 0.55 mm.

The diameters of the interference bore and the counter bores were all of the correct size and within tolerance.

[92] Both the oil feed stub pipe and the inner hub bore were subjected to significant temperature and force during the engine failure. As a result, the surfaces were subjected to scoring, distortion and fracture. The measurements taken reflect the post-event state of the components. Along with the measurement uncertainties, an estimate of the level of uncertainty has been included in the range of the measured true positions provided.

Figure 85: Graphical representation of the true positions of the bores and datum M on the oil feed stub pipe from hub 0225



Observations:

When hub 0225 was manufactured, the true position of the interference bore was probably in excess of the limits contained within the CMM program for the OP 70 inspection. Given the extent of the deviation of the as-measured true position from the acceptable tolerance, it would have been expected that the CMM detected the difference and produced an error on its report. The expected error was estimated to have been in the range from ∅0.40 to ∅0.48mm.

Due to the greater uncertainty in the measurement of the true position of the fine limit counter bore; there is uncertainty as to whether the CMM would have produced an error at OP 230. The CMM report for the oil feed stub pipe fine limit counter bore could have ranged from having no error message, up to an error of around ∅0.63 mm.

Because the CMM reports were not retained by the manufacturer, the ATSB was unable to positively determine whether any errors were produced and/or recorded at the time.

Although the step between the drilled and reamed (fine limit) counter bores was outside the limit specified in the drawings, it was not measured as part of the CMM program and was thus a missed opportunity to detect the non-conformance.

4.10 Major quality investigation

4.10.1 Background

The manufacturer's quality management system included a function for carrying out an internal major quality investigation when an event was identified that affected cost, quality, reputation or customer satisfaction to a significant degree.

In February 2007, a major quality investigation into one of the manufacturer's other facilities at Inchinnan, UK was completed.[93] The findings of that investigation included that:

> ... there were instances where the process of non-conformance sentencing did not follow the required authorisation. i.e. local acceptance of dimensional non-conformance by non-approved personnel ...

93 The manufacturer's facility at Inchinnan produced a number of other gas turbine engine components, including compressor blades. None of those parts directly related to the manufacture of the HP/IP hub assembly, but the major quality investigation was the initiator for later events at the Hucknall facility.

> ... Production Assembly Operations [were] audited and low levels of minor non-conformance (differences between drawing limits and quality acceptance standards and perceived measurement variability) [were] being accepted locally without concessions being raised ...
>
> ... Inspection records from [the] CMM inspection process were not kept in line with the requirements of the EASA part 21 standards ...

The primary factors behind those findings that were identified by the investigation included:

> Informal acceptance of non-conformance by ... shift supervisors
>
> Operator component and concession familiarity

The manufacturer reported that 'concession familiarity' was a local rule[94] whereby personnel would take it upon themselves to allow minor non-conformances to pass, as experience had led them to believe that the concession approval process would not fail an item, and was therefore a waste of time.

A follow-up action to the Inchinnan major quality investigation was to 'read across' the lessons learned other business units.[95]

4.10.2 Hucknall major quality investigation

In June 2007, two separate audits were carried out at the manufacturer's Hucknall facility in the UK. As a result of the audit findings the manufacturer launched a major quality investigation of Hucknall facility. The investigation defined the problem as:

> ... inspection records from the CMM inspection process were not kept in line with the requirements of the EASA part 21 standard ...
>
> Follow up actions to capture and review inspection records from the CMM over a 2 week period showed that there were high levels of non-conformance identified on the inspection reports that was not completely captured through the formal non-conformance management procedure ...
>
> The audit report also identified issues with the effective implementation of other key read across actions from [the Inchinnan major quality investigation] ...

As a consequence of the Hucknall investigation, the delivery of parts from Hucknall was stopped until the key elements of a revalidation and containment program were completed at the facility. Part of the containment actions was to cover the undeclared non-conformances with retrospective concessions. The manufacturer reported that because of the lack of CMM reports, retrospective concessions would be raised when it was identified that there was an undeclared non-conformance.

---

94 The perceived optimisation of a process taken by a person, or group, through the use of shortcuts and expectation-driven actions.

95 During a read across from a major quality investigation, other business units were required by the company to implement any changes applicable to their operations as a result of the subject investigation's findings.

Observation:

The Hucknall major quality investigation appeared to be focussed on preventing the further release of parts with undeclared non-conformances and approving the released parts by retrospective concession, if and when they were identified to be non-conforming. There was no evidence of an attempt by the manufacturer to accurately and reliably determine the conformity, and hence airworthiness, of parts that had left the facility and were in service.

The manufacturer's investigation found that the high level of undeclared non-conformance at Hucknall existed because the facility's management had not effectively read across the actions from the Inchinnan major quality investigation. The Hucknall investigation also found that the ineffective read across was primarily because at the time 'There was not a strong focus on quality within the business'.

In 2007, the manufacturer conducted a cultural survey of staff within another supply chain business unit located within the Hucknall facility. The survey was established as an organisation development action to highlight any underlying cultural issues and was referred to in the Hucknall major quality investigation as part of the implementation of corrective actions. The survey was not conducted as a formal part of the investigation and did not include the Transmissions, Structures and Drives staff at Hucknall that manufactured the HP/IP structure.

The cultural survey was conducted by holding a series of focus groups with representatives from leadership, machine operators, engineers, and inspectors to get their views. This was supplemented by asking individuals involved to anonymously complete an organisational culture assessment that was developed by an external company.

The manufacturer reported that the cultural survey identified genuine concern for product quality and did not discover evidence of intentional violations of procedures. The cultural survey identified that the issues emanated from different views or a lack of broader, shared understanding about what was expected from quality and inspection processes. The focus group discussions centred on several key themes covering:

- doubts about the inspection process
- the need for structure and order
- the 'golden past'
- focus on delivery to the customer.

The output was used as part of a briefing for all Hucknall employees, including those in the Transmissions Structures and Drives unit to heighten quality awareness.

In response to the findings of the Hucknall major quality investigation, the manufacturer took a number of actions. These included retraining relevant Hucknall personnel in the requirements of the non-conformance process and education on why it was important to follow the procedures. In addition, the Inchinnan major quality investigation 'read across' was reintroduced into the Hucknall facility, the quality governance processes were amended and there were a number of organisational changes made.

4.10.3 Regulatory oversight

In accordance with the requirements of their Production Organisation Approval, the manufacturer briefed the UK CAA in respect of the results of the major quality investigation at the Inchinnan facility and was subjected to special audits by the regulator. The UK CAA's regular surveillance of the manufacturer was also increased during the review period following the Inchinnan investigation.

The Hucknall major quality investigation disrupted production at the facility and required a comprehensive re-training program in key areas of non-conformance management and production records management. The manufacturer worked with the UK CAA during that process and was subjected to special audits by the regulator.

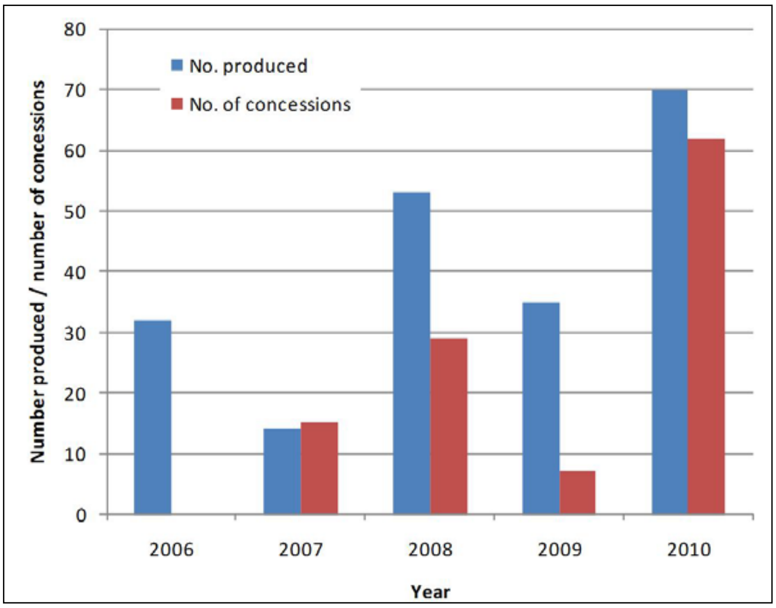
4.10.4 Concession review

A review of the manufacturer's concessions register found that, between February 2006 and July 2007, there was only one concession raised for the Trent 900 HP/IP assembly. Thirty-two assemblies were produced in that period. In contrast, there was a significant increase in the number of concessions raised on the HP/IP assembly following the Hucknall major quality investigation in 2007 (Table 11 and Figure 86).

Table 11: Hucknall facility HP/IP support assembly concessions

Year	No. produced	No. of concessions	No. scrapped
2006	32	0	2
2007	14	15	0
2008	53	29	4
2009	35	7	1
2010	70	62	6

Figure 86: Hucknall facility HP/IP support assembly concessions



The ATSB reviewed a number of concession forms raised at Hucknall after the major quality investigation. Each of those concession forms listed a number of non-conformances. The ATSB's review identified that those non-conformances included one relating to the oil feed stub pipe features.

Post-major quality investigation concessions

The concessions that were raised following the Hucknall major quality investigation identified a non-conformance in relation to the location of the four interference bores in the inner hub for the oil feed, vent and scavenge stub pipes. The non-conformance was listed on one of those concessions forms as:

> Hole positions at 4 stations identified PD, PJ, PE, PF [Oil feed, vent, scavenge, and vent positions, respectively] are up to 0.298 [mm] from True position.

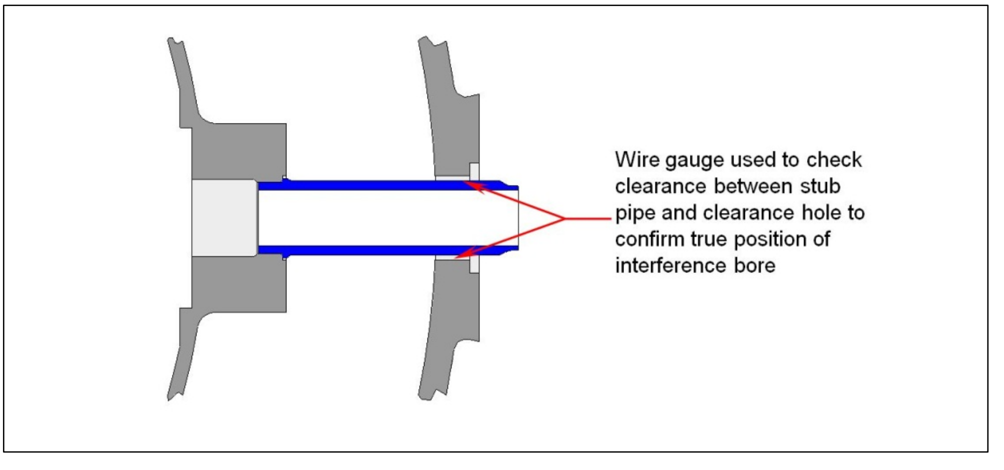
There was no specification in the concession as to which datum was used to reference the true position.

Annotated with the non-conformance description was a note stating:

> A min gap of 0.85 [mm] has been achieved at all positions.

The results from a wire gauge check of the clearance between the respective stub pipes and the outer hub clearance hole were attached to the concession (Figure 87).[96] Those results showed a consistent clearance around the pipe, indicating that the pipe was centrally located within the outer hub clearance hole.

Figure 87: Wire gauge check



The concession was reviewed and accepted by design, stress and air/fluid systems engineers and categorised as a Cat 3 non-conformance. The categorisation of non-conformances was discussed in section 4.8.4 of this investigation report.

Observations:

In sentencing those concessions, the non-conformance authority was required to compare the stated non-conformance with the specification on the design definition and assess the effect on the part's form, fit and function. Because the drawings referenced the true position of the interference bore for the oil feed stub pipe to datum AA (which

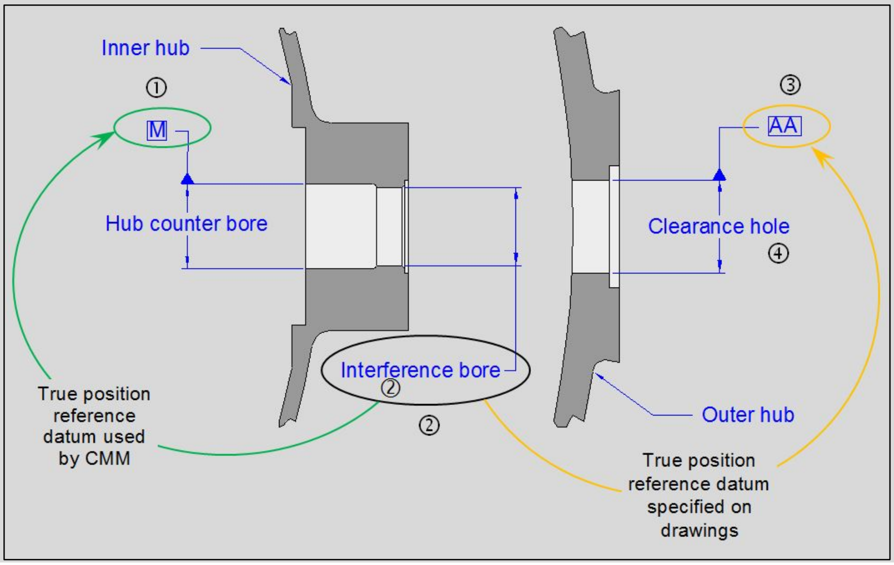
96 A wire gauge check consisted of inserting wires of a known size into the gap to determine the size of the gap.

was the outer hub clearance hole), the only indicator to the non-conformance authority was that there was a possible clearance issue where the stub pipe passed through the outer hub.

Both the manufacturing engineer who prepared the concession form and the non-conformance authority would have been unaware that the CMM had actually measured the true position of the interference bore relative to datum M, not datum AA as listed on the design definition (Figure 88). Given that the results of the wire gauge checks indicated that there was not a clearance issue, it was reasonable for the non-conformance authority to have determined that the non-conforming part was acceptable for use and to have approved the concession without understanding that there was a potentially thin wall on the stub pipe.

Because the manufacturing stage drawings also showed the position of the interference bore relative to datum AA, reference to those drawings during the preparation and sentencing of the concession would not have provided any further indication of the possible reduction in wall thickness.

Figure 88:Differences in the interference bore reference datums



4.11 HP/IP bearing support assembly production standards

At the time of the uncontained engine failure, there were three production standards for the HP/IP bearing support assembly, all of which were operating within the Trent 900 fleet. The different standards were identified as separate part numbers and were referred to as the A, B and C standards. ESN 91045 contained an A-standard HP/IP bearing support assembly.

The A-standard assembly was the initial production standard and a total of 38 units were produced between 1 February 2006 and 13 July 2007. Initially there was no life limit on the assembly but, following the discovery of a defect in the weld that secured a vent pipe to the outer hub on a development engine, a 2,000 cycle life limit was applied to A-standard bearing support assemblies.

The B-standard assembly was manufactured between 13 July 2007 and 11 March 2009, with a total of 67 units produced in that time. There were no differences between the A- and B-standards regarding the oil feed stub pipe. Following the discovery of a manufacturing quality issue in the B-standard support struts, a 14,800 cycle life limit was applied to the B-standard assembly.

The C-standard assembly had been in production since 20 March 2009 and was the current production standard at the time of the uncontained engine failure. Initially, there were no differences between the A-, B- and C-standards regarding the oil feed stub pipe; however, after producing six units, the reference datum for the oil feed stub pipe counter bore in the C-standard assembly was changed to the inner diameter of the stub pipe.

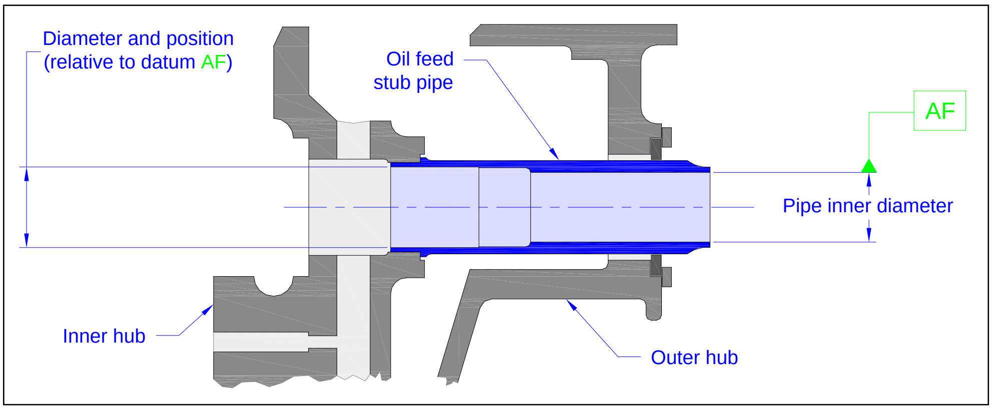
4.12 Manufacturing improvement and datum change

In March 2008, while the B-standard HP/IP bearing support assembly was still in production, a definition alteration request was raised by Hucknall Casings and Structures at the request of Manufacturing Engineering. The reason for the request was listed as:

> Manufacture of the Trent 900 HP/IP structure as it is currently defined is not possible without the use of concessions.

The definition alteration request listed the requested definition changes and included a proposed change in the reference datum for the counter bore in the oil feed stub pipe. This included the introduction of a new datum (AF) and a request to change the positional tolerance reference from AA to AF. Datum AF was defined as the inner surface of the oil feed stub pipe (Figure 89).

Figure 89: Revised oil feed stub pipe counter bore definition



The definition alteration request was approved in April 2008 with the justification that the requested changes did not affect the design intent of the assembly (form, fit and function) and was to be incorporated into the next HP/IP bearing support assembly standard (the C-standard). The alteration request also contained a handwritten note indicating that it had been incorporated at the request of Manufacturing. The design definition drawing for the B-standard HP/IP bearing support assembly was amended in July 2008 to incorporate the requested changes.

The change was not introduced into manufacturing until March 2009, which coincided with the introduction of new CMMs at Hucknall Casings and Structures. At that time, production of the B-standard HP/IP bearing support assembly had ceased and C-standard production commenced. The programs were rewritten for the CMM machines and the definition changes incorporated.

On 27 March 2009, while performing verification checks on the revised CMM program, a manufacturing engineer at Hucknall Casings and Structures identified that the counter bore on two previously manufactured HP/IP hub assemblies was out of position (or offset) by about ∅0.50 mm with respect to datum AF. Those counter bores had been measured with respect to datum M using the previous version of the CMM program and found to have been within tolerance. The engineer also identified the effect that the offset would have on the wall thickness of the oil feed stub pipe and brought this to the attention of their supervisor and a design engineer through discussion and an email.

The email from the manufacturing engineer detailed the dimensions for the second non-conforming part and a calculation of the oil feed stub pipe wall thickness using the nominal dimensions for the pipe and interference bore. The resulting wall thickness was calculated to be 0.703 mm. The engineer also suggested that a true position offset of ∅0.5 mm was representative of a fair portion of the parts previously passed as acceptable. This was qualified by noting that it was too early to determine the distribution of those positional offsets and that the previous data would not permit a retrospective analysis.

Observations:

The identification by the manufacturing engineer on 27 March 2009 of the offset oil feed stub pipe counter bore with respect to datum AF was reported by the engine manufacturer to be the first time that they had identified the misalignment of the counter bore with respect to the pipe's axis. Similarly, it was the first time that the reduction in the oil feed stub pipe wall thickness was identified.

Due to the design definition of the stub pipe, the true position of its inner diameter could be off-centre from the interference bore by up to 0.05 mm. By including this, and the most adverse tolerances for the counter bore and interference bore diameters, the ATSB found that the wall thickness could have been as low as about 0.60 mm.

The ATSB determined that the minimum wall thickness obtainable when conforming to the design definition drawings and datum AF was 0.80 mm.

4.13 Oil feed stub pipe counter bore retrospective concession

On recognising that the counter bores were misaligned on previously manufactured HP/IP bearing support assemblies (hubs), Hucknall Casings and Structures raised a retrospective concession to cover the 100 non-conforming oil feed stub pipes that had been released into service. In order to sentence those parts, their maximum non-conformance needed to be determined.

The manufacturer did not have any specified process to evaluate the degree of non-conformance of released components, although a number of tools were available to the manufacturing engineer. In the first instance, the engineer measured the true position of the oil feed stub pipe counter bores, with respect to datum AF, on the nine work-in-progress hub assemblies that were available at the facility. A statistical analysis of those measurements was carried out sometime between 27 March and 3 April 2009 by the engineer using a computer tool regularly used in the manufacturer's process control. The results of this analysis were presented as a 'process capability report' with the concession application. The process capability report presented, amongst other things, the mean and standard deviation for the measured data, based on a normal population distribution.

The non-conformance was listed on the concession form in the manner presented in Figure 90:

Figure 90: Description of the non-conformance on the retrospective concession form

3. DESCRIPTION OF NONCONFORMITY	3.1. Detailed Description of the nonconformity

Serial Number	Qty	DWG Ref Sheet/Loc	Defect Characteristics/Specified Reqs NOMINAL	MAX	MIN	ACT DIMENSION

```
Parts No's affected : FW42619, FW48020, FW59623, FW64481

1) 1        100   27/D3    POSITION Dia 0,10Max to AF    Up to 0,70
Note !
Data taken from the last 9-Off details produced and inspected correctly.
Mean at Posn Dia 0.31
Standard deviation 0.136
100 parts produced to date, est worse case: mean plus 3 std dev 0,70
```

Previous Submissions of a similar nature: | Previous Submissions to actual item:

Note: The 'Part number' box on the form listed only the FW64481 part number.

The manufacturing engineer was very experienced[97] in the concession process, but had limited experience in using the statistical analysis program. The engineer did not seek any input from more experienced users within the organisation during his consideration of the concession. Experienced users of the statistical analysis program advised the ATSB that the sample of nine work-in-progress hub assemblies was not statistically representative of the previously released and in-service assemblies. This was because of the number of assemblies measured, and the change in the method of manufacture (incorporating computer numerical control machining and CMM measurements).

The manufacturing engineer attempted to convey a qualitative assessment of the level of uncertainty within the analysis by including the abbreviation 'est' preceding 'worse case'. The engineer reported to the ATSB that 'est' was included to indicate the listed non-conformance was an estimate of the positional error.

The Non Conformance Authority[98] who reviewed and approved the retrospective concession had received some training in the statistical analysis program and method and did not identify any issues with the statistical analysis incorporated into the concession application. The authority reported having an expectation that the data presented in the concession application was processed by a suitably trained person within the Manufacturing Engineering Team.

The concession form was endorsed by design and stress engineers on 3 April 2009. Those engineers reported that their analyses of the oil feed stub pipe counter bores

97 The manufacturing engineer had been raising concession applications since 1978 when the training in support of that skill consisted of raising 50 concessions under supervision. The engineer had not received any refresher training on concession applications, other than in the late 1990's, when the manufacturer provided some training in the computer system that was being used at the time for concession applications. At the time the retrospective concession was raised, the engineer was generating about one concession per week.

98 A qualified engineer who had been involved in a number of engine projects with the manufacturer as a design engineer. At the time the retrospective concession was raised, the non-conformance authority had 2 years of experience as a design scheme supervisory signatory, was a design concession approval signatory and had been a regular attendee at concession review boards for over 1 year.

as presented on the form was based on a maximum non-conformance of ∅0.70 mm, and that they considered the effect of that non-conformance on the component's form, fit and function. They established the wall thickness of the oil feed stub pipes and used a computer structural analysis model to ensure that the stress in the pipes would remain within allowable limits.

Observations:

It was likely that the stress analysis carried out by the stress engineer was not as detailed as that carried out as part of the ATSB investigation, which had the benefit of additional data as a result of the uncontained engine failure (see section 2.5 of this investigation report). The stress analysis by the manufacturer as part of the ATSB investigation found that a very detailed model was required to capture the actual peak stress in the oil feed stub pipe wall, and to assess the resulting fatigue life.

The true position of the oil feed stub pipe counter bore in ESN 91045 was greater than ∅0.70 mm from datum AF. As shown later in section 4.15 of this investigation report, there were a number of other counter bores that had thinner wall sections and thus had a counter bore with a true position from datum AF greater than the ∅0.70 mm in the retrospective concession. Thus, the results of the statistical analysis used to sentence the retrospective concession did not accurately represent the population of the in-service oil feed stub pipe counter bores.

The ATSB calculated that the minimum wall thickness in the oil feed stub pipe when the true position varied from datum AF by ∅0.70 mm was 0.50 mm.

The retrospective concession form was sentenced on 3 April 2009 and categorised as a Cat 3. That categorisation was supported by the design and stress engineers who reviewed the concession. The ATSB compared the concession with the requirements of the group quality procedures and, in the context of the information in the concession alone, found that the categorisation was compliant with those procedures.

The manufacturer provided the ATSB with a copy of the notes from the Transmissions, Structures and Drives – Hucknall Casings and Structures Plant Quality Board (quality review board) convened on 10 June 2009 (the first Quality Review Board meeting convened after the retrospective concession). There was no reference in those notes of the oil feed stub pipe retrospective concession, indicating that it had not been reported to the Quality Review Board.

The non-conformance authority did not expect that the in-service components would be individually checked for conformance with respect to the limits specified on the retrospective concession. However, the authority did expect that the components would be tested for any cracking at the next overhaul as part of the normal overhaul procedure.

The completion of the retrospective concession form did not comply with the manufacturer's process in a number of areas, including multiple part numbers being listed on the one form.[99] Secondly, the retrospective concession had not been approved by either the Chief Engineer or the Business Quality Director. Upon reviewing the concession application form, as part of the ATSB investigation, the non-conformance authority recognised that it lacked those signatures, but could not recall why they had not been obtained.

99 There were four part numbers, which represented a development standard, and the A-, B- and C-standards.

The manufacturer provided the ATSB with a copy of the notes from the Transmissions Structures and Drives quality review board convened in June 2009 (two months after the retrospective concession). There was no reference in those notes of the retrospective concession, or any other significant non-conformances.

> **Observation**:
>
> By not obtaining the approval of the Chief Engineer and Business Quality Director for the retrospective concession that was issued in April 2009, the opportunity for their experience to have an effect on the consideration of the concession and the fleet-wide risk assessment was missed.

4.13.1 Containment of non-conforming parts

The manufacturer reported that when the non-conforming oil feed stub pipe counter bores were detected, all of the parts that had not been delivered to customers were quarantined in accordance with their containment procedure. They also reported that the Chief Engineer would normally decide what fleet action was required but, because in this instance the Chief Engineer was not included in the process, that level of decision was not taken.

On the local level, when the retrospective concession was sentenced it was determined to be a Cat 3, as it was assessed to have no effect on the customer. As such, it did not require the containment of in-service components.

4.13.2 Other retrospective concessions

A review of previous concessions by the manufacturer was carried out after the UERF involving VH-OQA and it was found that retrospective concessions made up approximately one in 1,000 of all concessions raised. The Non Conformance Authority that sentenced the retrospective concession on the oil feed stub pipe counter bores reported that in their time as an authority, they had reviewed in excess of 1,000 concessions. Of these, probably less than 10 were retrospective concessions.

In a review of the concessions completed by the Civil Large Engine operations[100] after the UERF involving VH-OQA, the manufacturer identified 138 retrospective concessions in the period from 2009 to 2011. Of these, 131 were not compliant with the process, and they were subsequently revalidated and sentenced by the Chief Engineer and Business Quality Director.

4.14 Partial first article inspection

Following the implementation of datum AF, Hucknall Casings and Structures performed a partial first article inspection of the HP/IP bearing support assembly. That inspection was limited to the features that had been varied by the definition alteration request.

[100] Civil Large Engine operations included a number of engine projects and numerous supply chain units.

4.14.1 Change in the first article inspection process

In the time since the original pre-production first article inspection, the first article inspection process had been modified in a number of ways. These modifications included:

- a requirement for a Design Authority to authorise the first article inspection report
- the inclusion of a requirement to certify the 'closure of non-conformance' identified during the first article inspection
- improved definitions.

The group quality procedures noted that design authorisation was only required when the measurement system capability was below target, multiple non-conformances were present, or the process capability was below target. Further, a field was included on the first article inspection report proforma for Design Authority signature, but in cases where there was no requirement for design authorisation, the field was to be marked 'N/A'.

The revised process included a section for the closure of non-conformance identified during the first article inspection (FAI). That section stated that:

> The FAI is not complete until the Organization closes all non-conformances affecting the part and implements corrective actions. The Organization shall re-do a FAI for those affected characteristics and shall record the results.

The definitions section of the revised process included a number of enhancements. Of particular note was the inclusion of a definition for the term 'definition', which precluded the use of stage drawings or locally generated definitions as follows [emphasis added]:

> Definition — The original engineering drawings, models, functional requirements and other elements that define the product under consideration. ... **Stage drawings or other locally generated definitions, may not be used for the completion of the FAI activity**.

4.14.2 Completion of the partial first article inspection on the HP/IP HP/IP bearing support assembly

The partial first article inspection was carried out on an HP/IP bearing support assembly that had been produced using the revised manufacturing method (including datum AF) on 29 April 2009. A first article inspection report was produced to record the completion and findings of the inspection. A number of items in the partial inspection report were of interest:

- The design authority signatory field was marked as 'N/A' and the justification given was that it was an existing part.
- The dimensional results from the first article inspection showed that the true position of the counter bore was ∅0.20 mm, while the maximum permitted was ∅0.10 mm. The overall depth of the counter bore was also outside the allowable limit. The non-conformances were noted on the report along with the associated concession number. The form was marked 'Conditional Accept. Non-conformance subject to concession.'

Also of note was that the 'max step' feature was not measured and the drawings attached to the inspection report were manufacturing stage drawings, not the design definition drawings.

Observation:

During the partial first article inspection, the true position of the fine limit counter bore was out of tolerance by twice the allowable limit. A concession was raised to cover that non-conformance, but this did not strictly meet the intention of the procedures for the closure of non-conformances found during first article inspections.

Given the context of the only recently raised concession allowing the continued use of parts in service with counter bores out of tolerance by seven times the allowable (maximum permitted of ∅0.10 mm compared to a maximum non-conformance of ∅0.70 mm), the non-conformance identified in the partial first article inspection was probably considered by the manufacturer to be acceptable.

Datum AF was introduced as part of a definition alteration request in order to reduce the rate of non-conformance. Yet the feature of concern did not conform, indicating that the change in datum did not successfully remedy the problem.

4.15 Other non-conforming Trent 900 HP/IP bearing support assemblies

Following the UERF involving VH-OQA, the engine manufacturer surveyed the Trent 900 fleet of engines in an attempt to determine the wall thickness of the oil feed stub pipe in all engines produced up to March 2011. In the case of the engines for which the manufacturer had retained the CMM reports, the wall thickness was calculated using the measurements taken during OP 70 and OP 230.

A number of other methods were used to determine, or estimate, the wall thickness for the other engines for which no CMM reports were held. This included the direct measurement of oil feed stub pipes that were removed from HP/IP bearing support structures when those structures were retired from service, and the on-wing measurement of the internal counter bore features using a borescope[101] or a rubber cast taken from an engine's oil feed stub pipe counter bore region.

The results of the calculations and estimates were provided to the ATSB and are presented in Figure 91.

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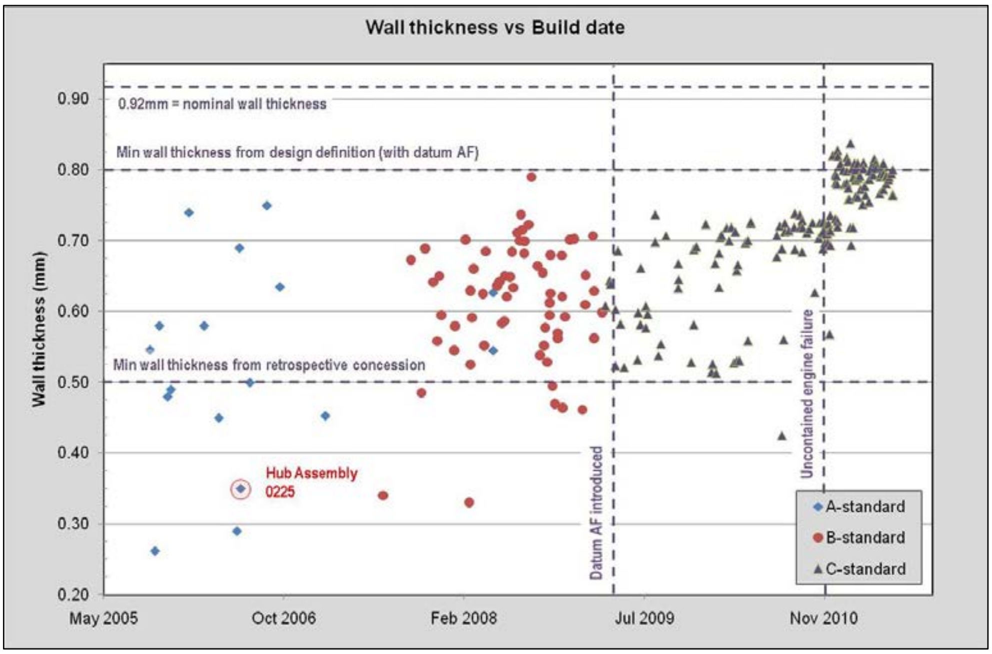
[101] A slender optical device that is capable of being inserted into narrow apertures to inspect the interior of machinery.
Measurements were taken between the internal pipe diameter and the drilled counter bore, and the wall thickness estimated from those measurements. There were a number of limitations on that method, including assumptions that the reamed counter bore was offset in the same direction as the drilled counter bore (which, in the case of ESN 91045 was not), of nominal dimensions for all other features, and that the inside diameter of the pipe was coaxial with the interference bore.

Figure 91: Oil feed stub pipe wall thickness survey, determination and estimates (No. 2 engine hub assembly from VH-OQA highlighted)



Note: The wall thicknesses from 17 of the 38 A-standard HP/IP bearing support assemblies produced were available to the ATSB.

The true position of the oil feed stub pipe counter bore was measured from both datum M and datum AF for two of the in-service oil feed stub pipes that were identified as having a reduced wall thickness. The results of those measurements are shown in Table 12.

Table 12: Comparison of the true position of oil feed stub pipe counter bore

Hub Assembly	True position relative to datum M (mm)	True position relative to datum AF (mm)
0248	∅0.169	∅1.184
0250	∅0.158	∅1.220

Observation:

Hub assemblies 0248 and 0250 were manufactured at a time when the CMM program measured from datum M and had a tolerance of ∅0.2 mm. Hence, those assemblies were unlikely to have produced an error when measuring the true position of the oil feed stub pipe counter bore at OP230. If the tolerance had been consistent with the design definition drawing and been ∅0.1 mm, they would likely have produced errors during that inspection.

The CMM reports for those assemblies were not retained by the manufacturer, so it could not be conclusively determined if an error was reported at OP70. However, given the degree of variation between the measurements from the two datums, it is likely that an error was reported.

5 ANALYSIS

5.1 Introduction

Shortly after takeoff from Changi Airport, Singapore, a Qantas Airbus A380-842 aircraft registered VH-OQA, sustained an uncontained engine failure. The engine, a Rolls-Royce Trent 972, was mounted on the No. 2 position on the wing. The debris released from the engine resulted in significant damage to the engine and airframe.

Although there was significant damage to the aircraft structure and systems, the aircraft was capable of continued safe flight and landing. Furthermore, the flight crew operated in accordance with standard operating procedures and practices. Therefore, this analysis is focused on the failure of the No. 2 engine and the factors that lead to the failure.

This analysis is structured to provide an overview of the engine failure mode, as detailed in Part 3 of this report, and examines the factors involved in the release of non-conforming Trent 900 series engines into operational service. It also examines the opportunities when the non-conformances could have been detected and preventative actions taken prior to the uncontained engine rotor failure (UERF) of the No. 2 engine on VH-OQA.

The analysis is concluded with a discussion on the opportunity presented to extend the knowledge base relating to the hazards from UERF events and to incorporate that into the airframe certification advisory material.

5.2 The occurrence event

The uncontained engine failure was the result of an internal engine fire in the high pressure/intermediate pressure (HP/IP) bearing support assembly that heated the intermediate pressure (IP) turbine drive arm until the material failed.

A fatigue crack had formed in the wall of the oil feed stub pipe over a period of time, and on the occurrence flight the crack reached a size that permitted the release of oil as an atomised spray into the buffer space between of the HP/IP bearing chamber. It was sufficiently hot within this chamber for the oil spray to auto-ignite.

The fire propagated through the bearing chamber buffer space, eventually impinging upon the IP turbine disc, resulting in the separation of the disc at the drive arm. The IP turbine disc moved rearwards and the engine surged. Within 4 seconds, the disc accelerated, and while the engine was running down from the surge, the HP system partially recovered, supplying additional energy to the IP turbine disc, allowing it to accelerate to a rotational speed in excess of its structural capacity.

The disc fractured into three main sections that had sufficient energy to penetrate the engine case and exit at high speed. Sections of the IP turbine disc and associated debris impacted and damaged the aircraft structures and systems.

The crack that formed in the oil feed stub pipe was a result of the normal operating movement of the HP/IP hub and the increased stresses in the pipe as a result of its reduced wall thickness. The reduced wall thickness was due to the counter bore for an oil filter being misaligned with the axis of the oil feed stub pipe during manufacture.

5.2.1 HP system recovery

The HP system recovery during the engine rundown that followed the separation of the IP turbine from the drive arm supplied additional energy to the separated IP turbine. That HP recovery had not been expected by the manufacturer, whose modelling during the design and certification for the engine did not predict such behaviour. The manufacturer's experience with other engines in the Trent family had indicated that the system would not recover from the surge and the IP turbine would not reach a speed in excess of its design capability.

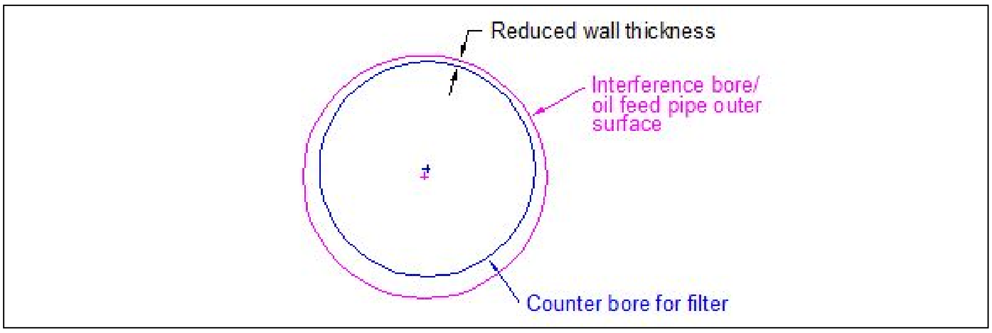
5.3 Misalignment of the oil feed stub pipe counter bore

How the counter bore in the oil feed stub pipe came to be misaligned, released into service and remain in service undetected was a complex combination of events and missed opportunities that started with the design of the engine.

The counter bore in the oil feed stub pipe became misaligned with the axis of the pipe due to movement within the hub structure during the manufacturing process. Between the time when the location of the timing pin was measured and the inner hub counter bore was formed, the inner hub moved relative to the milling machine. Because the interference bore, into which the oil feed stub pipe was later fitted, had previously been formed on the axis of the timing pin, the inner hub counter bore was not aligned with the axis of the interference bore, and hence the axis of the oil feed stub pipe.

After the oil feed stub pipe was fitted to the hub, the oil feed stub pipe counter bore was machined using the inner hub counter bore as the reference datum to position the counter bore. Because the inner hub counter bore was misaligned with the axis of the stub pipe, so too the oil feed stub pipe counter bore was misaligned with the axis of the stub pipe. This resulted in a locally thin wall in the oil feed stub pipe (Figure 92).

Figure 92: Effect of misaligned counter bore on pipe wall thickness



5.4 Detection of the misalignment during manufacture

Dimensional inspections were carried out on two occasions during the manufacture of the HP/IP bearing support hub structure that measured oil feed stub pipe counter bore features; OP 70 and OP 230. At OP 70, the coordinate measuring machine (CMM) used to measure the position of the interference bore was programmed to measure features relative to the inner hub counter bore (datum M). An error was included in the CMM report if the measured position of that feature was outside of the defined limits. Similarly, the CMM was programmed so that, at OP 230, the position of the oil feed stub pipe counter bore was measured relative to the inner hub counter bore and an error recorded in the report if outside of the defined limits.

5.4.1 Engine ESN 91045 – installed as the No. 2 engine on VH-OQA

According to post-accident measurements, an error should have been reported at OP 70 during the manufacture of the HP/IP bearing support structure hub assembly from engine serial number (ESN) 91045. Due to the failure-affected condition of the recovered components, it was less certain whether an error should have been produced at OP 230. Because the CMM records for the HP/IP bearing support structure in ESN 91045 was not retained by the engine manufacturer, the ATSB could not determine if an error was printed on the CMM report. Even if an error was produced, there may have been a number of reasons to explain why it was not reported as a non-conformance. This could have included a problem with the print-out, simply missing it when reviewing the report or an erroneous second measurement that indicated that there was no error.

5.4.2 Other affected Trent 900 engines

The ATSB identified a number of other HP/IP hub assemblies fitted to Trent 900 engines that contained misaligned oil feed stub pipe counter bores and had been released into service. There were a number of explanations as to how this occurred, but these could be divided into two groups: those released into service prior to the Hucknall major quality investigation and those following that investigation.

Prior to the Hucknall major quality investigation

One of the findings of the Hucknall major quality investigation was that the facility was releasing components with a large number of non-conformances that had not been declared in the documentation (undeclared non-conformances).

Post-accident determination of the oil feed stub pipe wall thickness from a number of the HP/IP hub assemblies produced prior to the Hucknall major quality investigation indicated that there were a significant number that did not conform to the design intent. It was therefore likely that HP/IP hub assemblies were released with undeclared non-conformances.

Given that there were a number of inspectors involved over the 3-year period leading up to the major quality investigation, it was likely that a culture[102] existed within the Hucknall facility that considered that not strictly adhering to the non-conformance management procedures in the group quality procedures was a viable form of behaviour. Specifically, it was considered acceptable to release parts with

[102] Defined as ‘shared values (what is important) and beliefs (how things work) that interact with an organisation’s structures and control systems to produce behavioural norms (the way we do things around here)’ Reason, J. 1997. *Managing the risks of organisational accidents.* Ashgate: Aldershot UK (p.192).

undeclared non-conformances that were considered by at least some of the inspectors to be 'minor'.

Many leading human factors researchers[103] have discussed the issues large aerospace organisations face to maintain an ultra-high level of safety. They state that large aerospace organisations are complex sociotechnical systems made up of organised humans producing highly technical artefacts for complex systems, such as modern aircraft. Due to the inherent nature of these complex sociotechnical systems, their natural tendency is to regress if not constantly monitored – and occasionally even when monitored vigorously. This natural regression can occur due to the pressures applied via global economic forces, the requirement for developing growth, profit and market share for stakeholders and the striving for greater efficiencies regarded as commercially essential in what is a very competitive market.

The often less than ideal interaction between humans and organisations' written processes and procedures can also influence in this situation. In this environment, aspects usually captured by the controls put in place by the organisation's quality management system can be missed.

The complex sociotechnical system movement identified by the previously mentioned human factors researchers may provide a high-level, overarching explanation as to why, at the Hucknall facility in mid-2006, a number of staff involved in the production of the HP/IP hub assemblies did not strictly adhere to procedures as laid down in the group quality procedures. The ATSB established that through the Hucknall major quality investigation and other initiatives, the manufacturer has recognised and addressed these issues.

Following the Hucknall major quality investigation

The step increase in the number of concessions being raised for the HP/IP support assemblies following the Hucknall major quality investigation indicated that there was a cultural shift following the investigation and its associated actions having effect. Those concessions included non-conformances relating to the oil feed stub pipe. Examination of the CMM records and the oil feed stub pipe concessions showed that for the support assemblies produced there was an error identified in the true position of the interference bore and that a wire gauge check had been carried out following installation of the oil feed stub pipe. The concessions only identified that the position of the interference bore had a non-conformance and made reference to the applicable design definition drawing.

In sentencing the concessions, the reviewing Non Conformance Authorities would have made reference to the design definition drawing. They would not have been aware that the CMM program measured the position of the interference bore from datum M rather than from datum AA, as shown on the design definition drawing.

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[103] Rasmussen, J. (1997). Risk management in a dynamic society: A modelling problem. *Safety Science*, 27(2/3), 183-213; Reason, J. (1997). *Managing the risks of organizational accidents.* Aldershot, UK: Ashgate; Hollnagel, E. (2004). *Barriers and accident prevention.* Aldershot, UK: Ashgate; Hollnagel, E., Woods, D.D., and Leveson, N. (2006). Resilience engineering: Concepts and precepts. Aldershot, UK: Ashgate; Dismukes, R. K., Berman, B.A., and Loukopoulos, L.D. (2007). *The limits of expertise: Rethinking pilot error and the causes of airline accidents.* Aldershot, UK: Ashgate; Hollnagel, E. (2009).*The ETTO principle: Efficiency-thoroughness trade-off: Why things that go right sometimes go wrong*, Aldershot, UK: Ashgate.

The manner in which the non-conformance was presented in the concession would have indicated that there was a possible lack of clearance between the oil feed stub pipe and the outer hub. However, given that this was not the case, and the wire gauge check showed that there was adequate clearance, the non conformance authorities sentenced the parts on that basis and determined that the assembly was acceptable for use.

Even though the errors identified by the CMM were an indication that there was a potentially thin wall on the oil feed stub pipe, the opportunity to detect this was missed because of the difference between the datum used in the CMM program and the one listed on the drawings.

5.4.3 Effectiveness of the OP 230 inspection

Because the feature used as datum M (the inner hub counter bore) was not constrained to the oil feed stub pipe location, the measurement of the oil feed stub pipe counter bore at OP 230 did not provide assurance that the counter bore was concentric with the oil feed stub pipe. Thus, once a non-conformance had been accepted during OP 70, either in accordance with the non-conformance management procedure or not, a reduced wall thickness could be produced without an error being reported in the CMM report at OP 230. This was shown to be the case on at least two occasions when post-accident measurements showed that an error at OP 230 was unlikely using datum M but, when measured from the stub pipe inner diameter axis (datum AF) showed a large error in position.

To ensure that an acceptable wall thickness was maintained, the designers applied a tight tolerance on the allowable deviation of the location of the interference bore and counter bore. However, when the manufacturing drawings were produced, those tolerances were increased - the location of the interference bore by a factor of 10 and the counter bore by a factor of two. Both further increased the potential size of any misalignment of the counter bore in the oil feed stub pipe. The ATSB was unable to determine why those tolerances had been increased.

5.5 Opportunities to detect the misalignment of the oil feed stub pipe counter bore

The ATSB identified a number of opportunities throughout the design and production of the Trent 900, prior to the uncontained engine failure event, where the potential for misalignment of the oil feed stub pipe counter bore could have been identified. These existed during the production of the manufacturing specifications and instructions, during the first article inspection, and when the manufacturing reference datum was changed.

5.5.1 Production of the manufacturing specification and instructions

The design definition required that the interference bore and oil feed stub pipe counter bore positions be constrained relative to the outer hub clearance hole (datum AA). In producing the design definition, the design engineers had not identified that datum AA became inaccessible when the oil feed stub pipe was fitted to the hub assembly. Neither was it identified during the manufacturing acceptance of the design definition drawings by the manufacturing engineers. From the evidence available, the ATSB could not determine why it was not identified. However, given the large number of features and associated manufacturing processes required to produce the hub assembly, it was concluded that it was probably an oversight.

The inaccessibility of datum AA was identified by the manufacturing engineers at some time following the manufacturing acceptance, and during the determination of the manufacturing specification and instructions. This provided an opportunity for the manufacturing engineers to discuss the problem with the design engineers; however, no discussion was found to have taken place and a separate manufacturing datum was introduced. This datum (datum M) was intended to replace datum AA during the manufacture of the hub assembly. At the time, there was no requirement for the manufacturing engineers to discuss the use of manufacturing datums with the design engineers and the decision to introduce datum M was likely made with the belief that the design intent would be maintained. The possibility of movement of the inner hub during manufacture and the effect it could have when using datum M was not recognised by the manufacturer.

5.5.2 First article inspection

The next opportunity for the misaligned oil feed stub pipe counter bore to have been identified was in May 2005 when the first article inspection was carried out prior to commencement of production. By using the manufacturing stage drawings the inspector found that all measurements conformed to that drawing, but this did not show conformity to the design definition drawing. Neither did it ensure that the design intent was achieved, because the lack of constraint on datum M meant that the alignment of the counter bores and the stub pipe could not be assured. Thus, the first article inspection did not 'close the loop' on the production method and could not confirm that the design intent was attained.

Had the inspector carried out the inspection using only the design definition drawings, they would have formally identified that they could not conform to those drawings. This should have resulted in the first article failing the inspection and the issue addressed.

The ATSB assessed that the wording within the first article inspection procedures had ambiguities that may have led the inspector to believe that the use of manufacturing stage drawings was acceptable. The first article inspection report was signed off by the Integrated Programme Team lead, indicating that this understanding of the process was not restricted to the inspector.

The ambiguities in the first article inspection procedure lay predominantly in the wording used in the definition of key terms. The wording of the definition for 'first article inspection' indicated that manufacturing stage drawings were included in the 'other applicable documents e.g. manufacturing definition' that could be used to show conformance to the design. The definition of 'definition requirement' did not clarify this any further as it permitted the use of a 'lower-level definition' that may have been interpreted as including manufacturing stage drawings.

The first article inspection procedure noted that part of its purpose was to provide confirmation that the production method resulted in a part that fully met the design intent. However, neither the procedure nor the associated guidance information required the inclusion of the applicable design engineers in the process. Due to the lack of guidance information and without including the design engineers in the process, the ATSB could not ascertain how the first article inspection team would ensure that the design intent was achieved.

5.5.3 Change in the oil feed stub pipe counter bore reference datum and the retrospective concession

The change in the reference datum for locating the oil feed stub pipe counter bore from datum M to datum AF occurred in March 2009. At that time a manufacturing engineer was alerted to the effect that the position of datum M was having on the wall thickness. This engineer also identified the likelihood of escaped parts in the in-service Trent 900 engine fleet. In an attempt to define the non-conformance, the engineer used a statistical analysis approach that was based on the available nine work-in-progress assemblies at the Hucknall facility. The statistical analysis tool used by the engineer was not capable of providing results that were representative of the variation in oil feed stub pipe counter bore position within the in-service fleet.

The manufacturing engineer's attempt to convey their level of uncertainty to the non conformance authority was not effective. The language used on the concession form to define the non-conformance likely gave the non conformance authority and the design and stress engineers the impression that the maximum non-conformance was 0.7 mm, rather than 0.7 plus or minus some unknown level of uncertainty. The use of terms such as 'up to' to define the non-conformance and 'est worse case' in a clarifying note were more consistent with a maximum case for consideration, rather than the result being an approximation with an inherent degree of uncertainty. The non conformance authority, design and stress engineers appear to have considered the concession on the basis that 0.7 mm was the maximum level of non-conformance.

Although the different production standard parts were similar, the parts that were measured by the manufacturing engineer were all C-standard parts. However, the retrospective concession for the oil feed stub pipes contained four part numbers on the one form, rather than one part number per form, in accordance with the requirements of the engine manufacturer's non-conformance management process. The use of measurements from a part number different to that on the form may have highlighted that the measurements were not necessarily applicable to that part and alerted the non conformance authority to the non-representative nature of the statistical analysis.

There was no indication that the non conformance authority who sentenced the concession identified that there was any uncertainty regarding the statistical analysis, or that they recognised that the results were not representative of all of the parts within the facility and in the in-service fleet. Because there was no requirement to obtain expert advice when statistical analysis was used in a concession application, further advice from the statistical process control experts within the organisation was not obtained.

Contrary to the manufacturer's additional requirements for a retrospective concession, neither the Chief Engineer nor the Business Quality Director approved the retrospective concession. Although this would not have necessarily ensured that the concession was not approved, and that additional safety actions were taken, it was another opportunity that was missed to detect the non-conformance in the Trent 900 fleet of engines before engine ESN 91045 sustained an in-flight uncontained failure.

Fleet wide risk assessment

Concessions were primarily sentenced within the business unit where the component was being produced, in this case Transmission, Structures and Drives. However, when sentencing a retrospective concession, the Chief Engineer's approval provided a broader fleet-wide assessment of the risks associated with permitting non-conforming parts to continue operation in the fleet. This related to whether the non-conformance posed a fleet-wide risk and what containment action may be required to minimise any risk, both in the short and long term.

Why the non conformance authority did not obtain the approvals from the Chief Engineer and Business Quality Director when sentencing the retrospective concession could not be conclusively determined. However, the ATSB determined a number of factors that may have combined to result in the omission as follows:

- the standard form used for both normal and retrospective concessions did not assist the non conformance authority to ensure that the additional approvals were obtained.
- the concession form contained a list of common specialist areas that may be required during the sentencing of a concession, but that list did not include the Chief Engineer and Business Quality Director, and as such there was no prompt to remind the non conformance authority to obtain those approvals.

Similarly, there was nothing on the concession form to explicitly indicate that the concession was retrospective. Such an indication may have prompted the non conformance authority to realise that it was not a routine concession application and, as such, required additional actions.

The number of concessions being raised by the manufacturing engineer and assessed by the non conformance authority at the time indicated that the raising and sentencing of concessions was a routine process; however, the raising and sentencing of a retrospective concession was not routine. Because the majority of the processes were common to both normal and retrospective concessions, it was possible that the non conformance authority processed the retrospective concession in the same manner as a normal concession.

By not specifying when in the retrospective concession process the Chief Engineer's approval was to be obtained, there was potential for their expert consideration to be missed if the process was not completed, as occurred in this case.

In contrast to the retrospective concession process, the need for an early fleet-wide risk assessment was inherent in the process for a Safety Alert Report. In the safety alert process the airworthiness department and the Chief Engineer were involved early in the decision making process. This ensured that the Chief Engineer's expert consideration was included in the determination of the extent of, and risks associated with, the potential issue. In the case of the non-conformance identified within the oil feed stub pipe, based on the manufacturing personnel's knowledge and understanding at that time, they did not identify a potential safety issue, determined that a retrospective concession was suitable, and the raising of a Safety Alert Report was not required.

Quality review board

Although the Quality Review Board was not responsible for a technical review of retrospective concessions, one of their functions was to ensure compliance with the quality management system. One of the means of carrying out this function was through receipt of reports regarding significant product non-conformances. The board was not responsible for the detailed review of each concession to determine its compliance with the process; however, they were responsible for monitoring reported significant non-conformance and the management of the effective closure of any corrective actions.

Given that the concession covered all of the parts manufactured up to that date and released into service, the ATSB considered that from a quality assurance perspective, the non-conformance should have been deemed to be significant. The engine manufacturer's procedures did not provide any guidance on how to determine the significance of a non-conformance. Therefore, it was unlikely that Hucknall Casings and Structures personnel would have reported the retrospective concession to the Quality Review Board because they considered that from an engineering point of view the non-conformance was minor, as indicated by the Cat 3 categorisation, and therefore not significant.

Even though the non-conforming oil feed stub pipes should have been reported to the Quality Review Board, it was unlikely, due to their terms of reference and composition, that the board would have identified the incomplete retrospective concession process. While this did not represent an opportunity to ensure that a fleet-wide risk assessment was made, it highlights the need to provide suitable guidance to manufacturing personnel to ensure that significant non-conformances are appropriately managed within the manufacturer's quality management system.

5.6 Quality assurance

The investigation examined the quality assurance function of the oversight of documentation, processes and procedures relating to the management of non-conforming parts, specifically the raising of retrospective concessions. Quality assurance, as with quality control, was a function of the engine manufacturer's Quality Management System. Evidence showed that, during the raising of the retrospective concession for the oil feed stub pipe interference bore position, a number of the existing risk controls for management of non-conforming components were ineffective.

The ineffectiveness of the risk controls influenced the handling of the non-conformance process surrounding the retrospective concession activity of April 2009. Also, the lack of a requirement for a statistical process control expert review of the statistical analysis was a risk control missing from the system that may have provided another layer of protection.

5.7 Regulatory oversight

The ATSB determined that the United Kingdom Civil Aviation Authority performed their responsibilities within the requirements of the regulatory framework by monitoring the engine manufacturer's compliance with their quality management system.

5.8 Classification of the HP/IP bearing support assembly

A review of the certification failure modes, effects and criticality analysis found that a failure of the oil feed stub pipe, amongst other failure modes, could result in a hazardous engine failure. As such, the 'unclassified' classification applied to the assembly at certification was not appropriate for the identified failure modes. Therefore, the need for appropriate levels of process control, to manage the associated hazard risks, would not have been communicated to the manufacturer's staff.

5.9 Partial first article inspection

The partial first article inspection of the HP/IP support assembly that was carried out in late April 2009, following the revision to the oil feed stub pipe counter bore datum, did not represent a specific opportunity to identify the risk to the fleet. However, it did highlight several issues that remained within the manufacturer's Hucknall Casings and Structures facility.

The report for the partial first article inspection showed that it was still considered acceptable within Hucknall Casings and Structures to use manufacturing stage drawings rather than design definition drawings during this inspection. This was despite the procedure being revised beforehand, which specifically clarified that they were not to be used for showing conformity.

It also showed that Hucknall Casings and Structures were still not able to manufacture the article to comply with the design definition, and were applying a concession to the article and continuing with manufacture. This was not in accordance with the manufacturer's group quality procedure, which required corrective action to be taken for any non-conformances and for the first article inspection to be redone.

5.10 Airframe certification standard for an uncontained engine rotor failure

The certification of the A380 was based upon the minimisation of hazards to the aircraft resulting from an uncontained engine rotor failure (UERF). Advisory Material – Joint (AMJ) 20-128A and Advisory Circular (AC) 20-128A provided a means of demonstrating compliance with Joint Aviation Requirements (JAR) 25.903(d)(1) and Code of Federal Regulations, Title 14 (14 CFR) 25.903(d)(1) respectively. The advisory material evolved from the lessons learned from the investigation of a number of UERF accidents, where the resulting damage was analysed by the relevant airworthiness authorities in response to recommendations arising from those investigations.

The amount of damage to VH-OQA resulting from the No. 2 engine UERF was greater than the modelling outlined in the advisory material. Primarily, this was the result of combined damage from multiple fragments on different trajectories that affected redundant system paths. This included systems such as the flammable fluid shut-off valves (fuel low pressure shut-off valves for the No. 1 engine, and the hydraulic shut-off valve for the No. 2 engine) and the fire extinguishing system of the No. 1 engine (refer to Damage to the aircraft in section 1.3 and Appendix B).

Notwithstanding this, the aircraft was capable of continued safe flight and it landed at Changi Airport, Singapore.

This accident represents an opportunity to extend the knowledge base relating to the hazards from UERF events and to incorporate that knowledge into the airframe certification advisory material in order to further minimise the effects of UERF on future aircraft designs.

5.11 Wing fire

The ATSB determined that there was a fire in the left wing inner fuel tank. A large hot fragment from the IP turbine disc penetrated the left inner fuel tank and was likely to have been the ignition source for the fire.

The absence of thermal damage to the internal and external wing structure and painted surfaces indicated that the associated temperature rise was relatively low, consistent with a short duration (flash) fire.

The conditions within the tank were suitable for a flash fire; however, they were not suitable to sustain the fire (refer to Appendix D).

5.12 The landing distance performance application

The inherent conservatism in the landing distance calculation by the landing distance performance application, for aircraft weights below the maximum landing weight and multiple failures, had the potential to increase flightcrew workload. Further, if the calculation incorrectly showed that the aircraft was unable to land safely at the nearest airport, it had the potential to influence crews to delay landing to reduce weight or to divert to an alternate airport.

5.13 Summary

A number of opportunities existed for the reduced wall thickness in the oil feed stub pipes, that resulted from a misaligned counter bore, to be identified before, during and after manufacture. However, these opportunities were missed by the manufacturer, resulting in a number of engines being produced and released into service with non-conforming oil feed stub pipes with consequent thin walls. Some of those oil feed stub pipe wall thicknesses were sufficiently reduced to pose a hazard to the safety of the engine and the aircraft to which they were fitted.

Ultimately, this led to the HP/IP support assembly in the No. 2 engine of VH-OQA being released into and remaining in service until the oil feed stub pipe cracked, resulting in an internal engine fire and subsequent uncontained engine rotor failure overhead Batam Island on 4 November 2010.

The ATSB also identified that this accident presents an opportunity for the aviation industry to enhance its knowledge about minimising the effects of UERF events on aircraft airframes and systems.

6 FINDINGS

The following findings are made with respect to the uncontained engine failure that occurred overhead Batam Island, Indonesia on 4 November 2010 and involved Airbus A380 aircraft, registered VH-OQA. They should not be read as apportioning blame or liability to any particular organisation or individual

6.1 Contributing safety factors

6.1.1 Disc failure during the occurrence flight

- Over time, a fatigue crack had developed in the thin-wall section of the oil feed stub pipe in the No. 2 engine to the extent that, during the occurrence flight, opening of the crack through normal movement within the engine released oil into the HP/IP buffer space.
- Auto-ignition of the oil leaking from the oil feed stub pipe created an intense and sustained fire within the HP/IP buffer space that resulted in localised heat damage to the intermediate pressure (IP) turbine disc.
- The IP turbine disc separated from the drive arm and accelerated.
- Following the separation of the IP turbine disc from the drive arm, the engine behaved in a manner that differed from the engine manufacturer's modelling and experience with other engines in the Trent family, with the result that the IP turbine disc accelerated to a rotational speed in excess of its design capacity whereupon it burst in a hazardous manner. *[Safety issue]*

6.1.2 Manufacture and release into service of engine serial number 91045

- During the manufacture of the HP/IP bearing support assembly fitted to the No. 2 engine (serial number 91045), movement of the hub during the machining processes resulted in a critically reduced wall thickness within the counter bore region of the oil feed stub pipe.
- It was probable that a non-conformance in the location of the oil feed stub pipe interference bore was reported by the coordinate measuring machine during the manufacturing process, but that the non-conformance was either not detected or not declared by inspection personnel, resulting in the assembly being released into service with a reduced wall thickness in the oil feed stub pipe.

6.1.3 Opportunity to manage the non-conforming oil feed stub pipes in the Trent 900 fleet

- The statistical analysis used to estimate maximum likely oil feed stub pipe counter bore misalignment, and resulting thin wall section, did not adequately represent the population of actual misalignments in engines already released into service, nor did it implicitly provide a level of uncertainty in the results.

- The language used to define the size of the non-conformance on the retrospective concession form did not effectively communicate the uncertainty of the statistical analysis to those assessing and approving the concession.
- The engine manufacturer did not have a requirement for an expert review of statistical analyses used in retrospective concession applications. *[Safety issue]*
- The engine manufacturer's process for retrospective concessions did not specify when in the process the Chief Engineer and Business Quality Director approvals were to be obtained. Having them as the final approval in the process resulted in an increased probability that the fleet-wide risk assessment would not occur. *[Safety issue]*
- The retrospective concession was not approved by the Chief Engineer and Business Quality Director, as required by the group quality procedures relating to retrospective concessions, denying them the opportunity to assess the risk to the in-service fleet.

6.2 Other Safety Factors

6.2.1 Release of Trent 900 engines with non-conforming oil feed stub pipe counter bores

- Numerous other engines within the Trent 900 fleet were also found to contain a critical reduction in the oil feed stub pipe wall thickness. *[Safety issue]*
- During preparation of the manufacturing process for the HP/IP bearing support assembly structure, a manufacturing datum was introduced because the location of the oil feed stub pipe counter bore could not be referenced to the design definition datum. That manufacturing datum was not constrained to the location of the oil feed stub pipe and as such could not ensure that the counter bore was concentric with the stub pipe, as the designers had intended.
- The engine manufacturer did not require its manufacturing engineers to consult with the design engineers to ensure that design intent would be maintained when introducing manufacturing datums. *[Safety issue]*
- The use by an inspector, during the first article inspection process, of the manufacturing stage drawings to verify the oil feed stub pipe counter bore features precluded the inspection from showing that the manufacturing process could produce an item that conformed to the design definition, or the intention of the design.
- The procedure for the first article inspection process contained ambiguities that resulted in an interpretation whereby the use of the manufacturing stage drawings was deemed to be acceptable. *[Safety issue]*
- A culture existed within the engine manufacturer's Hucknall facility where it was considered acceptable to not declare what manufacturing personnel

determined to be minor non-conformances in manufactured components. *[Safety issue]*

- The coordinate measuring machine was programmed to measure the location of the oil feed stub pipe interference bore with respect to the manufacturing datum, instead of the design definition datum as specified on both the design and manufacturing stage drawings. *[Safety issue]*
- During the production of a number of HP/IP bearing support assemblies, the coordinate measuring machine identified a non-conformance in the location of the oil feed stub pipe interference bore.
- It was likely that when making the determination that the non-conforming HP/IP bearing support assemblies were acceptable for use, the manufacturing personnel did not know that the coordinate measuring machine referenced a different datum to the design definition drawings and unknowingly released thin-walled pipes into service based on an alternative (wire gauge) measurement method.
- The engine manufacturer's group quality procedures did not provide any guidance on how manufacturing personnel were to determine the significance of a non-conformance, from a quality assurance perspective. *[Safety issue]*
- When the retrospective concession was processed, the non-conformance was not reported to the Quality Review Board because manufacturing personnel did not identify it as a significant non-conformance.
- The manufacturer's classification, relating to the criticality of failure, of the HP/IP bearing support assembly was inappropriate for the effects of a fire within the buffer space and hence, the requirement for an appropriate level of process control was not communicated to the manufacturing staff. *[Safety issue]*
- The partial first article inspection carried out in 2009 on the HP/IP support structure was not compliant with the engine manufacturer's procedures.

6.2.2 Minimisation of hazards resulting from an uncontained engine rotor failure

- The evolution of the current advisory material relating to the minimisation of hazards resulting from uncontained engine rotor failures was based on service experience, including accident investigation findings. The damage to Airbus A380-842 VH-OQA exceeded the modelling used in the UERF safety analysis and, therefore, represents an opportunity to incorporate any lessons learned from this accident into the advisory material. *[Safety issue]*

6.2.3 Landing distance performance application

- The calculation method in the aircraft manufacturer's landing distance performance application was overly conservative and this could prevent the calculation of a valid landing distance at weights below the maximum landing weight with multiple system failures. *[Safety issue]*

6.3 Other Key Findings

- A large fragment of the hot intermediate pressure turbine disc passed through the left inner fuel tank, which resulted in a short-duration low-intensity fire. Conditions within the fuel tank were not sufficient to sustain combustion and a hazardous situation did not result.
- There was a short-duration oil fire in the No. 2 engine lower nacelle that self-extinguished, resulting only in localised damage to the nacelle.
- Despite significant system and structural damage following the uncontained engine rotor failure, the aircraft was capable of continued safe flight and landing.
- The duration of the CVR recording was insufficient to capture the entire occurrence that included the engine failure and subsequent ground operation up to passenger disembarkation.
- The crew’s decision to perform a precautionary disembarkation via the stairs likely provided the safest option, particularly given the low immediate safety threat and the elevated risks associated with an emergency evacuation into a potentially hazardous external environment.
- The flight and cabin crew managed the event as a competent team in accordance with standard operating procedures and practices.

7 SAFETY ACTION

The safety issues identified during this investigation are listed in the Findings and Safety Actions sections of this report. The Australian Transport Safety Bureau (ATSB) expects that all safety issues identified by the investigation should be addressed by the relevant organisation(s). In addressing those issues, the ATSB prefers to encourage relevant organisation(s) to proactively initiate safety action, rather than to issue formal safety recommendations or safety advisory notices.

7.1 Preliminary safety actions

Immediately after, and then progressively during the preliminary stages of the investigation, before any specific safety issues had been identified, the ATSB was advised by a number of involved organisations of various proactive safety actions.

7.1.1 Qantas Airways Ltd grounding of their A380 fleet

Grounding

On 4 November 2010, after notification of the uncontained failure of the No.2 engine in VH-OQA, Qantas Airways Ltd (Qantas) elected to immediately cease operations with their fleet of A380 aircraft. This grounding remained in effect until further information on the sequence of events leading up to the uncontained engine failure was available that would inform Qantas of the associated safety risk to its operations.

Return to service

Qantas commenced the reintroduction of its A380 fleet into service on 27 November 2010. This followed the airline's own investigation and analysis of the occurrence as ratified by the engine manufacturer and agreed by the Civil Aviation Safety Authority (CASA).

With respect to the decision by Qantas to commence returning their A380 aircraft to service, on 30 November 2010 CASA advised the ATSB that:

> Qantas provided CASA with extensive documentation to support the planned return to service as well as a number of briefings by key personnel. Qantas' plans as presented and analysed by CASA's technical experts detailed a conservative approach and called for the implementation of additional safety mitigation strategies above the requirements of the engine manufacturer.
>
> CASA is satisfied that Qantas' decision is appropriate.

Only flights that did not require the use of maximum engine thrust were permitted. That decision was based on advice from the engine manufacturer.

7.1.2 Rolls-Royce plc - Trent 900 engine inspections

Non-modification service bulletin 72-G589

On 4 November 2010, Rolls-Royce issued non-modification service bulletin (NMSB) 72-G589 that required a series of checks on all in-service (or on-wing) Trent 900 engines. This included a one-time inspection of the turbine blades and the high pressure/intermediate pressure (HP/IP) turbine bearing area in each engine prior to the next flight. Any indication of oil in the buffer zone in the HP/IP bearing support structure was of particular interest.

NMSB 72-AG590

On 10 November 2010, Rolls-Royce issued NMSB 72-AG590, requiring the inspection of all Trent 900 series engines for evidence of oil leaks into specific turbine area components.

On 12 November 2010, Rolls-Royce advised that:

> … process of inspection will continue and will be supplemented by the replacement of the relevant module according to an agreed programme.

On 18 November 2010, Rolls-Royce issued Revision 2 of NMSB 72-AG590, detailing further Trent 900 engine inspections, including for defects in a number of turbine area oil and air feed pipes.

7.1.3 Airbus SAS – Trent 900 engine inspections

On 5 November 2010 Airbus SAS (Airbus), in response to this occurrence, released All Operators Telex A380-728002 that required all operators of A380 aircraft to comply with the engine inspection requirements of Rolls-Royce NMSB 72-G589.

Airbus also issued a number of Accident Investigation Telexes to all of its A380 customers informing them of the progress of its own investigation and of the details of the aircraft's recovery at Changi Airport, Singapore, and confirming their intent to continue as an adviser to the ATSB investigation.

7.1.4 European Aviation Safety Agency

On 10 November 2010, the European Aviation Safety Agency (EASA) issued emergency airworthiness directive EASA AD: 2010-0236-E in respect of the operation of the Rolls-Royce RB211 Trent 900 series engines. The airworthiness directive required the periodic inspection of the HP/IP engine structure for any abnormal oil leakage. If any discrepancy was identified, the further operation of that engine was prohibited.

That action by EASA was based on a preliminary analysis of the circumstances of the engine failure in VH-OQA by Rolls-Royce. That analysis had indicated that an oil fire in a cavity within the HP/IP structure may have caused the failure of the intermediate pressure turbine disc.

A full copy of EASA AD: 2010-0236-E is available at:

http://ad.easa.europa.eu/ad/2010-0236-E

The EASA emergency AD was superseded on 22 November 2010 by AD 2010-0242-E that incorporated the contents of Rolls-Royce NMSB 72-AG590 (Revision 2). AD 2010-0242-E is available at:

http://ad.easa.europa.eu/ad/2010-0242-E

7.1.5 Australian Transport Safety Bureau

On 30 November 2010 the ATSB had, in close consultation with Rolls-Royce and the UK Air Accidents Investigation Branch, established that the occurrence was directly related to the fatigue cracking of an oil feed stub pipe within the No.2 engine's HP/IP bearing support structure. The ATSB identified the following safety issue:

Safety issue

Misaligned stub pipe counter-boring is understood to be related to the manufacturing process. This condition could lead to an elevated risk of fatigue crack initiation and growth, oil leakage and potential catastrophic engine failure from a resulting oil fire.

Action taken by the ATSB

On 1 December 2010, the ATSB issued the following safety recommendation to Rolls-Royce.

ATSB safety recommendation AO-2010-089-SR-012

The Australian Transport Safety Bureau recommends that Rolls-Royce plc address this safety issue and take actions necessary to ensure the safety of flight operations in transport aircraft equipped with Rolls-Royce plc Trent 900 series engines.

A full copy of the ATSB safety recommendation is available at:

www.atsb.gov.au/publications/investigation_reports/2010/aair/ao-2010-089

Initial action taken by Rolls-Royce

In response to the developing understanding of this safety issue, on 2 December 2010 Rolls-Royce issued NMSB 72-G595 to operators of the Trent 900 engine, which required the specialised examination, measurement and reporting of the stub pipe counter bore geometry in these engines. No assessment or engine rejection criteria were included in the NMSB.

A 20 flight cycle compliance limitation was specified for the completion of the oil feed stub pipe examination.

ATSB assessment

Despite the initial Rolls-Royce action to release NMSB72-G595, the ATSB was concerned that the bulletin did not place assessment and engine rejection criteria on the measurement of the stub pipe counter bore geometry. In addition, the ATSB did not consider the 20 cycle limitation as adequately addressing this safety issue. The ATSB consulted with CASA, who initiated the actions as detailed below.

Action taken by CASA

On 2 December 2010, CASA issued a maintenance direction to Qantas under Regulation 38 of the *Civil Aviation Regulations 1988.* That direction required that Qantas:

> (a) Comply with Rolls-Royce plc Service bulletin number 72-G595 subsequent and any amendment or revision of it, within two cycles from the issue of this direction;
>
> (b) In the event abnormal or eccentric counter-boring of the tubes described in the service bulletin is identified, this must be recorded as a major defect of the engine;
>
> (c) Upon completion of compliance with the service bulletin an entry shall be made in the aircraft's maintenance records stating what actions were taken to comply with the service bulletin and this direction;
>
> (d) Upon completion of compliance with the service bulletin a written report shall be furnished to [CASA] stating how the service bulletin and this direction were complied with and the outcome of compliance with the service bulletin.

ATSB assessment of the CASA action

The ATSB is satisfied that the action taken by CASA adequately addresses the immediate safety of flight concerns in respect of Qantas operation of A380 aircraft equipped with Trent 900 series engines. Therefore the ATSB makes no recommendation in relation to this issue.

Further action taken by Rolls-Royce in response to the safety recommendation

On 2 December 2010 Rolls-Royce issued Revision 1 to NMSB 72-G595. This revision incorporated assessment and engine rejection criteria for the measurement of potential counter bore misalignments, and in particular, a tightening of the compliance from 20 to two flight cycles.

ATSB assessment of the Rolls-Royce action

The ATSB is satisfied that the action taken by Rolls-Royce adequately addresses the immediate safety of flight concerns in respect of Qantas operation of A380 aircraft equipped with Trent 900 series engines.

Action taken by Qantas

On 2 December 2010, Qantas advised that:

> ...in response to Service Bulletin RB211-72-G595 (Revision 1), and in line with ATSB Safety Recommendation AO-2010-089-SR-012, Qantas will conduct a focused borescope measurement inspection of the HP/IP turbine bearing support structure oil feed tube for concentricity of the counter-bore and inspection of the related components on its RB211 Trent 900 series engines. The inspection results will be sent to Rolls Royce for evaluation. Rolls Royce will then provide Qantas with formal confirmation as to the serviceability of the engine.

> These inspections will take place within the next 24 hrs on engines in place on A380 aircraft currently in service, and before further flight on engines on aircraft not yet returned to service.

ATSB assessment of the Qantas action

The ATSB is satisfied that the action taken by Qantas adequately addresses the immediate safety of flight concerns in respect of the operation of its A380 aircraft equipped with Trent 900 series engines. Therefore the ATSB makes no recommendation in relation to this issue.

7.2 Subsequent safety actions

The ATSB's understanding of the occurrence event and the factors contributing to it evolved over the course of the investigation. Further contributing safety factors and other safety factors were identified and the associated safety actions are set out below.

7.2.1 Intermediate pressure turbine overspeed and burst following failure of drive arm due to internal engine fire

Safety issue

Following the separation of the IP turbine disc from the drive arm, the engine behaved in a manner that differed from the engine manufacturer's modelling and experience with other engines in the Trent family, with the result that the IP turbine disc accelerated to a rotational speed in excess of its design capacity whereupon it burst in a hazardous manner.

Action taken by Rolls-Royce

On 3 December 2010, Rolls-Royce released NMSB RB.211-73-AG639, advising Trent 900 operators of the introduction of a revised standard of engine management software that featured an IP turbine overspeed protection system (IPTOS).

The IPTOS was intended to detect engine conditions with the potential to lead to an IP turbine over speed. In response, IPTOS would shut down the engine before the IP turbine disc reached its critical burst speed. Shaft breaks and disc separation, such as occurred in VH-OQA can occur for mechanical reasons such as component fatigue, an over torque being applied to the shaft or a manufacturing defect, or by localised heating such as from an oil-fed fire. During the course of the investigation into the No.2 engine failure in VH-OQA, the ATSB was provided a detailed summary of the IPTOS protection system, which works on the following logic:

> The first element arms the system, and is based on detecting a prescribed rate of temperature change of turbine cooling air at the front (TCAF) or rear (TCAR) of the IP turbine. Such rates of change indicate that a fire has developed within the engine that may lead to localised heating of the IP turbine disc or shaft.

> Once armed, if an abnormally high rapid rate of compressor deceleration is detected, a shaft break or disc separation is indicated and the EEC (engine electronic controller) will instantly shut off the fuel, open all the bleed valves and close the variable stator vanes.
>
> Flight crew are alerted to an IP shaft failure through a flight deck annunciator alert that raises the message 'ENG FAIL-SHAFT FAILURE'.

Rolls-Royce reported that the engine EEC software upgrade that included the IPTOS functionality was incorporated across the Trent 900 fleet by 6 December 2010.

Action taken by Airbus

On 9 December 2010, in conjunction with the release of the Trent 900 IPTOS as advised in Rolls-Royce NMSB RB.211-73-AG639, Airbus released service bulletin A380-73-8011 to operators of Trent 900 equipped A380 aircraft. This bulletin required the IPTOS to be installed across the Trent 900-equipped fleet.

Action taken by EASA

On 13 December 2010, EASA issued airworthiness directive AD: 2010-0262 in respect of modifying the Trent 900 EEC software by incorporating the IPTOS logic, as detailed in Rolls-Royce NMSB RB.211-73-AG639. The airworthiness directive required all Trent 900 engines to be modified within 10 flight cycles.

A full copy of EASA AD: 2010-0262 is available at:

http://ad.easa.europa.eu/ad/2010-0262

ATSB assessment of Rolls-Royce, Airbus and EASA safety action

The ATSB is satisfied that the action taken by Rolls-Royce, Airbus and EASA adequately addresses the safety issue in respect of the risk of an IP turbine overspeed and burst. Therefore the ATSB makes no recommendation in relation to this issue.

7.2.2 Release of non-conforming oil feed stub pipes into service

Safety issue

Numerous other engines within the Trent 900 fleet were also found to contain a critical reduction in the oil feed stub pipe wall thickness.

Action taken by Rolls-Royce

In December 2010, in response to this safety issue, Rolls-Royce focussed on assessing the oil feed stub pipe counter bore geometries across the Trent 900 engine fleet. Following a stress analysis and numerical modelling of the stub pipe counter bore geometry, a minimum calculated stub pipe wall thickness acceptance limit of 0.5 mm was established in order for engines to remain in service. Any engine with a stub pipe thickness below this limit was removed from service. Wall thicknesses were established across the fleet using either:

- a specialist borescope visual inspection and measurement of the oil feed stub pipe counterbore (NMSB 72-G595)
- examination of a ‘replicast’ (a rubber-like mould) of the oil feed stub pipe’s internal features (Technical Variance 108953)
- a borescope inspection to identify the serial numbers of relevant HP/IP bearing support structures fitted to Trent 900 engines (NMSB 72-643)
- existing manufacturing data.

The borescope inspection technique introduced by NMSB 72-G595 was successful in identifying in-service oil feed stub pipes with reduced wall sections. However, based on the results, the tolerances were not sufficient to provide confidence for accurate service management.

The available manufacturing data was analysed by Rolls-Royce from early December 2010 to calculate the oil feed tube wall thickness in some B-standard HP/IP bearing support structures and all C-standard structures. Rolls-Royce elected to withdraw all of the A-production standard HP/IP bearing support structures due to their manufacturing records being unavailable.

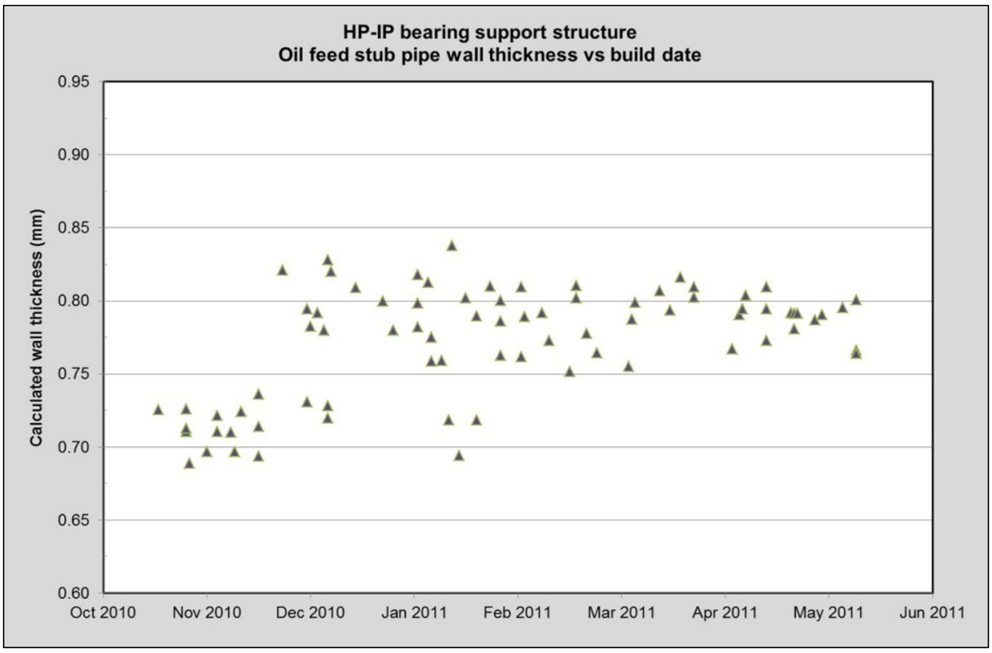
As a result of this action, 40 engines were removed from service having been identified with an oil feed stub pipe wall thickness of less than 0.5 mm. This resulted in the removal from service of the following engines:

- 14 engines with an A-production standard HP/IP bearing support structure
- 23 engines with a B-production standard HP/IP bearing support structure. Of these, five were removed from Qantas-operated A380 aircraft
- Three engines with a C-production standard HP/IP bearing support structure.

Following the occurrence, the stub pipe wall thickness production limit was restricted to 0.70 mm for all newly manufactured engines (Figure 93). The revised limit was introduced in December 2010, along with enhanced techniques for the measurement of critical dimensions within the counter bore region.

Rolls-Royce production data showed that, after the introduction of this revised limit in December 2010, the quality control of the manufacture of the HP/IP bearing support structure at the Hucknall facility had improved (Figure 93).

Figure 93: Production data for the HP/IP bearing support structures that were manufactured at the Hucknall facility



ATSB assessment

The ATSB is satisfied that the action taken by Rolls-Royce adequately addresses this safety issue and therefore makes no recommendation.

7.2.3 Consultation between manufacturing engineers and design engineers to ensure maintenance of design intent

Safety issue

The engine manufacturer did not require its manufacturing engineers to consult with the design engineers to ensure that design intent would be maintained when introducing manufacturing datums.

Action taken by Rolls-Royce

Rolls-Royce advised the ATSB that:

> In January 2011, a revision to GQP [group quality procedure] ‘Manufacturing Technical Package’ was issued that provided greater structure and guidance for buy-off [manufacturing acceptance] between design and manufacturing personnel.

ATSB assessment

The ATSB is satisfied that the action taken by Rolls-Royce adequately addresses this safety issue and therefore makes no recommendation.

7.2.4 Use of manufacturing stage drawings for the first article inspection

Safety issue

The procedure for the first article inspection process contained ambiguities that resulted in an interpretation whereby the use of the manufacturing stage drawings was deemed to be acceptable.

Action taken by Rolls-Royce

As part of their ongoing quality assurance activities, Rolls-Royce had previously revised the group quality procedure for first article inspections to explicitly preclude the use of manufacturing stage drawings during the inspection. In addition, Rolls-Royce advised the ATSB that in January 2011:

> A revision to the First Article Inspection (FAI) process was also issued that provided further guidance to personnel if the design intent could not be met.

ATSB assessment

The ATSB is satisfied that the actions taken by Rolls-Royce adequately addresses this safety issue and therefore makes no recommendation.

7.2.5 Culture of acceptance of ‘minor’ non-conforming components during manufacture at the Rolls-Royce Hucknall facility

Safety issue

A culture existed within the engine manufacturer's Hucknall facility where it was considered acceptable to not declare what manufacturing personnel determined to be minor non-conformances in manufactured components.

Action taken by Rolls-Royce

During the ATSB’s investigation, Rolls-Royce advised that, in June 2007, an independent product process audit was conducted at the manufacturer’s facility at Hucknall. The audit found that items being produced at Hucknall contained high levels of non-conformance that were not being reported through the existing non-conformance management process.

The manufacturer reported the following safety actions had been taken in response to those audit findings:

> All output from Hucknall [Casings and Structures] (HCAS) was stopped.
>
> Civil and Defence engineering teams were engaged to assess any non-conformance in order to identify anything that could affect fit, form or function.
>
> The CAA were informed of a ‘compliance issue at Hucknall’.

> All HCAS employees were briefed in July 2007, and again in October/November 2007. The key message to employees was to emphasise the concession process requirement that all non-conformances to the engineering drawings must be identified and assessed.
>
> A major quality investigation (MQI) was raised on 15 August 2007 to investigate systems, processes and behaviours.

These actions were completed in June 2008.

ATSB assessment

The ATSB is satisfied that the action taken by Rolls-Royce adequately addresses this safety issue and therefore makes no recommendation.

7.2.6 Difference between drawing datum and coordinate measuring machine datum

Safety issue

The coordinate measuring machine was programmed to measure the location of the oil feed stub pipe interference bore with respect to the manufacturing datum, instead of the design definition datum as specified on both the design and manufacturing stage drawings.

Action taken by Rolls-Royce

Rolls-Royce advised the ATSB that in July 2008, the design definition and manufacturing stage drawings had been changed to use the inner diameter of oil feed stub pipe as the datum for the oil feed stub pipe counter bore. During March and April 2009 both the manufacturing process and the coordinate measuring machine program were changed to use that revised datum. Use of the original manufacturing datum was discontinued.

ATSB assessment

The ATSB is satisfied that the action taken by Rolls-Royce adequately addresses this safety issue and therefore makes no recommendation.

7.2.7 Expert review of statistical analysis in support of retrospective concessions

Safety issue

The engine manufacturer did not have a requirement for an expert review of statistical analyses used in retrospective concession applications.

Action taken by Rolls-Royce

On 25 May 2011, Rolls-Royce advised the ATSB that a major corrective action, which arose from an internal major quality investigation, was to remove the existing retrospective concession procedures from their quality system, and replace them with a new Global Process titled *Management of Undeclared Non-Conformance in Delivered Product*.

The global process was developed to ensure an improved and more consistent approach across the company when it is identified that parts containing undeclared and non-conforming features have been released to the customer for entry into service. The global process included a technical review by a statistic expert.

The global process was incorporated into the engine manufacturer's quality management system on 4 July 2011.

ATSB assessment

The ATSB is satisfied that the action taken by Rolls-Royce adequately addresses this safety issue and therefore makes no recommendation.

7.2.8 Chief Engineer and Business Quality Director review of retrospective concessions

Safety issue

The engine manufacturer's process for retrospective concessions did not specify when in the process the Chief Engineer and Business Quality Director approvals were to be obtained. Having them as the final approval in the process resulted in an increased probability that the fleet-wide risk assessment would not occur.

Action by Rolls-Royce

On 25 May 2011, Rolls-Royce advised the ATSB that a major corrective action, which arose from an internal major quality investigation, was to remove the existing retrospective concession procedures from their quality system, and replace them with a new Global Process titled *Management of Undeclared Non-Conformance in Delivered Product*.

The global process was developed to ensure an improved and more consistent approach across the company when it is identified that parts containing undeclared and non-conforming features have been released to the customer for entry into service. The global process requires the Chief Engineer and the Business Quality Director to be involved in the process at a much earlier stage to ensure that the fleet-wide risk assessment is conducted.

The global process was incorporated into the engine manufacturer's quality management system on 4 July 2011.

Additionally, Rolls-Royce carried out an independent audit and review of the retrospective concession activity for the 2009 to 2011 period. The review revealed that only 7 out of 138 retrospective concessions that had been raised within the Civil Large Engine business unit were compliant with the engine manufacturer's procedures.

All non-compliant retrospective concessions that had been raised since 2009 were subsequently identified and revalidated by the appropriate Chief Engineer and Business Quality Director. Other than the retrospective concession regarding the misalignment of the oil feed stub pipe counter bores, no safety concerns were identified and no in-service activity required.

ATSB assessment

The ATSB is satisfied that the actions taken by Rolls-Royce adequately addresses this safety issue and therefore makes no recommendation.

7.2.9 Reporting of significant non-conformances to the quality review board

Safety issue

The engine manufacturer's group quality procedures did not provide any guidance on how manufacturing personnel were to determine the significance of a non-conformance, from a quality assurance perspective.

Action taken by Rolls-Royce

On 25 May 2011, Rolls-Royce advised the ATSB that a major corrective action, which arose from an internal major quality investigation, was to remove the existing retrospective concession procedures from their quality system, and replace them with a new Global Process titled *Management of Undeclared Non-Conformance in Delivered Product*.

The global process was developed to ensure an improved and more consistent approach across the company when it is identified that parts containing undeclared and non-conforming features have been released to the customer for entry into service. The global process includes involvement of quality assurance personnel from the initiation of the process through to its completion.

The global process was incorporated into the engine manufacturer's quality management system on 4 July 2011.

ATSB assessment

The ATSB is satisfied that the action taken by Rolls-Royce adequately addresses this safety issue and therefore makes no recommendation.

7.2.10 Classification of the HP/IP bearing support assembly

Safety issue

The manufacturer's classification, relating to the criticality of failure, of the HP/IP bearing support assembly was inappropriate for the effects of a fire within the buffer space and hence, the requirement for an appropriate level of process control was not communicated to the manufacturing staff.

Action taken by Rolls-Royce

In February 2012, Rolls-Royce advised the ATSB of the initiation of a major quality investigation into Trent 900 failure modes effects and criticality analysis

(FMECA) inaccuracies. That investigation was commenced after it was identified that the potential for an IP turbine disc failure was not reflected in the Trent 900 FMECA certification documentation as a hazardous event. Safety action by Rolls-Royce included:

> ...the Trent 900 FMECAs have been reviewed and updated in light of the QF32 event. The Oil System and Transmissions FMECAs have now been updated.

The manufacturer advised that as a result of the review of the FMECA, it had reclassified the HP/IP bearing support from ‘unclassified’ to ‘reliability sensitive’. This change in classification would require the appropriate level of process control.

ATSB assessment

The ATSB is satisfied that the action taken by Rolls-Royce adequately addresses this safety issue and therefore makes no recommendation.

7.2.11 Landing distance calculation at aircraft weights below the A380 maximum landing weight

Safety issue

The calculation method in the aircraft manufacturer’s landing distance performance application was overly conservative and this could prevent the calculation of a valid landing distance at weights below the maximum landing weight with multiple system failures.

Action taken by Airbus

On the 28 September 2011, in response to this safety issue, Airbus advised the ATSB of the following safety action:

> Airbus has developed a product improvement with the in-flight landing distance application OIS 2B+, available to A380 Operators 4 October 2011 with SB A380-46-8046, that ensures consistency of computation results whatever the landing weight,
>
> Airbus has informed all A380 Operators at March 2011 Flight Safety Conference and at May 2011 Performance and Operations Conference.
>
> A further product improvement will be introduced with future OIS standards planned to be available by the third quarter of 2013 that will optimize performance calculation and therefore improve consistency of in-flight landing distance prediction to actual aircraft capability.

ATSB assessment

The ATSB is satisfied that the action taken by Airbus adequately addresses this safety issue and therefore makes no recommendation.

7.2.12 Airframe certification standards in the case of an uncontained engine rotor failure

Safety issue

The evolution of the current advisory material relating to the minimisation of hazards resulting from uncontained engine rotor failures was based on service experience, including accident investigation findings. The damage to Airbus A380-842 VH-OQA exceeded the modelling used in the UERF safety analysis and, therefore, represents an opportunity to incorporate any lessons learned from this accident into the advisory material.

Action taken by ATSB

As a result of the identified safety issue, coincident with the release of this investigation report, the ATSB has issued the following safety recommendations to the European Aviation Safety Agency and the United States Federal Aviation Administration.

<u>ATSB safety recommendation AO-2010-089-SR-039</u>

The Australian Transport Safety Bureau recommends that the European Aviation Safety Agency, in cooperation with the US Federal Aviation Administration, review the damage sustained by Airbus A380-842, VH-OQA following the uncontained engine rotor failure overhead Batam Island, Indonesia, to incorporate any lessons learned from this accident into the advisory material.

<u>ATSB safety recommendation AO-2010-089-SR-040</u>

The Australian Transport Safety Bureau recommends that the US Federal Aviation Administration, in cooperation with the European Aviation Safety Agency, review the damage sustained by Airbus A380-842, VH-OQA following the uncontained engine rotor failure overhead Batam Island, Indonesia, to incorporate any lessons learned from this accident into the advisory material

7.3 Proactive actions

7.3.1 Action taken by Airbus

Although not specifically associated with any of the safety issues identified by the ATSB investigation, Airbus advised on 9 March 2013 of the following software enhancements for the A380 aircraft.

Trim tank availability

A software upgrade to the A380 ECAM was released to all A380 operators via Service Bulletin A380-42-8022 on 25 April 2013. The upgrade further emphasises the status and availability of fuel trim tank services.

Operator compliance with the service bulletin was ‘recommended’ by Airbus.

Electrical generation and distribution

Airbus' analysis of the available data after the occurrence established that when the aircraft electrical system (feeder cables) were damaged from the liberated engine debris, the No. 2 generator line contactor (GLC2) physically opened in order to isolate the Primary Electrical Power Distribution Center (PEPDC) from potential damage (that is, from short circuits or current spikes). Airbus advised the ATSB that the associated current monitoring system onboard the aircraft had actually detected that the GLC2 remained in a closed position, even though the contacts had physically opened.

Airbus advised the ATSB that in the first-quarter of 2014, software enhancements will be available for the A380 electrical generation, distribution and monitoring system. The introduction of a revised software, 'GGPCU Standard-18', is intended to enhance the capabilities of the of the aircraft electrical system. The software is designed to monitor the electrical current at the feeder block to which each GLC is connected (as close to the GLC as possible), such that even in the hypothetical situation of a short-circuit:

- residual current from the VFG will no longer affect the monitoring of the GLC, with reliable detection of the GLC in the open or closed state
- recovery of the associated AC BUS bar will no longer be prevented.

APPENDIX A: ELECTRONIC CENTRALISED AIRCRAFT MONITORING PROCESS WORKFLOW AND TIMELINE

Introduction

This appendix outlines the Electronic Centralised Aircraft Monitoring (ECAM)[104] procedures that the flight crew most likely were presented with, and then followed after the No. 2 engine failed. A brief description of the A380 cockpit layout and information display is provided, followed by a description of ECAM, its function and use, and how warnings and associated system messages are displayed to the crew. Finally, a list of warnings as recalled by the flight crew, followed by the most likely ECAM workflow as recreated using data sources on board the aircraft and other associated documentation, is highlighted.

Flight crew – aircraft system monitoring and interaction

The A380 cockpit was designed to allow the aircraft to be operated by two flight crew members. There was additional seating for up to another three flight crew members behind the two operating crew members. Information pertinent to the operation of the aircraft was presented on a number of displays. The flight deck instrumentation layout is displayed in Figure A1.

[104] Electronic Centralised Aircraft Monitoring (ECAM), which is a process undertaken by a number of aircraft sub-systems. It is not a specific system by itself.

Figure A1: A380 flight deck instrumentation layout

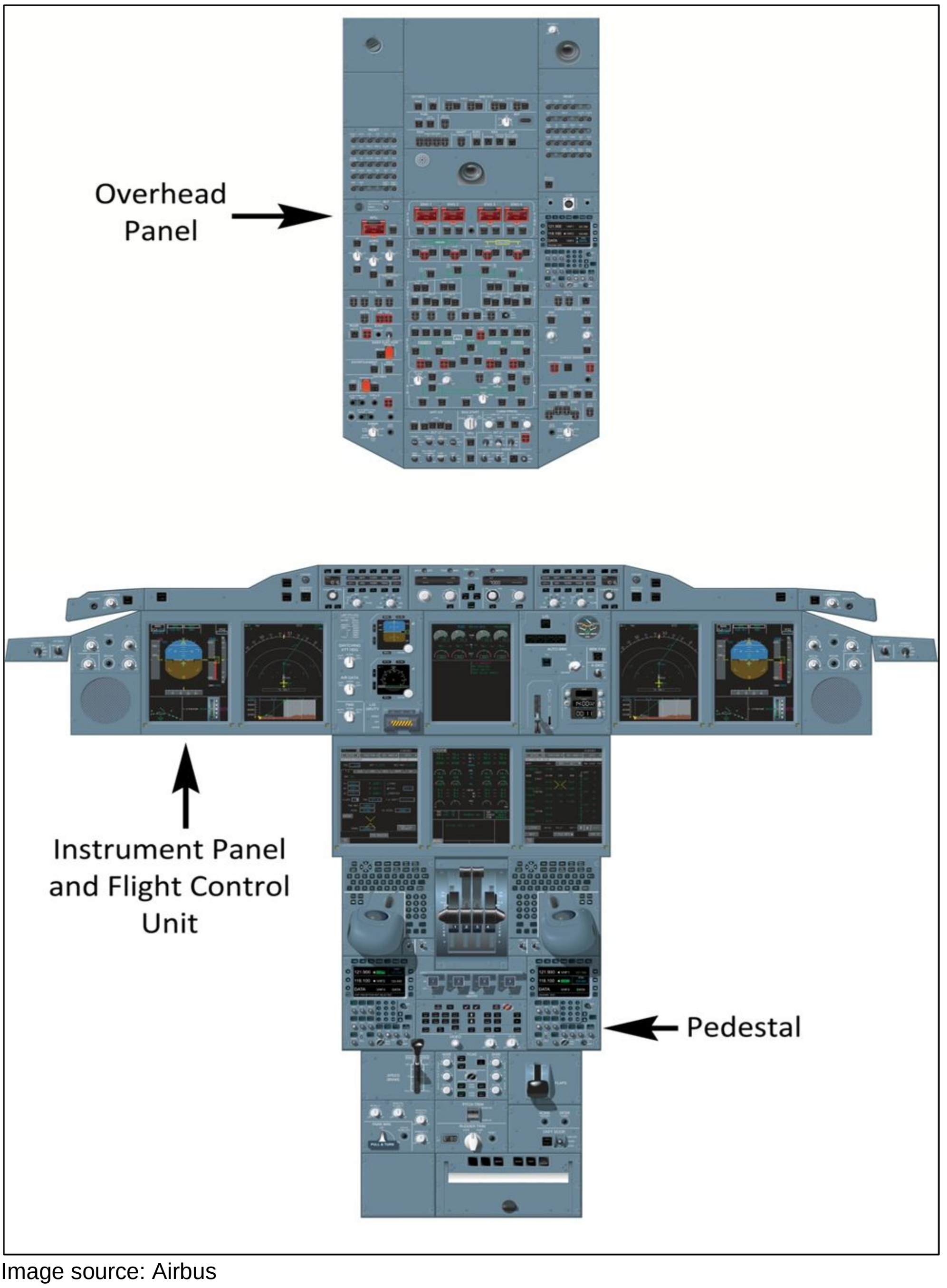


Image source: Airbus

There were eight display units that presented various information to the flight crew on the operation of the aircraft and its systems. These were:

- Two Primary flight displays (PFD) that provided short-term flight information such as flight attitude, altitude and speed information.
- Two Navigation displays (ND), which provided medium and long-term navigation information.
- One Engine warning display (EWD), which had two displays. In the upper section of this display, engine operating parameters were shown. The lower section was the warning display (WD) and could display ECAM checklist items, alerts, procedures and memos to assist the crew.

- One System display (SD), which was of three sections. The upper section displayed aircraft system information and the middle section permanently displayed aircraft information. This included the external temperature, time, aircraft weight and centre of gravity and fuel on board. The lower section displayed the ‘mailbox’, which allowed the crew to communicate with air traffic control (ATC) via controller/pilot data link communications (CPDLC).
- Two Multi-function displays (MFD) that provided the crew with an interface with the flight management system (FMS), ATC communications and the radar surveillance pages. The MFD also provided a backup input system for the aircraft’s flight control unit (FCU).

Each of the displays could be re-selected to display any of the other display units in the event of a single display unit failure. The normally-selected position of each of the displays is at Figure A2.

Figure A2: Normal display locations

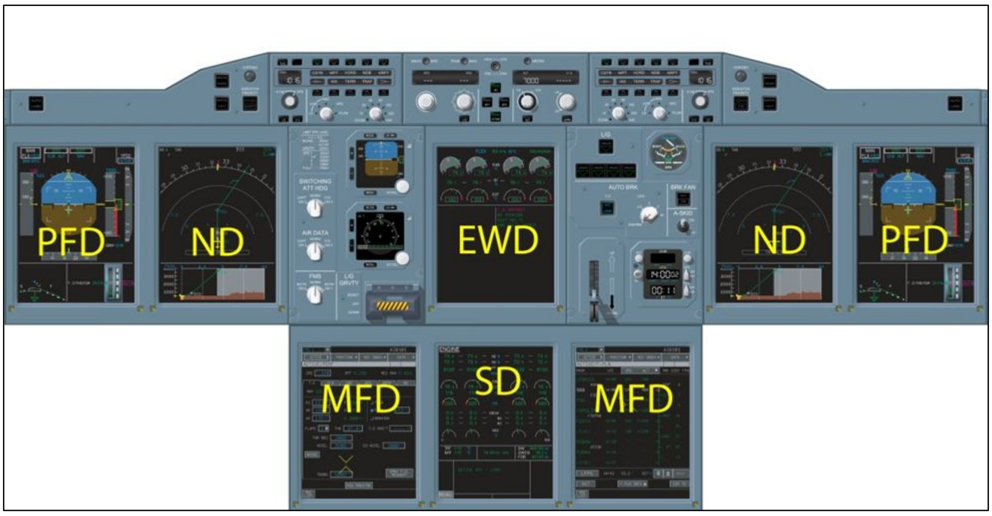


Image source: Airbus

ECAM description

ECAM is a function that continuously monitors the aircraft, its systems and operating parameters and provides necessary information and warnings to the flight crew, in both normal and abnormal operations.

ECAM integrates this information from a number of onboard systems, including (Figure A3):

- two flight warning systems (FWS)
- one ECAM control panel (ECP)
- four master warning and caution lights
- four loudspeakers.

The information is displayed to the crew via the:

- EWD for normal checklists, abnormal and emergency procedures, aircraft limitations and memos

- SD for information relating to specific aircraft systems, the current aircraft status, and permanently-displayed aircraft data
- lower section of the PFD for limitations that have an immediate effect on the flight.

Figure A3: ECAM control locations within the flight deck instrument layout

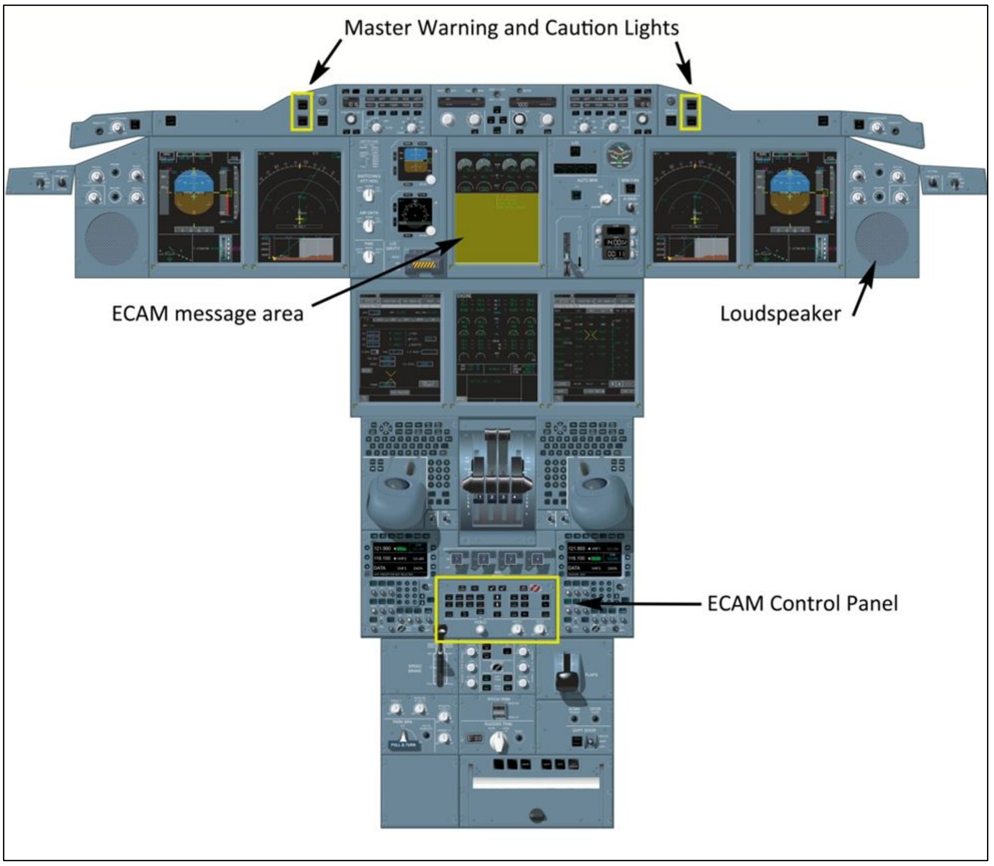


Image source: Airbus

Operating Modes

ECAM will automatically select one of four different modes:

- In the normal mode of operation, ECAM automatically displays systems and memos to the crew according to the applicable flight phase[105] at that time.
- In failure mode, when a system failure is detected, ECAM automatically displays the appropriate emergency or abnormal procedure with its associated system display.
- The advisory mode provides the crew with a system display when a monitored system parameter starts to move out of its normal position and

[105] To minimise distractions during specific important flight phases (such as during takeoff and the initial climb), many abnormal and emergency procedures detected by ECAM are inhibited from being displayed. There are 12 separate flight phases on the A380, commencing when electrical power is first switched on and ending 5 minutes after the last engine is shut down at the end of the flight.

reaches predetermined triggering levels (for example, with an increasing oil temperature).

- In manual mode the crew are able to use ECAM to manually select and display any applicable system checklist or procedure to assist them with operating the aircraft.

Overall, the ECAM continuously monitored the aircraft and its systems and, when conditions were detected that were outside normal monitoring parameters, it automatically provided the crew with a warning or a caution depending upon the severity of the condition. The appropriate procedure to deal with the condition was simultaneously displayed on the lower part of the EWD and the appropriate aircraft system page on the SD.

ECAM Alerts

There were three levels of warnings and cautions within ECAM, each based upon the associated operational consequences of the detected failure. These levels of failure included:

- Level 3 – the highest level of failure indicating an impact on the safety of the aircraft. This failure indication was colour coded red and had an associated aural warning (master warning). The recommended crew action was to respond immediately to the warning.
- Level 2 – this failure indication defined an abnormal condition to the crew. It was colour coded amber with a different aural warning to that of a Level 3 failure (master caution). The recommended crew action was to be aware of the condition and then take appropriate action.
- Level 1 – defined as a degradation in an aircraft system, this failure indication was also colour coded amber but there was no associated aural warning (master caution visual only). The recommended crew action was to be aware of the condition.

When multiple failures or out-of-limits conditions were detected by ECAM they were prioritised according to the programmed ECAM logic.[106] In that event, the highest priority procedure relating to the sensed condition was displayed first.

When ECAM detects multiple system failures and has displayed them to the crew (abnormal procedure), the in-built ECAM logic meant that the crew are presented with the most important procedures first. Airbus operational philosophy means that each abnormal procedure, in order of importance, has to be completed by the flight crew before they move onto the next procedure. If the flight crew were attending to a lower priority ECAM, a higher priority ECAM would cease that procedure and display the higher priority ECAM for action.

If an ECAM message disappeared either before the crew could action the procedure or while they were actioning the procedure, the *Flight Crew Training Manual* (FCTM) – Operating Philosophy – Abnormal operations and ECAM section indicated that the crew could consider that the warning or caution was no longer applicable. Application of any current procedure could be stopped.

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[106] ECAM system logic is programmed to deal with every system requiring an alert as per aircraft certification rules.

The information displayed in response to ECAM detection was colour coded to provide assistance to the crew in determining its importance. The colour codes were:

- Red – configurations or failures that required immediate action by the crew. This included specific limitations on crew action such as LAND ASAP (as soon as possible).[107]
- Amber – configurations or failures that the flight crew should be aware of, but did not require immediate action by the crew. When conditions permitted, these were required to be attended to without delay to prevent further degradation of affected systems. It also displayed specific limitations to the crew such as LAND ANSA (land at nearest suitable aerodrome).[108]
- Green – information relating to the procedure, checklist items already completed by the crew and memo items resulting from completed procedures.
- White – procedures completed by the crew, condition items and titles of procedures.
- Blue – actions to be completed in a procedure, limitations to be followed, checklist items not yet completed.
- Grey – checklists completed by the crew, and actions not yet validated by the crew.

ECAM Operation

An example ECAM procedure as displayed to the crew on the lower part of the EWD is depicted in Figure A4.

[107] Defined by Airbus as being part of specific emergency procedures, in which case the applicable ECAM procedure contained a 'LAND ASAP' message. This indicated that the crew should land as soon as possible at the nearest suitable airport at which a safe approach and landing can be performed.

[108] Defined by Airbus as part of certain abnormal procedures. In these cases, the applicable ECAM procedure contained a 'LAND ANSA' message. This indicated that the crew should consider landing at the nearest suitable airport.

Figure A4: ECAM Procedure Display (representative)

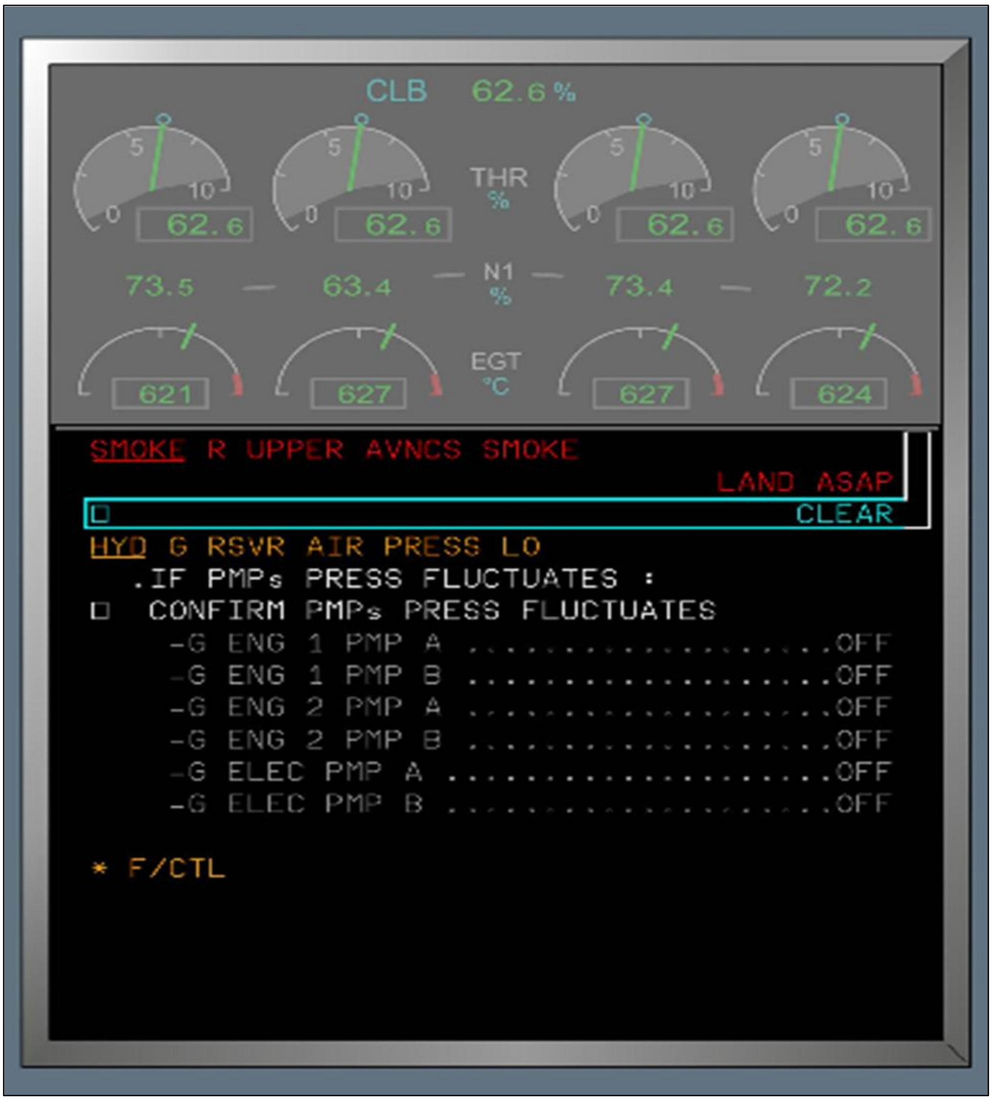

Image source: Airbus

Crews were required to work through the procedure from the top to the bottom, moving the applicable cursor as they did so and completing the required actions until the procedure was completed. The normal operating philosophy was that the pilot who was not handling the aircraft's controls managed the ECAM procedures and sought confirmation from the handling pilot as required.

For each item in a procedure, the pilot who was not handling the controls read each step in the procedure out aloud, carried out the applicable action and then marked that action as complete using the ECP. Some procedure items required the correct switch or control to be positively identified by both crew members before being activated. As a consequence, the workload for each procedure varied.

Once a crew had dealt with all displayed ECAM messages, the system displayed a STATUS (STS) screen. This screen:

- allowed the crew to review any applicable information on the impact of the ECAM procedures on the applicable aircraft systems
- alerted the crew to any procedures requiring completion before entering certain phases of flight (for example, descent or approach), and any

operational limitations imposed by the failure or degradation of aircraft systems

- presented pertinent information relating to the operation of systems that may have been affected as a result of the detected failures
- indicated a list of inoperative systems on the aircraft.

If there were significant lists of information that needed to be presented to the crew, this would be indicated by a MORE indication at the bottom of the STATUS page.

Crew recollection of the initial ECAM messages

The flight crew recalled the following systems warnings and inoperative system messages on ECAM after the failure of the No. 2 engine:

- engines No. 1 and 4 were operating in a degraded mode
- Green hydraulic system – low system pressure and low fluid level
- Yellow hydraulic system – engine No. 4 pump errors
- failure of the alternating current (AC) electrical No. 1 and 2 bus systems
- flight controls operating in alternate law
- wing slats inoperative
- flight controls – ailerons partial control only
- flight controls – reduced spoiler control
- landing gear control and indicator warnings
- multiple brake system messages
- engine anti-ice and air data sensor messages
- multiple fuel system messages, including a fuel jettison fault
- centre of gravity messages
- autothrust and autoland inoperative
- No. 1 engine generator drive disconnected
- left wing pneumatic bleed leaks
- avionics system overheat.

Manufacturer assessment of ECAM warnings

The aircraft manufacturer, Airbus, advised the ATSB that all ECAM messages that were displayed to the crew were associated with an actual aircraft condition and were therefore considered to be valid messages.

When the flight crew noticed that the YELLOW Hydraulic system was affected, yet all the damage was on the side of the aircraft in which the GREEN hydraulic system was located, they questioned the validity of the ECAM messages being presented to them. A discussion ensued, the outcome of which was that the crew agreed to assess and action the ECAM messages as they were displayed to them.

Reconstruction of ECAM sequence

The voice recording of crew actions and dialogue immediately following the engine failure was not available to the ATSB due to limitations of the 2 hour cockpit voice recorder and the extended duration of the ground operations. The available recorded voice commenced after the crew had completed all of the applicable ECAM actions and the aircraft was on approach to Changi Airport, Singapore. The ATSB used other data sources and the crew's recollection of events to recreate the ECAM timeline and associated workflow.

The time taken by the flight crew to respond to specific ECAM warnings and cautions was not directly recorded in any of the data sources on board the aircraft. However, the times and methods of the applicable system page display were recorded and the ECAM warnings and cautions were recorded in one of the onboard computer systems. The time each warning was CALCULATED, DISPLAYED and RESET[109] was also recorded in that data.

ECAM logic provides a particular priority to each specific warning, and certain messages may be inhibited during certain flight phases. There are 12 separate flight phases on the A380. During each of these flight phases, depending upon the type of procedure and its importance to the particular phase of flight, specific ECAM messages may be inhibited from display to the crew so as not to cause a distraction. When a large number of ECAM messages are generated, the priority logic ensures that the crew see the most important ones first. Less important ones may be at the end of the queue and therefore not initially visible to the crew due to the length of the preceding procedures on the ECAM display.

The FCOM indicated the annunciation of each warning to the crew in terms of whether there was an audio alert, a visual alert, a local light and/or the automatic display of the applicable system page. By correlating the applicable system page, the recorded ECAM warning and its relevant priority, the DISPLAY time and FCOM indications and the changes in aircraft systems and control positions, the most likely ECAM sequence processed by the crew was recreated.

A number of ECAM procedures were recorded on more than one occasion during the sequence as a result of the particular ECAM being marked as RESET before the triggering conditions reoccurred. Where a recording time was indicated, this corresponded to the time that the system display (SD) page displayed the specific ECAM message, or the time at which specific events took place. These times are displayed in Coordinated Universal Time (UTC). Therefore the order in which the ECAM messages are listed in the following reconstruction is the likely order in which they presented to the crew.

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109 There were three separate conditions for each ECAM system warning or caution. If the parameters for a warning or caution were met, then the ECAM recorded that the warning was CALCULATED. The applicable ECAM procedure DISPLAYED on the EWD and, if there was an applicable system page to be called up as part of the warning, this was DISPLAYED on the SD monitor. Certain flight phases could inhibit the warning. Once a crew actioned the procedure, or the conditions were no longer present, the system RESETs the warning so that it is available later in the flight if needed.

ECAM procedure timeline

The ECAM procedure timeline has been broken up into smaller segments to aid in reading and understanding of the ECAM procedures affecting the crew. The timeline is contiguous from one diagram to the next, that is, the next diagram starts immediately after the end of the previous one.

The following information is provided to enhance reader understanding of the ECAM timeline:

- Each system page is displayed via a different colour as follows:

  System page called by Warning

  System page called by Advisory

  System page called by Flight Phase

  System page called manually by flight crew

- The start time and duration each page display is shown as 02:02:10 – 02:02:28 (in this case meaning an 18 second display).
- The name of the applicable system page is displayed below the time scale.
- The applicable ECAM procedure that was most likely to have triggered the system page display, or was found from the data analysis to have occurred at a particular time, is displayed in the following convention:

  FUEL FEED TK 2 MAIN+STBY PMPS FAULT.

  An arrow indicates the message's place on the timeline.

- Specific activities are commented on by the ATSB such as – *APU STARTED*

The timeline starts at 02:00:00 and ends after the aircraft stopped on the runway at 03:46:45.

ECAM following the engine failure

The first ECAM message was displayed at 02:01:08 and confirmed by the flight crew as ENG 2 TURBINE OVHT (Figure A5). There was no system page displayed with this message. Therefore the applicable system page at that time was the CRUISE[110] page. Many other ECAM warnings continued to be generated following the engine failure. At 02:01:30, nearly all the warnings were recorded as being RESET. The warnings were recorded as reappearing some 27 seconds later and were again displayed to the crew.

The data showed that at 02:09:56 the crew manually selected the ENGINE system page. The flight crew indicated that at this time, they could see that one of the upcoming ECAM warnings related to the Yellow hydraulic system. Knowing that the Yellow system was powered by the still-running engines No. 3 and 4, the crew were questioning the effect of any aircraft damage on the functionality of the ECAM itself. The crew reported that, to assist their decision making as to whether the ECAM was functioning correctly, they selected the ENGINE and

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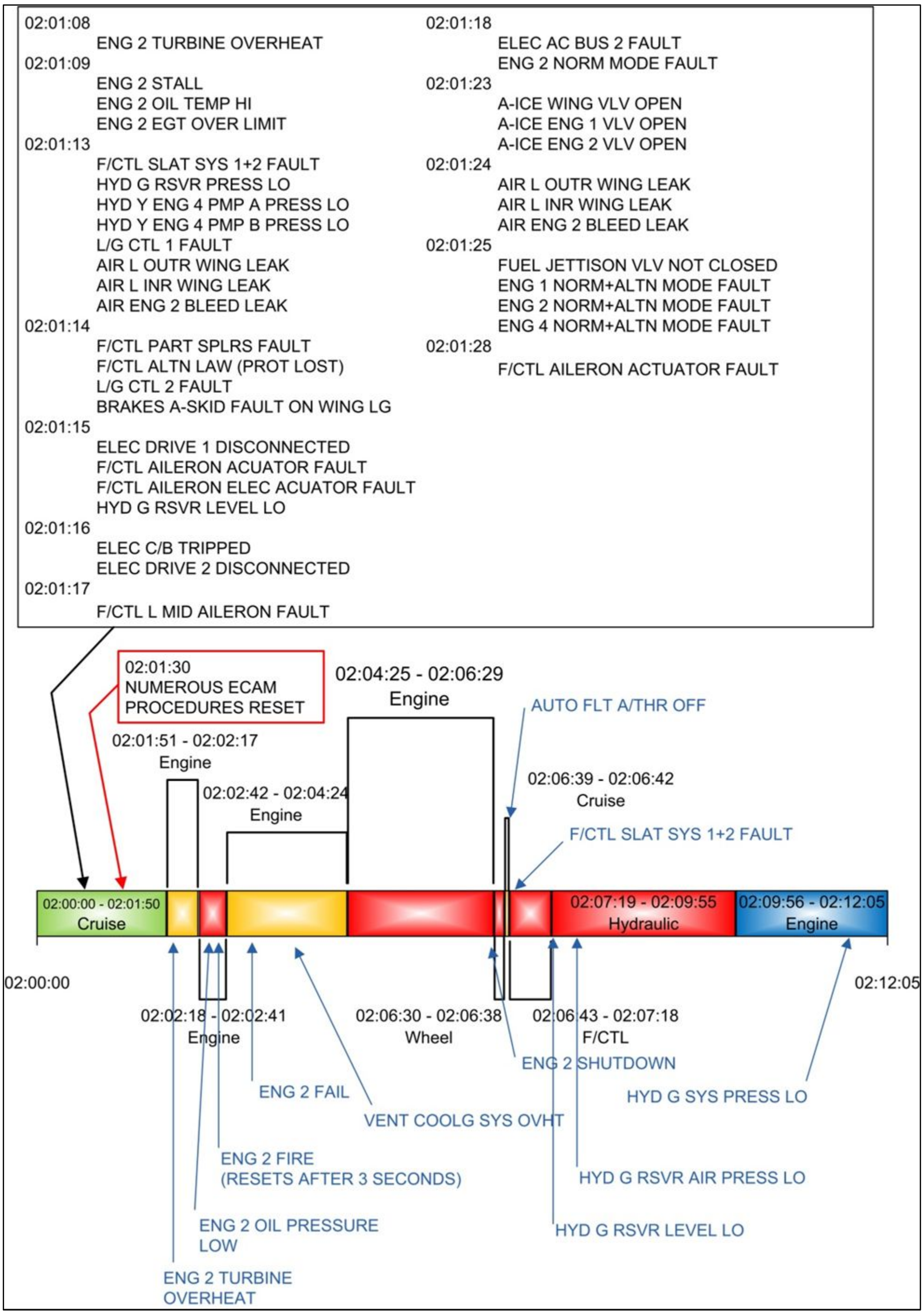
[110] The CRUISE system page is displayed automatically when the aircraft passes 1,500 ft above the airport of departure or on reaching the thrust reduction altitude, whichever is higher.

HYDRAULIC pages to identify any damage to engines No. 3 and 4, or to the Yellow hydraulic system.

Figure A5: ECAM timeline 02:00:00 to 02:12:05

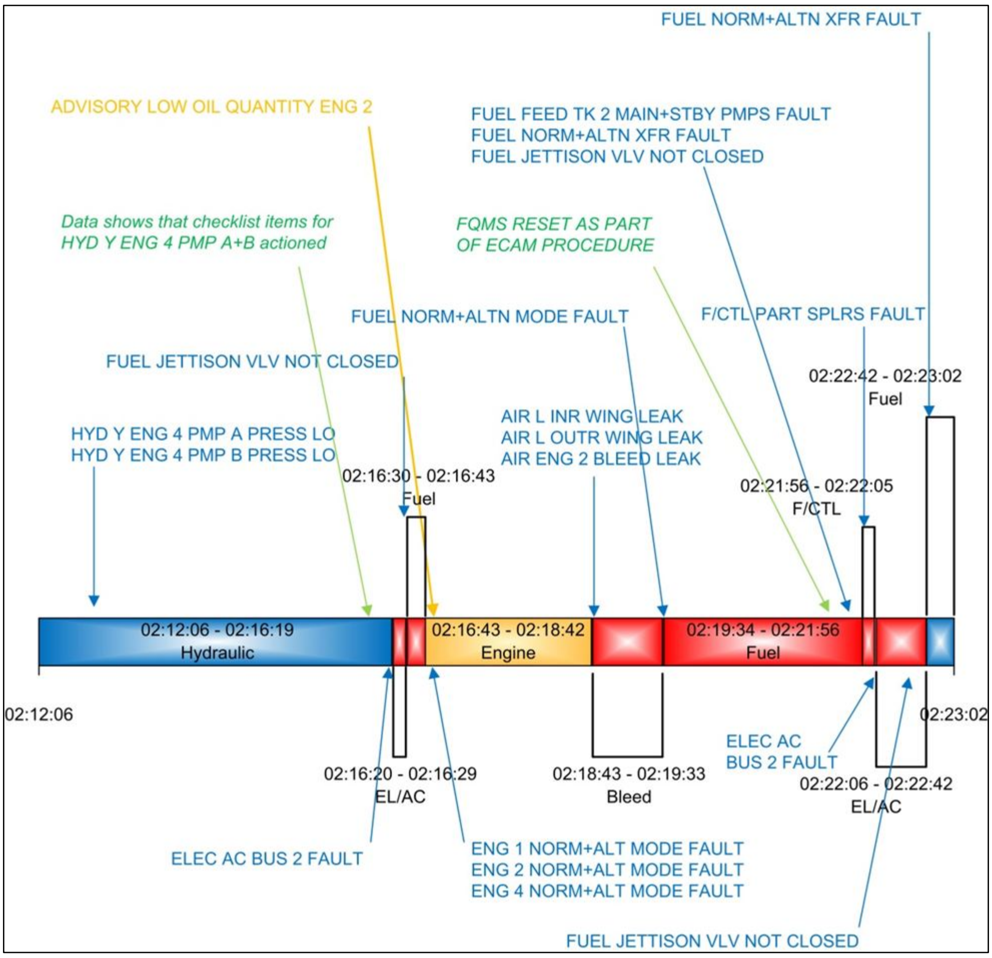


The crew manually selected the HYDRAULIC system page at 02:12:06, again to assist their decision making in relation to the displayed ECAM warnings. They discussed the information available to them, and considered the information from the system pages. The crew recalled that this discussion took some time, after which they decided that it was unlikely that ECAM was malfunctioning and that they should action the ECAM procedures as displayed. Recorded data showed that the crew actioned the ECAM procedure for the Yellow hydraulic system and the applicable position of the HYD Y PMP A+B action is displayed accordingly in Figure A6.

ECAM procedures after the aircraft entered the holding pattern

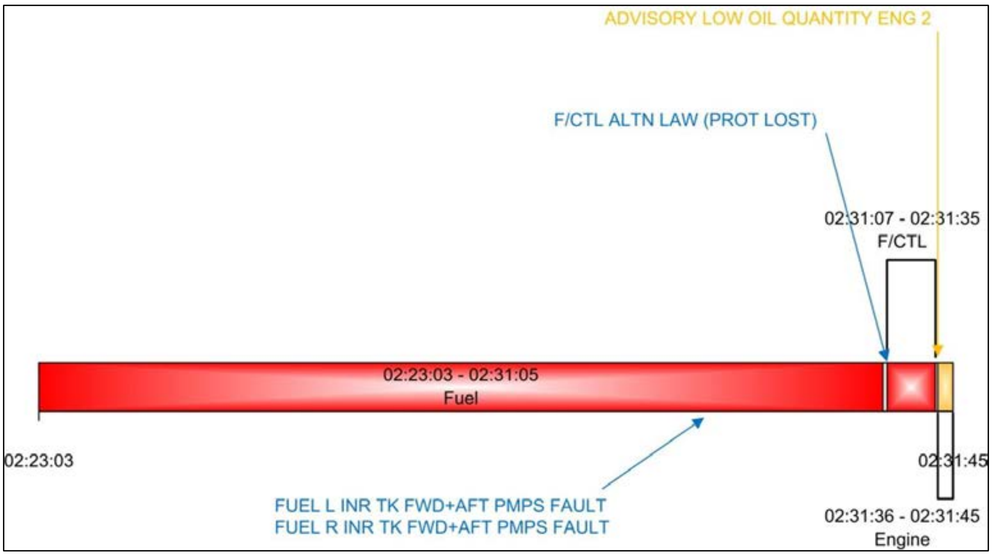
The aircraft entered the holding pattern at 02:18:05. The crew reported that a number of already actioned ECAM procedures, specifically related to the fuel system, reappeared during the ECAM sequence after that time. Airbus indicated that one of these procedures was to reset the fuel quantity management system (FQMS) and as part of that reset, any applicable ECAM procedure already completed would be recalled. The FQMS reset time is displayed in Figure A6.

Figure A6: ECAM timeline 02:12:06 to 02:23:02



The crew reported that whenever a fuel-related ECAM displayed, and due to the extensive damage to the fuel system and number of problems displayed on the associated FUEL system page, they took considerable time understanding the ECAM procedure and the action in response. The crew took care to fully understand each system display before following a procedure on a system that was displaying significant damage. Figure A7 records the FUEL system page display for 8 minutes and 2 seconds and accords with the flight crew's recollection.

Figure A7: ECAM timeline 02:23:03 to 02:31:45



The first officer recalled that, during the time that the ECAM messages were being processed, there was a recurring ADVISORY LOW OIL QUANTITY ENG 2 message. This advisory can be seen in a number of places throughout the timeline (Figure A7 and Figure A8), with two occurrences shown in Figure A8 and another in Figure A9.

Figure A8: ECAM timeline 02:31:46 to 02:34:45

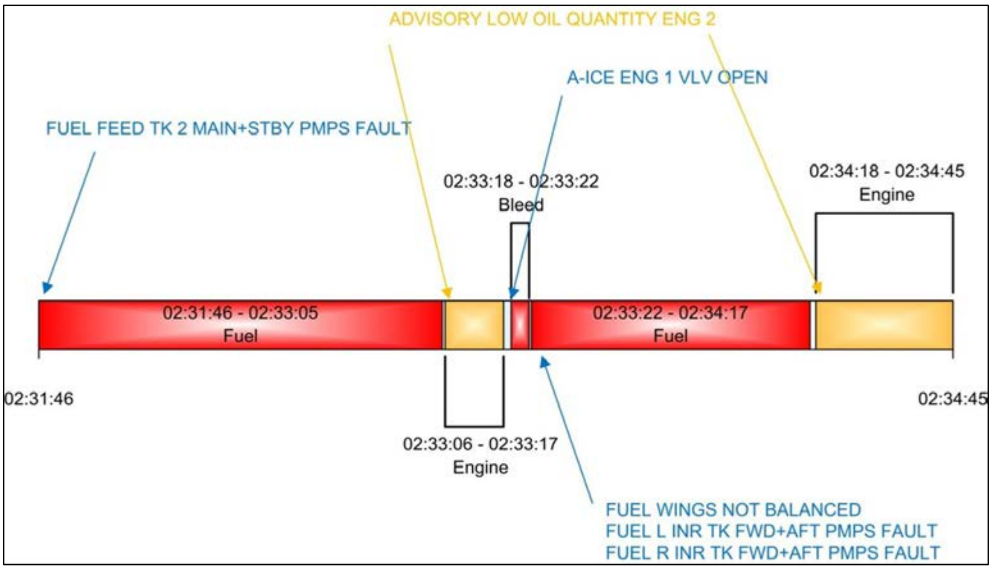
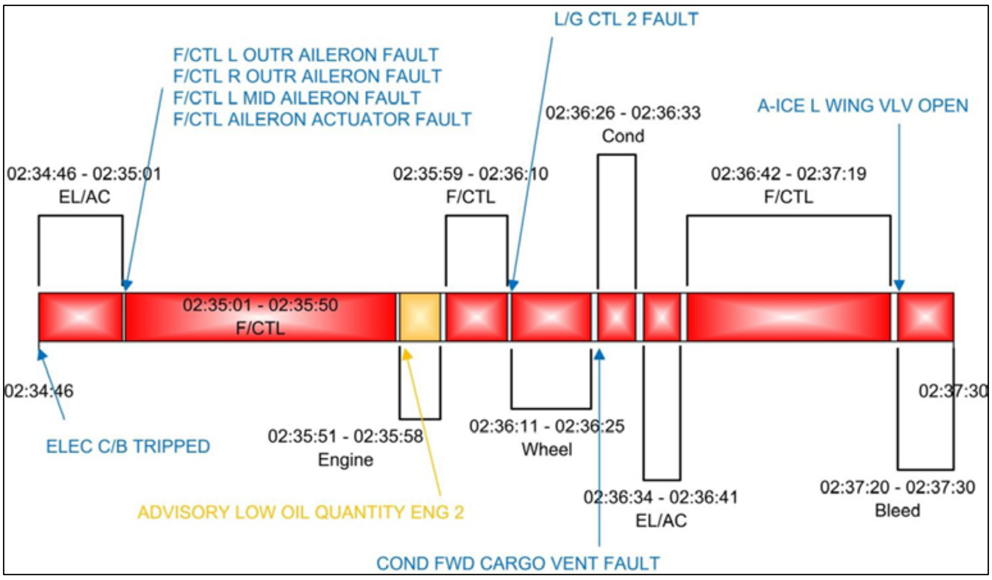
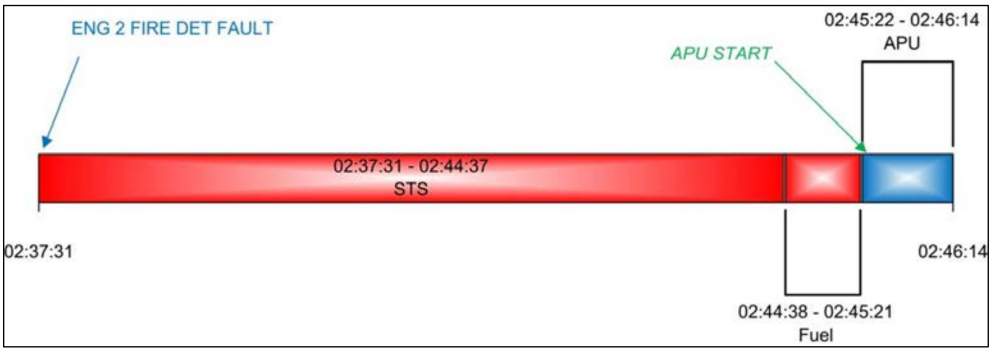


Figure A9: ECAM timeline 02:34:46 to 02:37:30



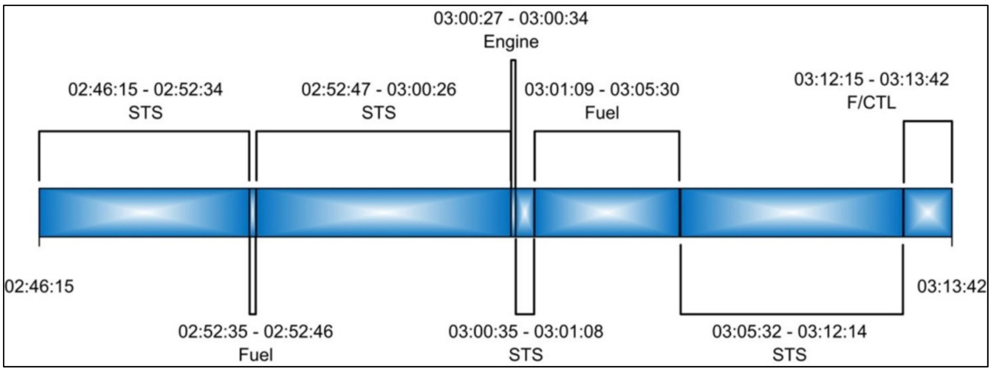
At 02:37:31, the crew actions in respect of the list of ECAM messages were complete and the STATUS (STS) page was displayed to the crew. Over the next 7 minutes, the crew reviewed the aircraft systems and associated procedures and limitations via that page. Subsequently the crew decided to start the APU in an attempt to alleviate some of the electrical system problems. The crew manually selected the APU page and started the unit at 02:45:22 (Figure A10).

Figure A10: ECAM timeline 02:37:31 to 02:46:14



Once the APU was running, the system display pages changed to those being manually selected, rather than being automatically displayed as part of the ECAM function. That was consistent with the crew progressing through a number of different systems and their recollection of seeking to understand what damage had occurred, and what systems functionality remained. These pages are displayed in Figure A11.

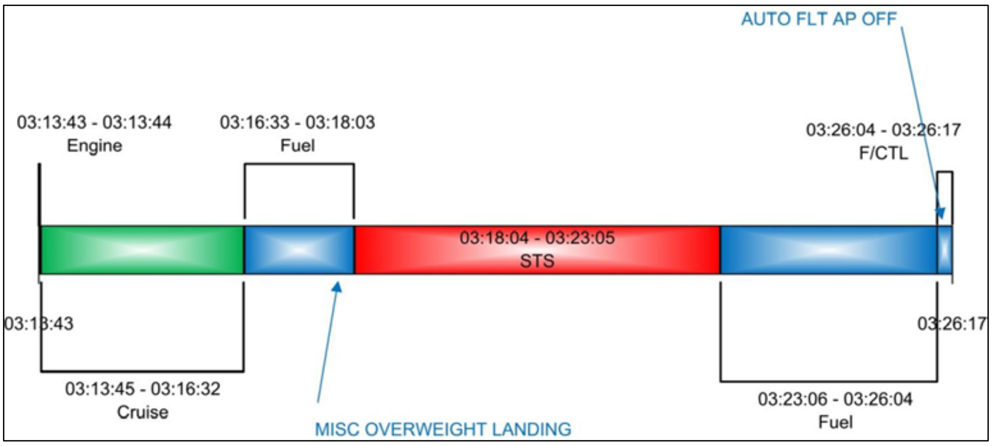
Figure A11: ECAM timeline 02:46:15 to 03:13:42



At 03:13:43, the CRUISE page was once again selected. All of the required ECAM actions were completed by that time and the crew later reported that they had begun to understand the effect of the engine failure on the aircraft and its systems.

The flight crew commenced preparing the aircraft for landing at Changi Airport and accessed the Landing Distance Performance Application to determine the performance in the case of an overweight landing. The supporting ECAM procedure for OVERWEIGHT LDG is displayed in Figure A12. The de-selection of autopilot is also shown, consistent with the conduct by the crew of a manual control check of the aircraft.[111]

Figure A12: ECAM timeline 03:13:43 to 03:26:17



ECAM procedures after the aircraft left the holding pattern

The aircraft left the holding pattern about 03:28:00 and was vectored to the final approach course. Figure A13 shows the system page display as the crew manually selected various system displays to further enhance their understanding of the condition of the aircraft and its systems. It also has two occurrences where the page reverted to the CRUISE page and where the crew selected the flight control page. The crew reported that the selections of the flight control page were to further assess the aircraft's controllability during reconfiguration.

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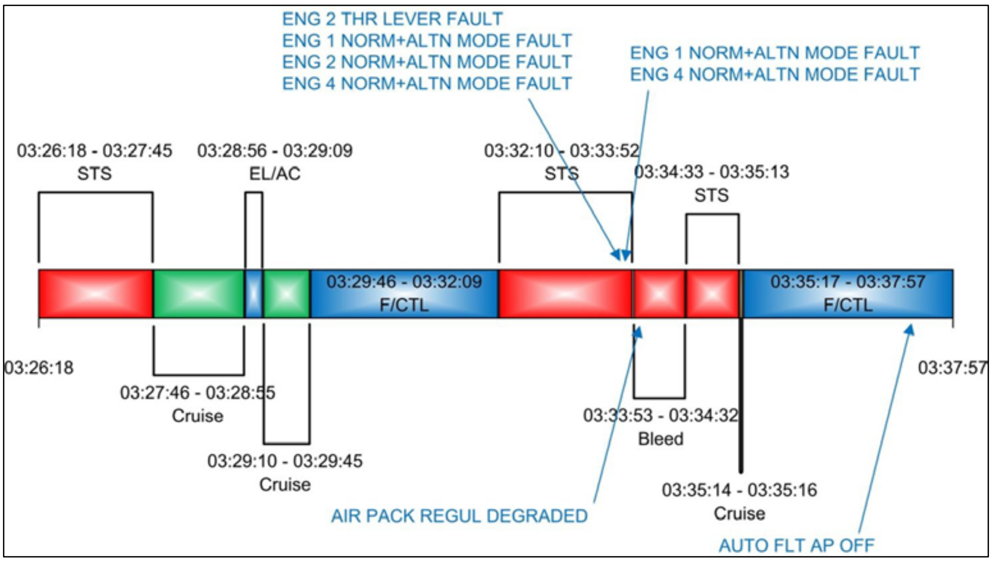
111 The actual flight controls (ailerons, spoilers, elevators and rudders) are not directly viewable from the cockpit. The only way that a crew can assess control functionality is to select the applicable F/CTL system page and monitor the displayed deflection of each control as they manoeuvre their aircraft.

Figure A13: ECAM timeline 03:26:18 to 03:37:57



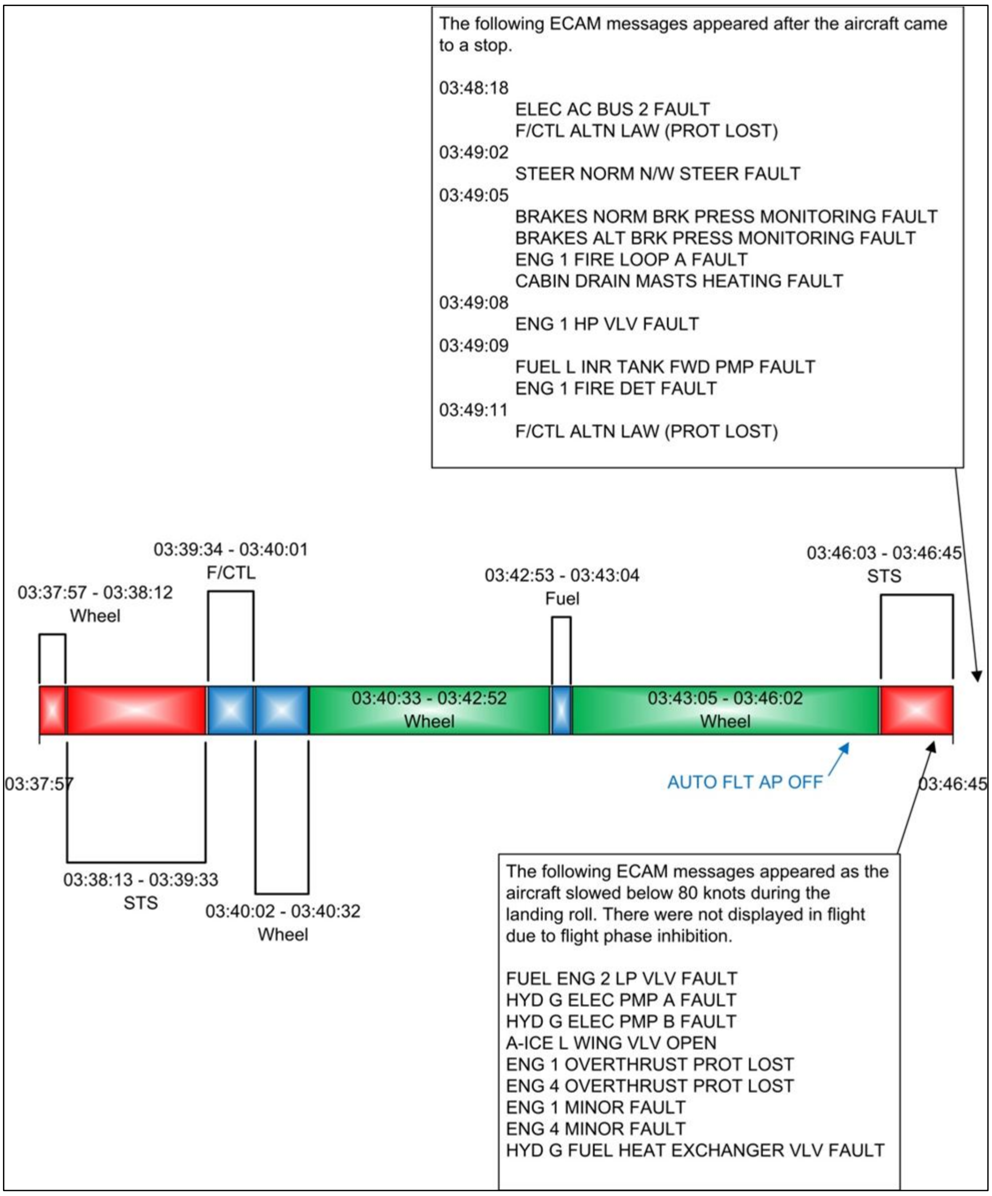
ECAM procedures during the final approach and landing

Figure A14 shows the aircraft during the final approach to Changi Airport and includes the manual extension of the landing gear. It also shows that the landing gear was down and locked at about 03:37:57 as indicated by the automatic display of the WHEEL page.[112]

After the aircraft landed and the speed passed below 80 kt, a number of ECAM messages that were previously inhibited by the flight phase were displayed to the crew. Once the aircraft came to a stop and the crew had, to the extent possible shut down the aircraft, additional ECAM messages that would normally have been displayed at that time remained inhibited due to ongoing phase of flight parameters. These parameters normally ceased 5 minutes after the final engine was shut down. In this instance, the continuing operation of the No. 1 engine meant that the aircraft had not yet reached the final flight phase.

[112] The WHEEL system page is called automatically when the aircraft landing gear is lowered on approach to land.

Figure A14: ECAM timeline 03:37:57 to 03:46:45



APPENDIX B: DETAILED DAMAGE DESCRIPTION

Introduction

The No. 2 engine sustained an uncontained failure when the intermediate pressure (IP) turbine disc burst into three main fragments, resulting in a complete circumferential breach of the engine case. Once the engine case was breached, high energy disc segments, associated turbine blades and other engine debris were released and impacted the airframe.

A large fragment measuring approximately 40% of the total disc circumference was recovered from Batam Island, Indonesia.

It is likely that at least one of the three main disc fragments broke up into smaller pieces during its initial ballistic trajectory through the engine case, before impacting the left wing structure. A smaller disc fragment and associated engine debris penetrated through the wing-to-body fairing and impacted airframe structure. Smaller, high-energy disc fragments and engine debris also impacted the airframe, resulting in significant damage (Figure B1).

Figure B1: Trajectory of the major disc segments. Two main fragments impacted the wing. A smaller, high energy fragment followed a different ballistic trajectory and penetrated the fuselage belly fairing structure.

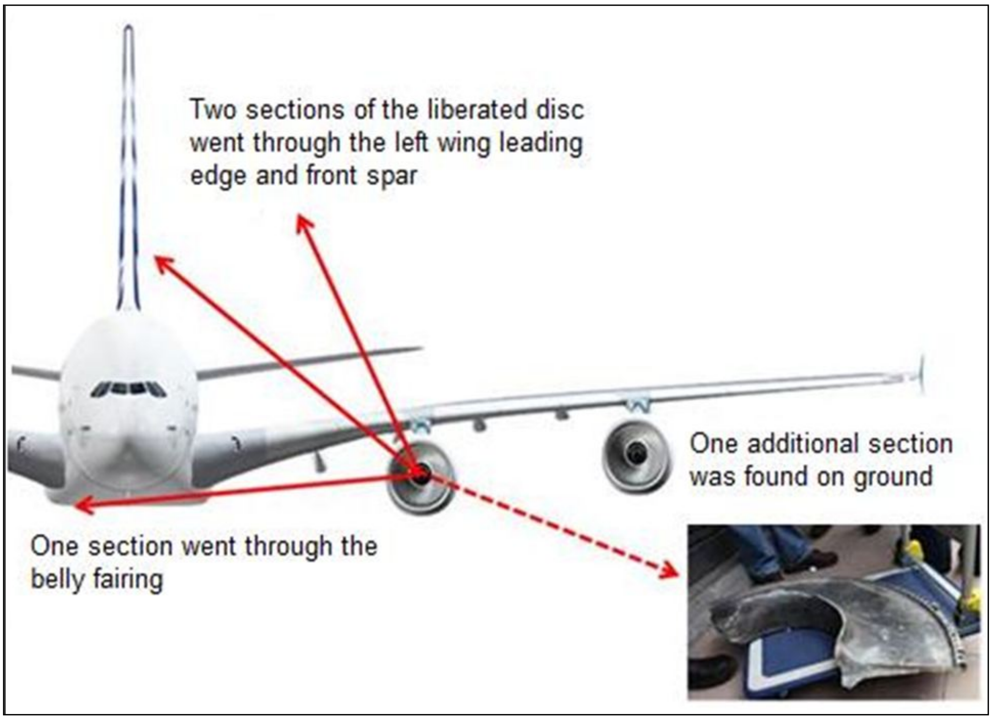


Image source: Airbus

Damage to wiring looms located in the left wing and the fuselage belly fairing resulted in the loss of a number of redundant systems that used segregated control wiring. Those systems included:

- loss of the fuel isolation valves (LPSOV) for the No. 1 and No. 2 engine

- loss of the fire protection system for the No. 1 engine
- all means of shutting down the No. 1 engine
- loss of function of the Green hydraulic system
- loss of fuel transfer system.

This Appendix describes the structural airframe and system damage/degradation following the failure.

Airframe structural damage

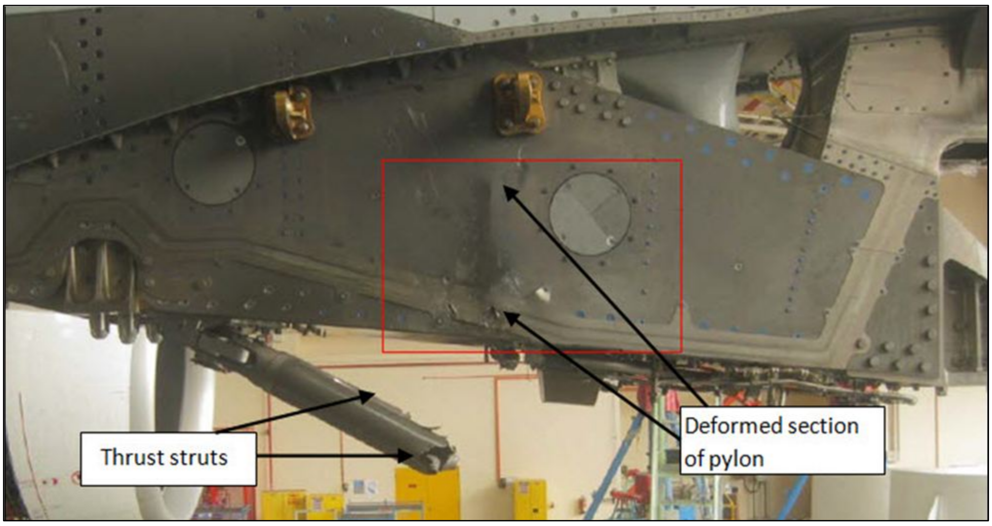
The uncontained engine failure caused significant damage to the following airframe structures:

- No. 2 engine pylon
- No. 2 engine thrust links
- left wing box structure, including the:
  - front spar
  - upper wing skin
  - lower wing skin
  - wing ribs
- fuselage butt-straps number three and four.

No. 2 engine pylon

The engine pylon attached the engine to the wing. The No. 2 engine pylon sustained impact damage to its sidewalls including bending, scratches and small gouges. A section of the pylon was deformed along its length and width (Figure B2).

Figure B2: Inboard side of the No. 2 pylon highlighting the deformation and the severed thrust struts (links)



No. 2 engine thrust link assembly

The thrust link assembly transfers engine thrust load to the wing through the pylon and is comprised of a cross beam and two thrust struts (Figure B3). Each thrust strut is designed to withstand the limit loads of the engine thrust. Both thrust struts were severed as a result of impacts from the liberated engine debris (Figure B4).

Figure B3: Engine thrust link assembly location in relation to the engine

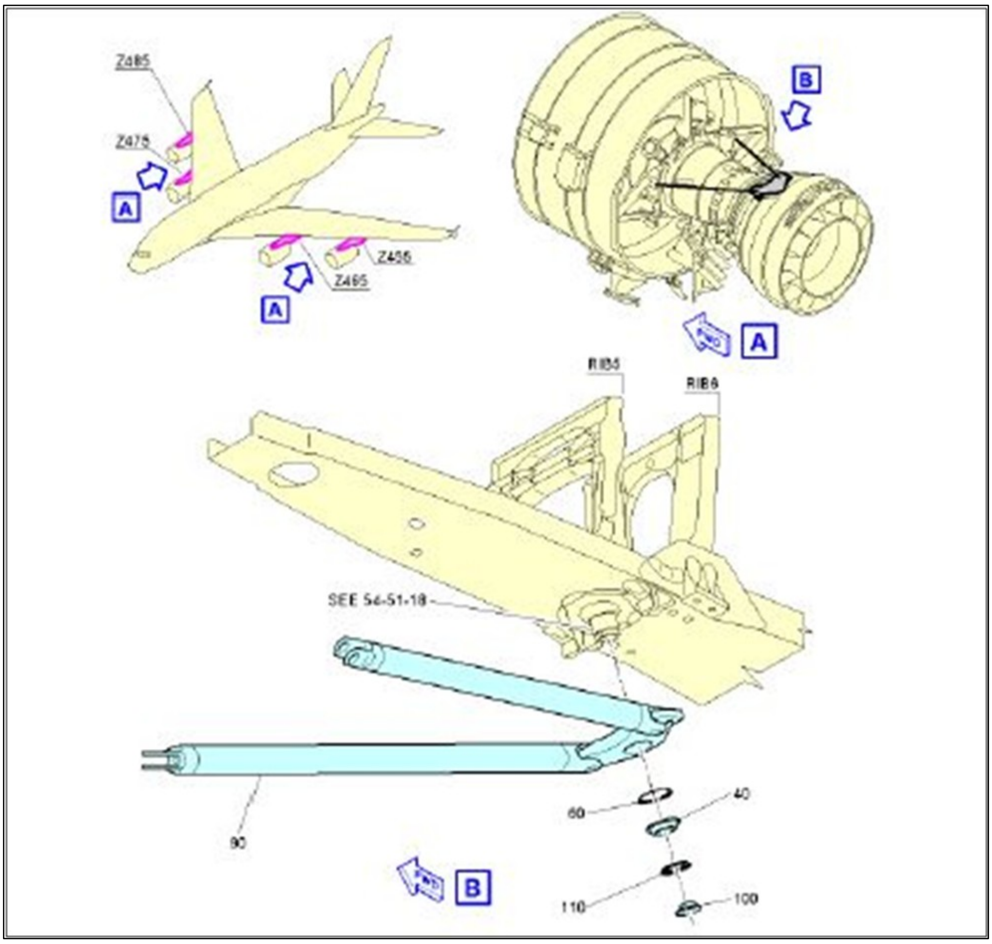
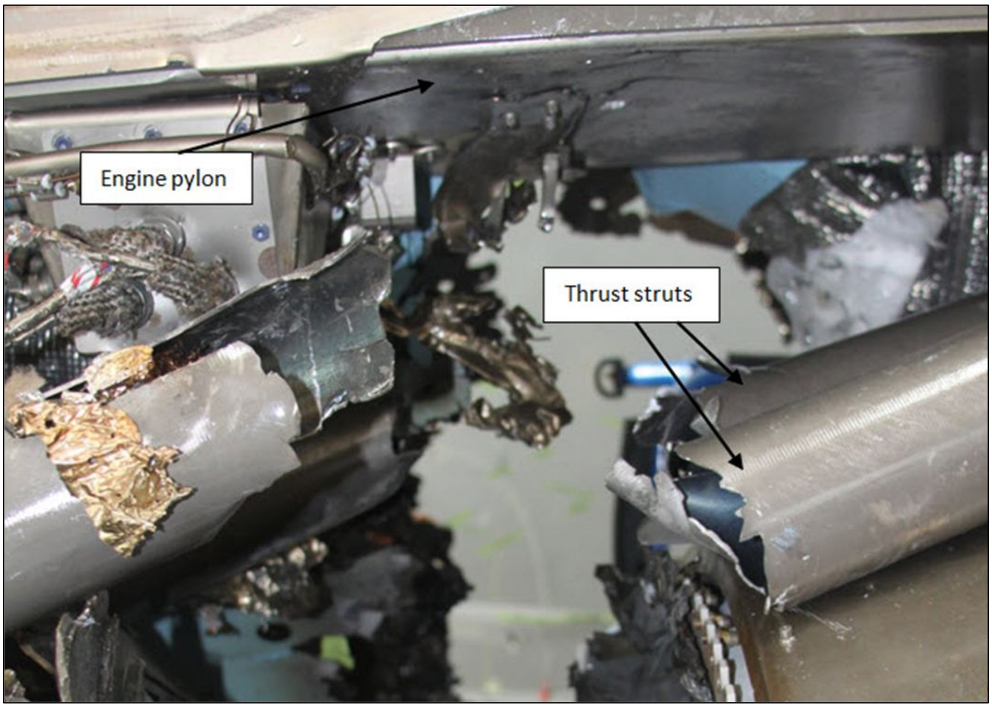


Image source: Airbus

Figure B4: Fractured tubular thrust struts



Left wing

The aircraft's left wing structure (Figure B5) comprised of the following main components:

- the centre wing carry through structure, which forms part of the fuselage and gives attachment points for the wings
- the left and right wing structures, each of which includes the wing box, wingtip, leading edge and leading edge devices and the trailing edge devices.

The critical structural component of the wing is the wing box, which is comprised of the wing spars, upper and lower wing skins and ribs.

Figure B5: Left wing general layout

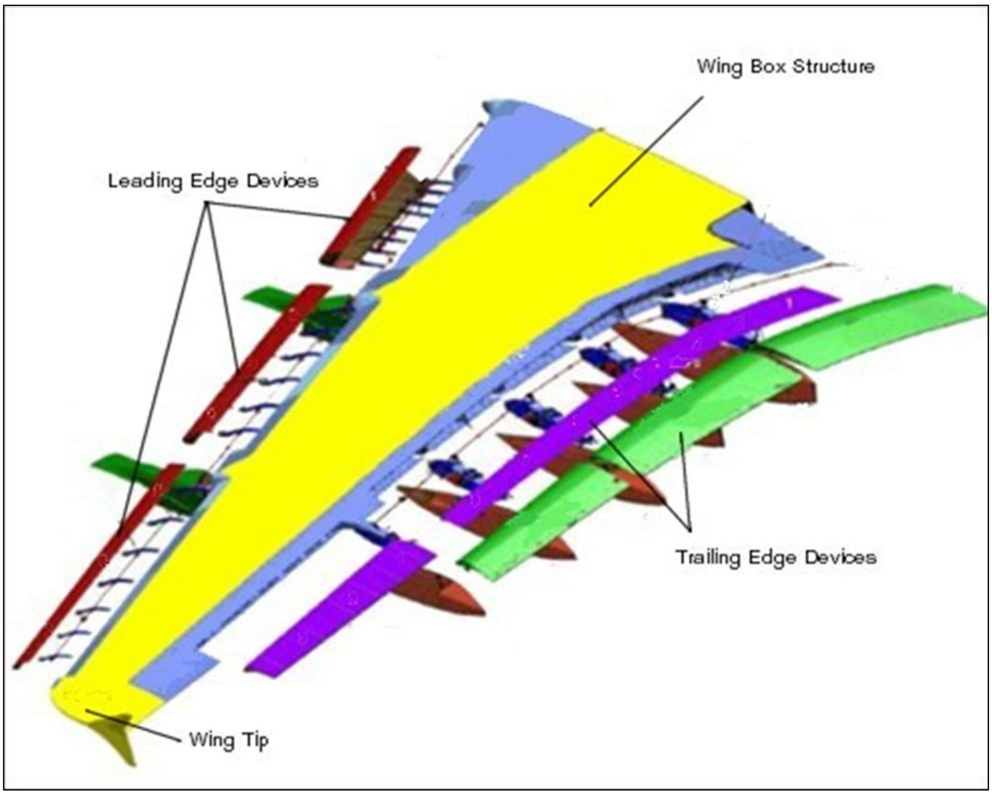


Image source: Airbus

Wing damage

A section of the fractured turbine disc penetrated through the engine case, before continuing through the wing's non-structural composite leading edge and wing spar and then exiting through the upper wing skin (wing box structure) in a trajectory that projected over the fuselage. The areas of significant damage to the wing are shown in Figure B6.

Figure B6: Left wing schematic showing the locations of significant wing skin damage and distribution of debris path

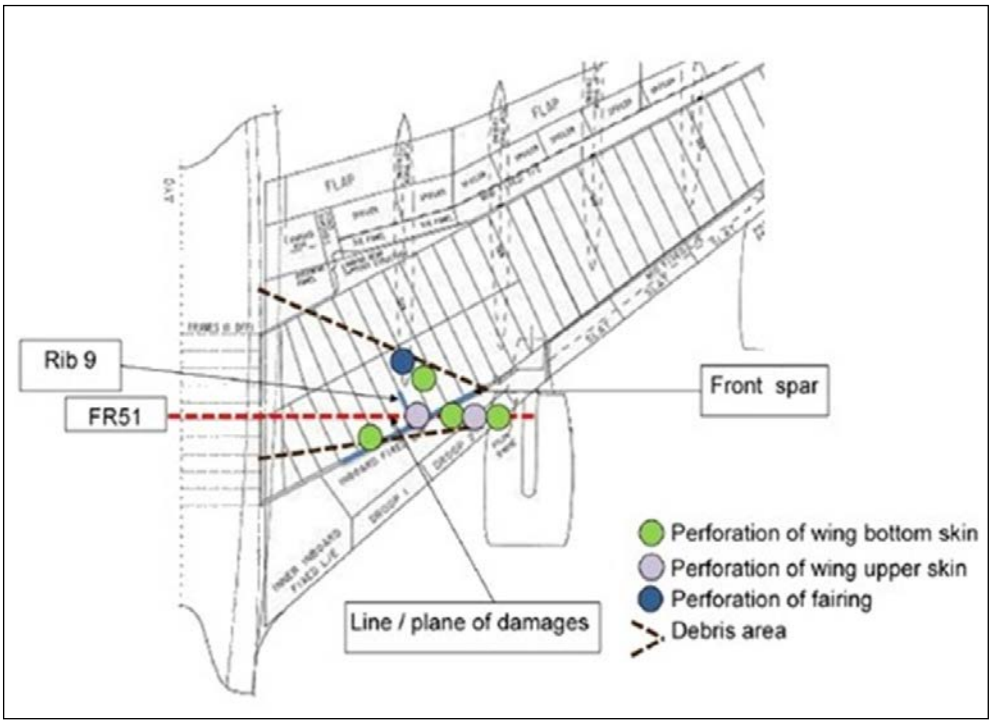


Image modified from an Airbus supplied image

Left wing upper and lower skin

The upper and lower wing skins were each comprised of five separate panels (Figure B7). The multiple wing panel design provided a degree of protection against crack propagation across the wing skin.

Figure B7: Upper wing skin panels

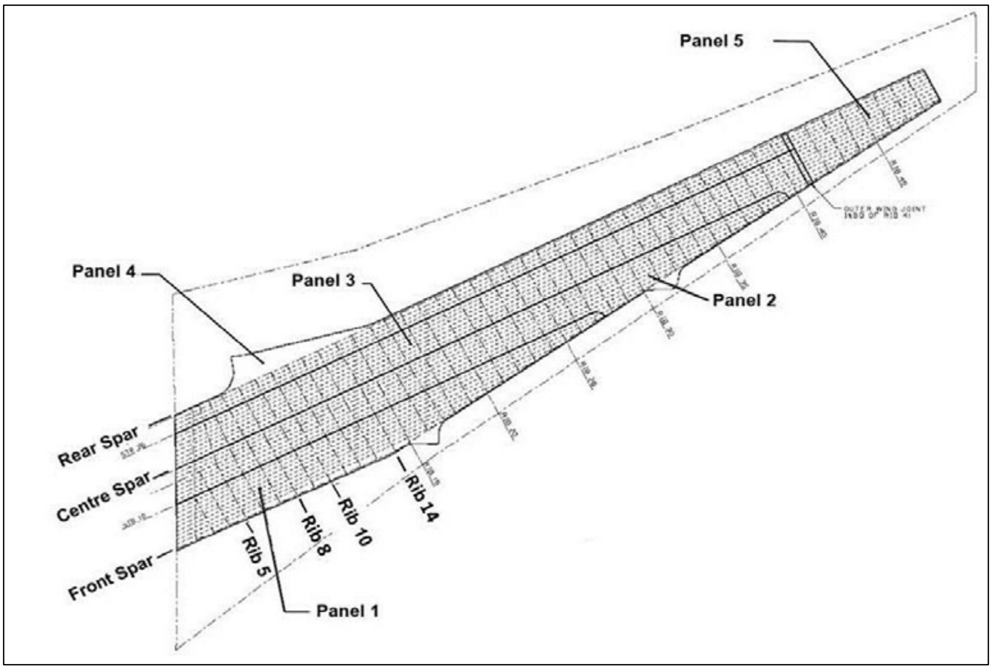


Image source: Airbus

Left wing lower wing skin damage

Lower wing skin (Figure B8) comprised of panels 1, 2, 3 and 4 (Figure B9). Impact damage was evident in 35 locations, including 2 perforations through to feed tank 2 (Figure B10 to Figure B13).

Figure B8: Lower wing structure highlighted in orange

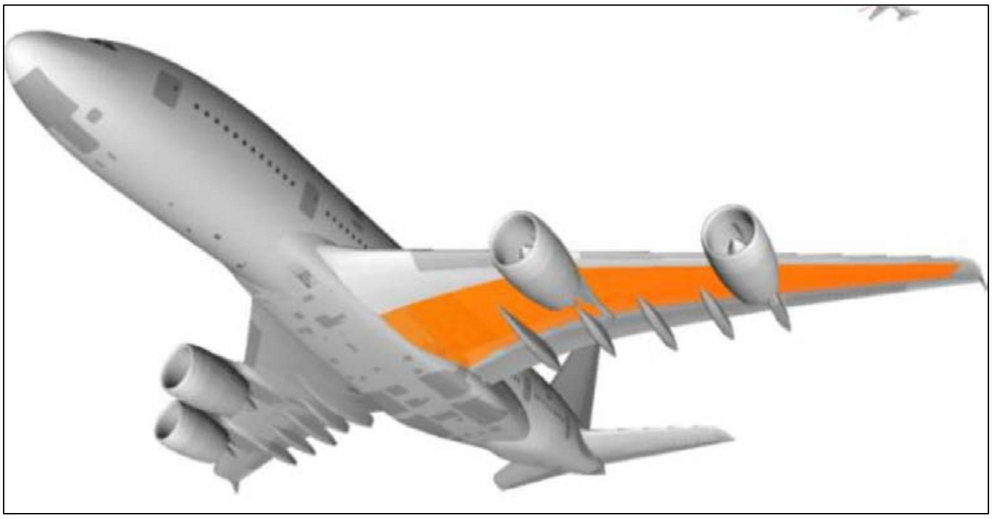


Image source: Airbus

Figure B9: lower wing skin panels

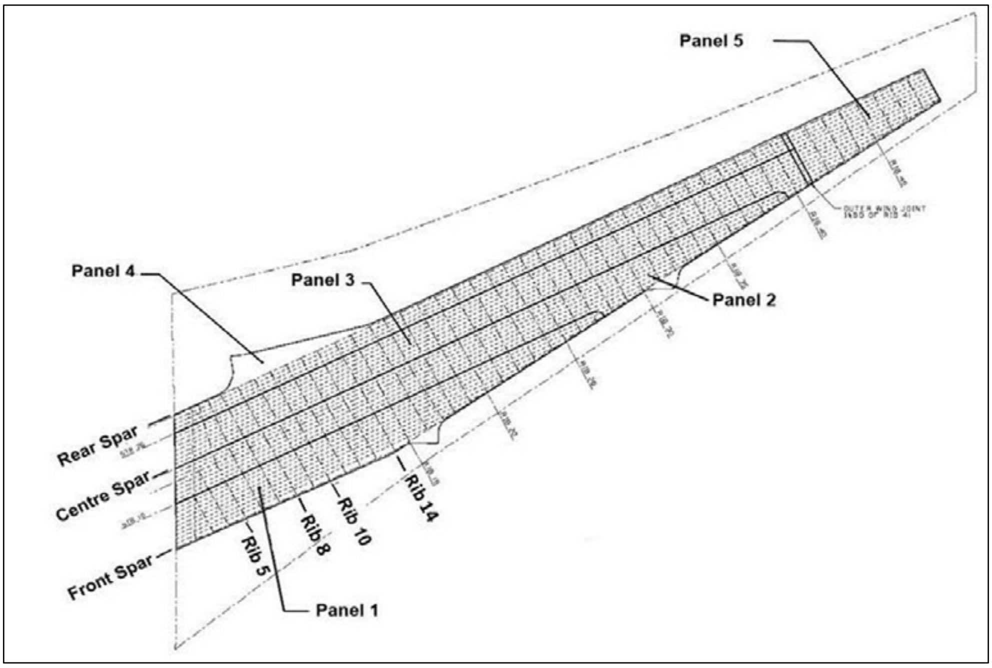


Image source: Airbus

Damage to the left wing lower skins included two puncture holes and impact damage including scores, dents and gouges measuring from a few millimetres to about 200 mm in length (Table B1). Most of the damage was concentrated between ribs No. 8 to Rib No. 14, with additional damage evident further outboard. A significant amount of fuel leaked from feed tank No. 2 as a result of the puncture holes.

Table B1: Wing lower skin damage

Rib location	Stringer number	Damage type	Damage length mm	Damage width mm	Damage depth mm
4	8-9	Score damage	28	14	0.75
4-5	5-6	Score damage	12.5	7.4	0.32
6-7	7-8	Score damage			
6-7	9	Dent & score	58	14	3
7	5-6	Score damage	35	7	0.65
7-8	5-6	Dent & score	75/35	25/25	6/0.6
7-8	6	Dent & score	13	5	1.5
7	FS	Hole & dent	220	90	Hole
6-7	FS	Dent & score	18	12	2.2
7-8	FS	Dent & score	45	20	2
7-8	3	Dent & score	30	15	1.3
8	3-4	Dent & score	70	18	5
8-9	5-6	Dent & score	23	10	2.2
8-9	1	Dent & score	10	6	1.2
9-10	1	Dent & score	22	7	1.2
10-11	6	Dents & scores	45	10	1
10-11	1	Scores	55	20	6
10-11	6	Hole, several dents and scores	140	110	Hole
10-11	4	Several dents and scores	80	40	0.5
10-11	FS	Hole and scores	60	60	Hole
12-13	FS	Dent & score	40	30	4.9
12-13	4-5	Score damage	34	6	1.5
13-14	3-4	Score damage	23	10	0.8

Figure B10: Penetration damage to the lower wing skin and droop nose flap from major IP turbine disc fragments and other engine debris that were ejected after the burst

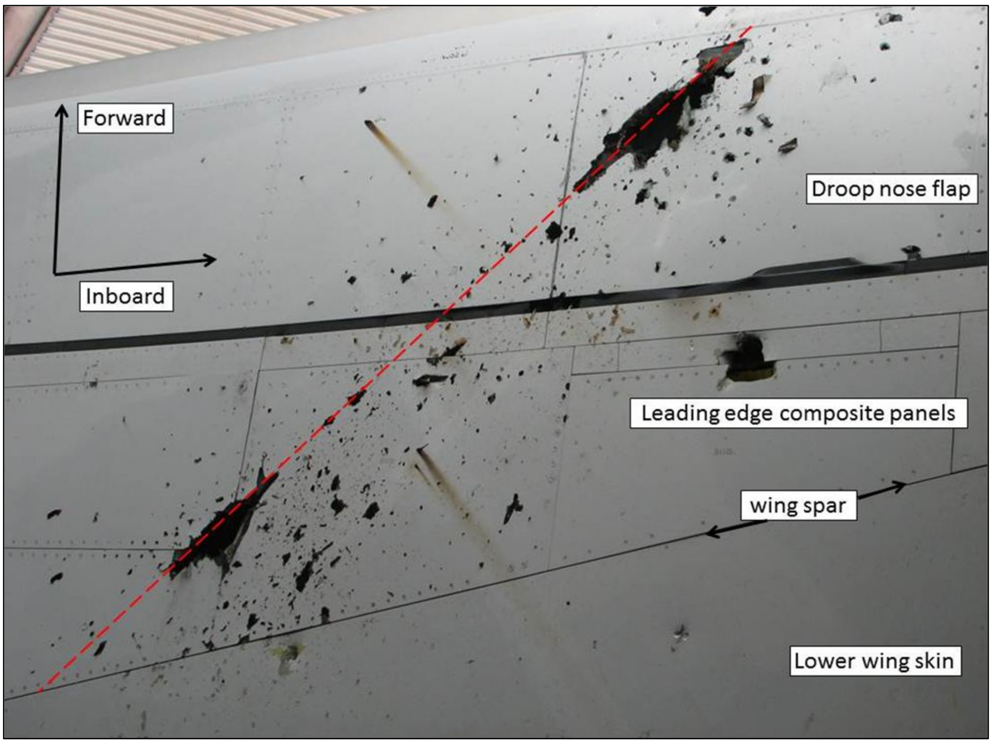


Figure B11: Lower wing skin perforations (circled) leading to fuel leakage from the No. 2 feeder tank

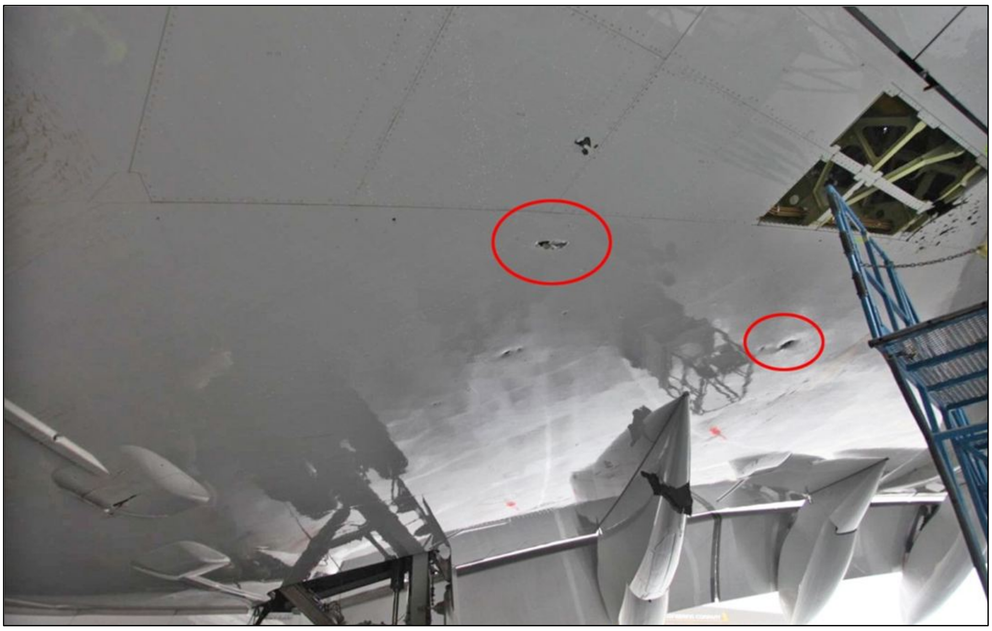


Figure B12: Puncture damage to the left-lower wing skin looking forward from inside the wing

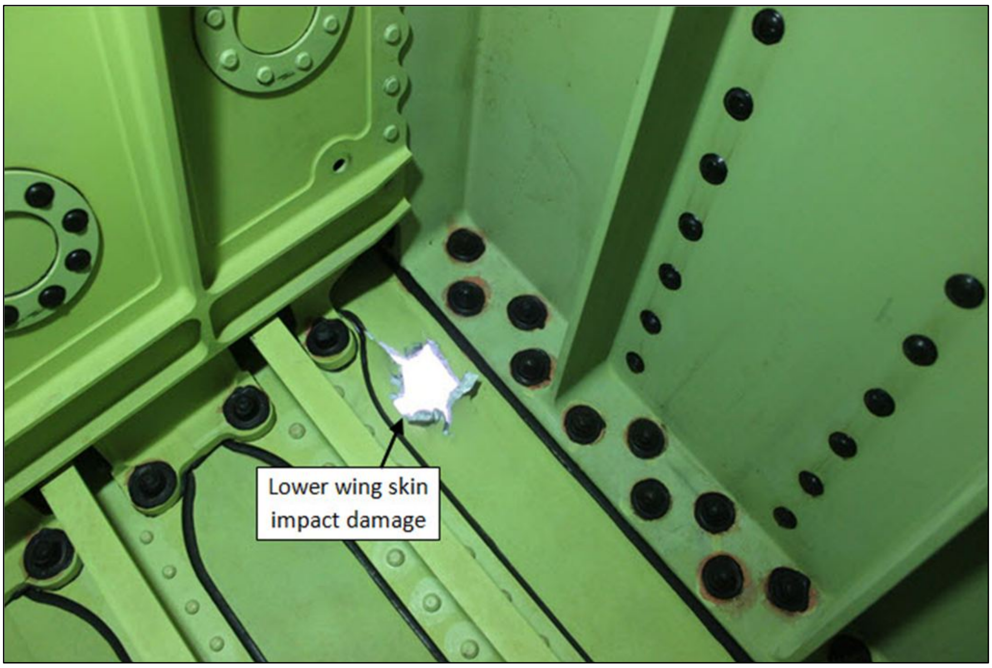
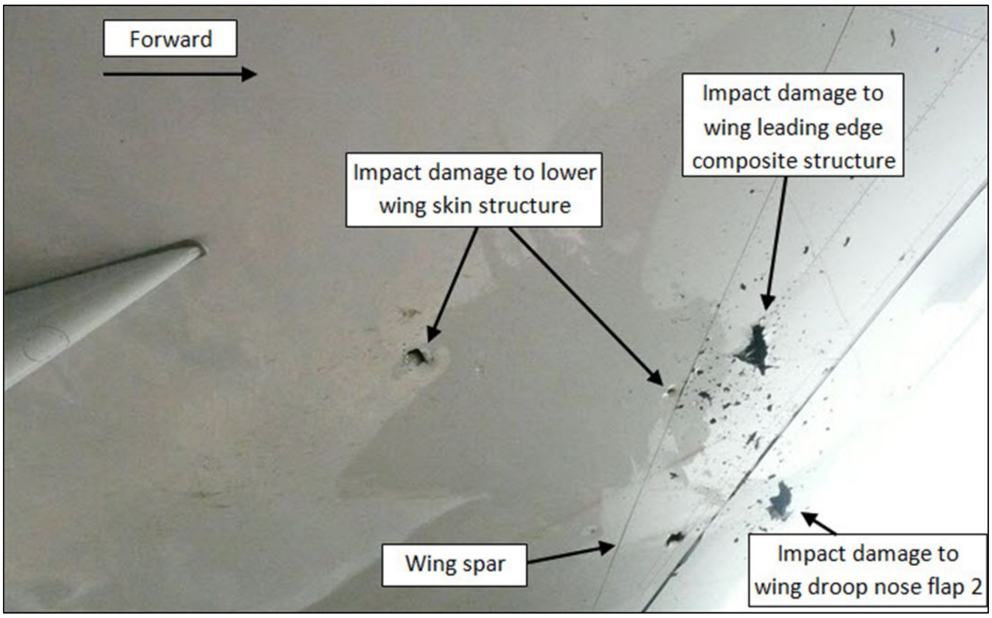


Figure B13: Impact damage to the lower wing skin



Upper wing structure damage

The upper wing skin (Figure B14) had significant damage from a disc fragment that had passed through the engine case, front spar and exited through wing upper skin panel No. 1, between Ribs 9 and 10 and stringer 2 and the front spar (Table B15 and B16).

The exit damage hole measured approximately 450 mm by 100 mm, and was located 100 mm rearward of the front spar between Rib 9 and Rib 10. Approximately 166 cm^2 of upper wing skin material was liberated.

A chordwise crack about 300mm length also emanated from the forward point of the damage.

Figure B14: Upper wing skin structural panels highlighted in orange

Image source: Airbus

Figure B15: Upper wing damage showing structural and non-structural damage

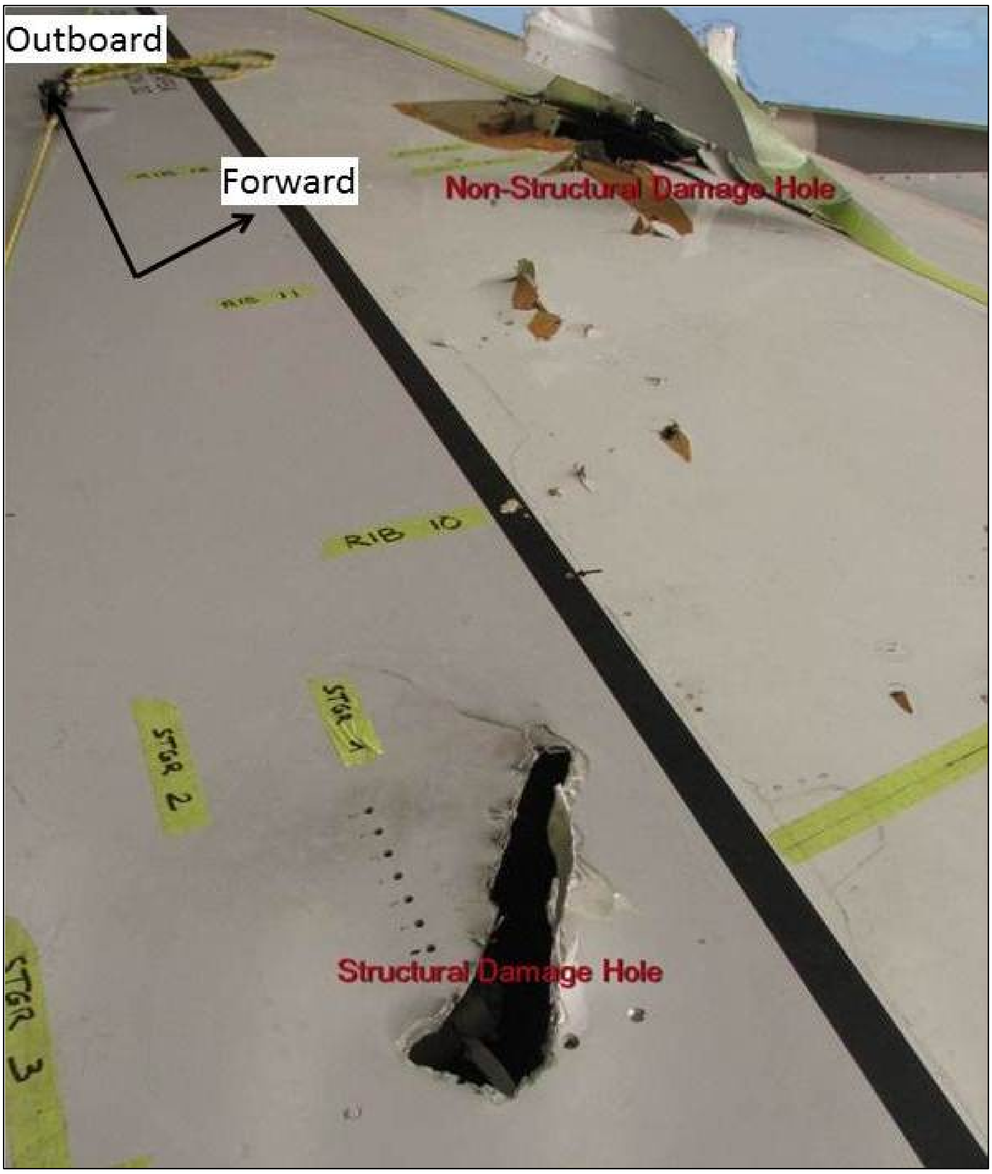
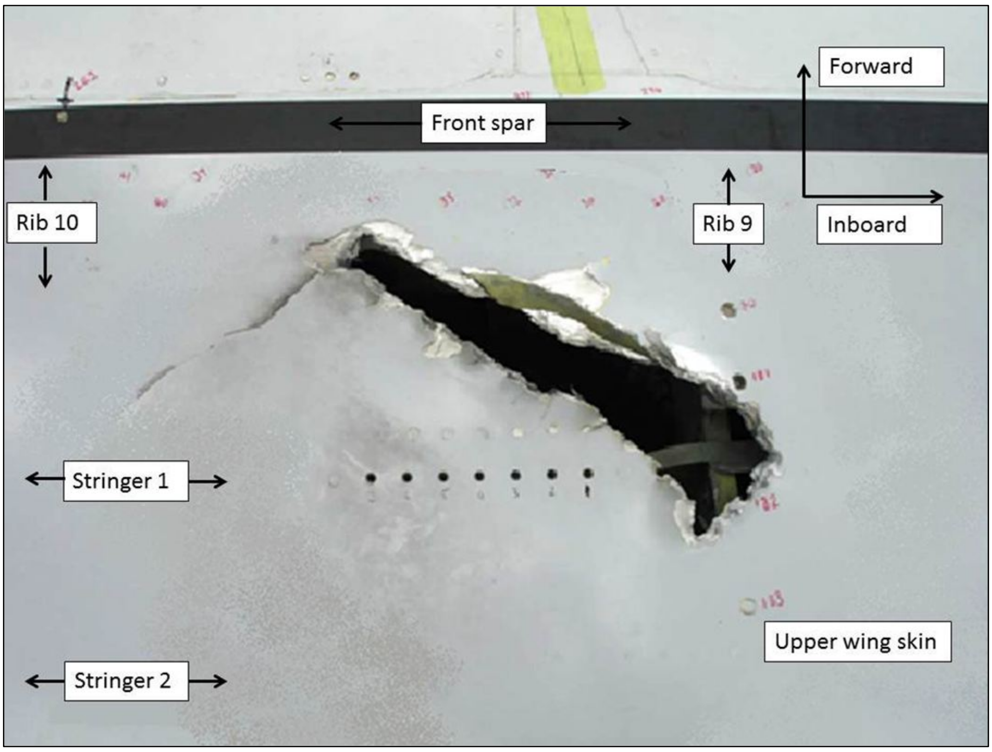


Figure B16: Damage to the upper wing skin from a significant disc fragment, panel No. 1 between Ribs 9 and 10



Left wing front spar

The front spar is comprised of five segments and formed part of the box structure and the integral wing fuel tank. Joint plates connected these segments to make one structure. A feature of the spar's construction was the closely pitched vertical and horizontal stiffeners. In the event of damage, these stiffeners were designed to act as shear panel boundaries by providing edge support and to limit crack propagation and the subsequent loss of structural integrity of the remaining intact panels.

The front spar was punctured in four areas between the No. 2 engine and the fuselage. It was evident from streaking of the anti-corrosion coating that fuel had escaped through the damaged structure (Figure B17).

The largest hole had an area of about 35 cm x 35 cm due to impact by a section of the liberated turbine disc (Figure B17). There were four visible cracks emanating from that hole. Vertical and horizontal stiffeners on the aft surface of the spar were also damaged. There were three other smaller holes of about 35 cm^2, 10 cm^2 and 7 cm^2 (Figure B18 and Table B2).

Figure B17: Impact damage from penetration of a disc fragment through the front spar of the left wing. The fragment continued in its trajectory to exit through the upper wing skin

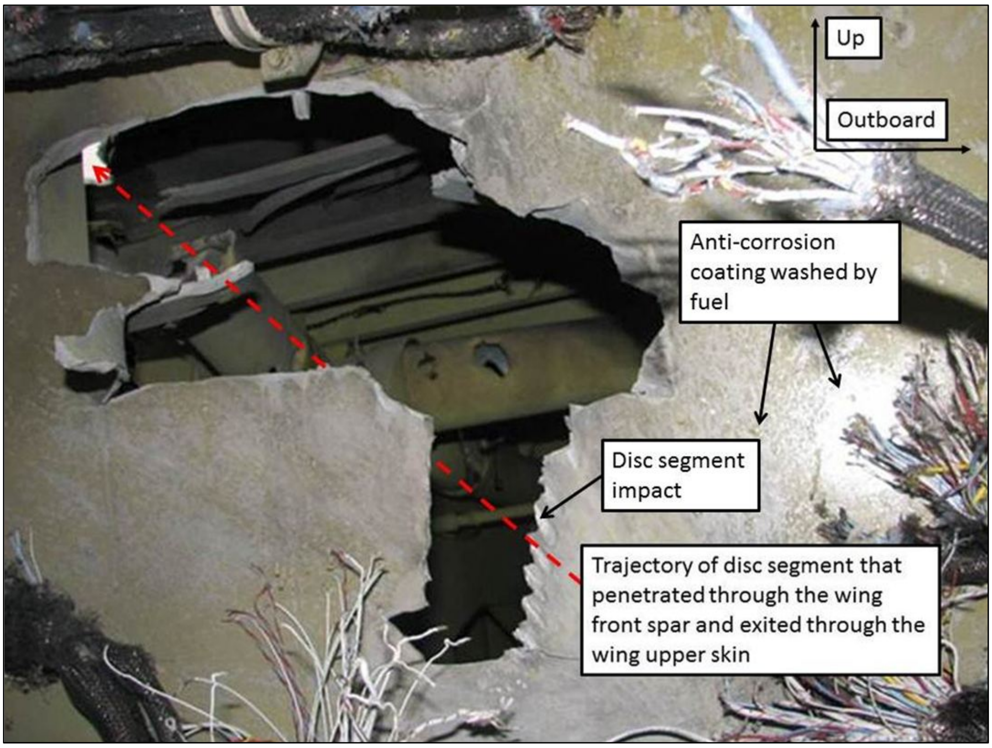


Figure B18: Damage schematic of the left wing front spar, looking at the forward side that extended between Rib 8 and Rib 14

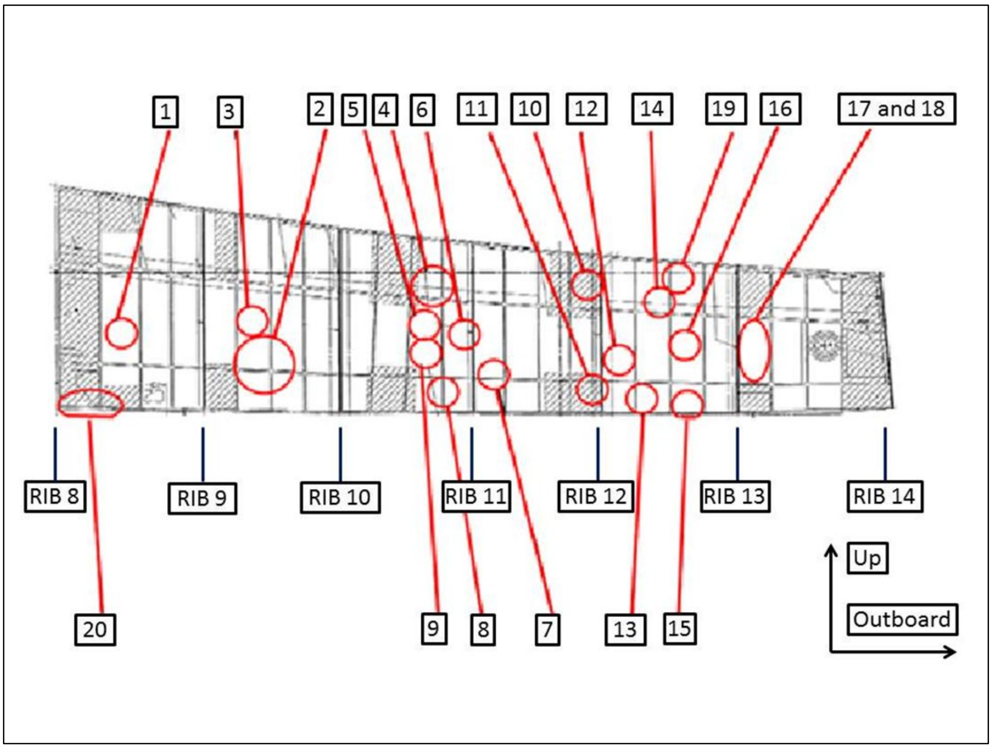


Image source: Airbus

Table B2: Wing front spar damage

Item	Rib	Damage type	Length	Width	Depth
1	8-9	Scrape	73	18	2
2	9-10	Hole & impact damage	350	350	Hole
3	9-10	Hole & impact damage	82	20	Hole
4	10-11	Hole & impact damage	150	120	Hole
5	10-11	Scrape	16	15	4
6	11	Scrape	7	5	1.25
7	11-12	Scrape	18	4	1
8	10-11	Scrape	12	5	2.1
9	10-11	Several scrapes	70	40	1.6
10	11-12	Several scrapes	22	11	3.5
11	12-13	Several scrapes	17	5	1
12	12-13	Hole	86	40	Hole
13	12-13	Scrape	5	2	1
14	12-13	Scrape	20	5	1.3
15	12-13	Scrape	22	10	0.4
16	12-13	Scrape	25	12	0.1
17	13-14	Scrape	9	5	0.35
19	13-14	Scrape	16	10	0.4
19	12-13	Scrape	22	1	0.1
20	8	Impact damage	nil		

Internal rib damage

The wing ribs are located between the spars and formed part of the integral wing box structure. Wing ribs 8 to 14 had impact damage from disc fragments, engine and airframe debris (Figure B19).

Engine debris was recovered from inside the wing between ribs 8-10 (Figure B20), including portions of IP turbine blade roots and engine case.

Figure B19: Internal wing rib damage showing damage in 5 locations. Viewed from lower wing skin access panel

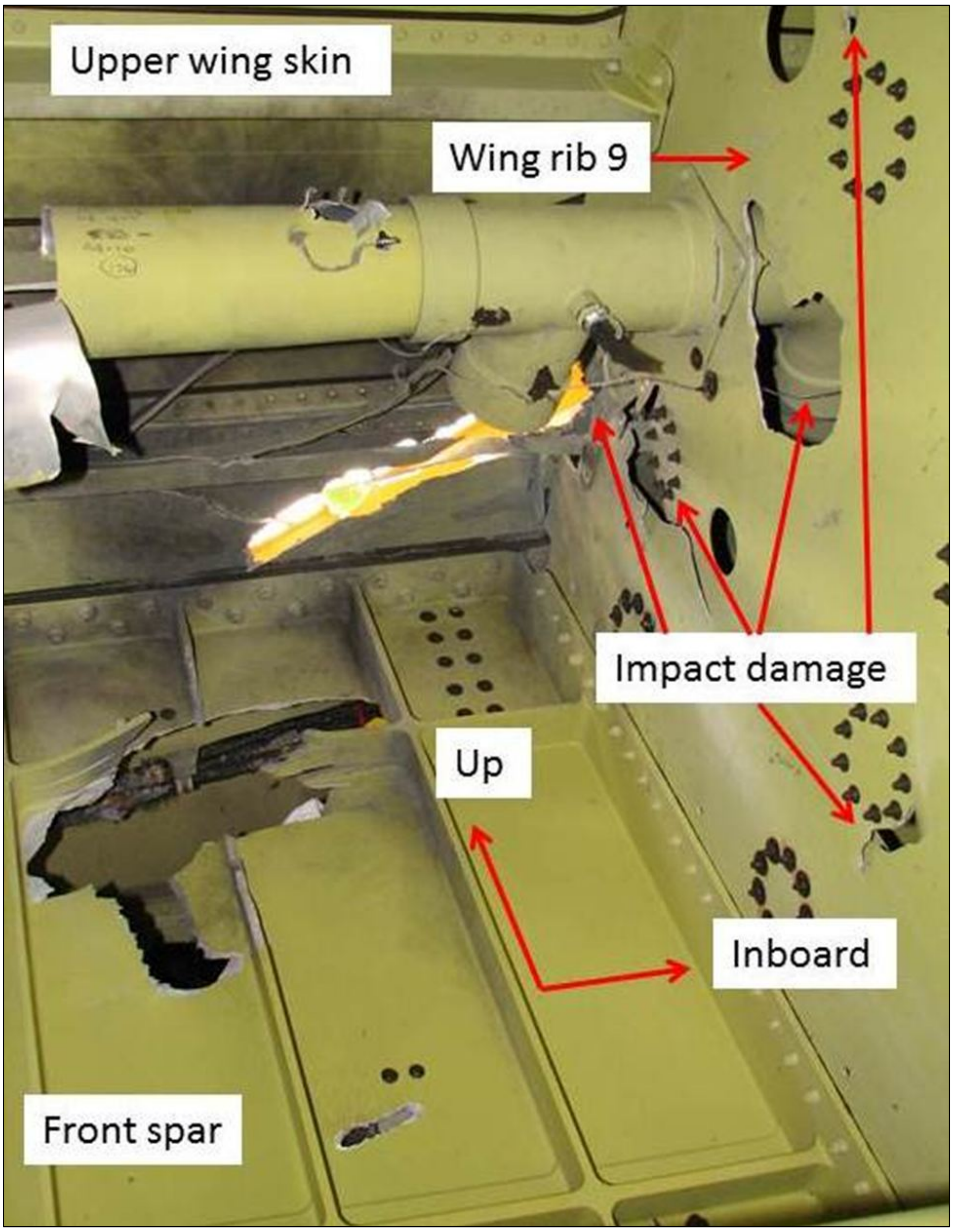


Figure B20: Location of major damage to ribs No. 8 to 14 (highlighted) and the front spar within the wing structure

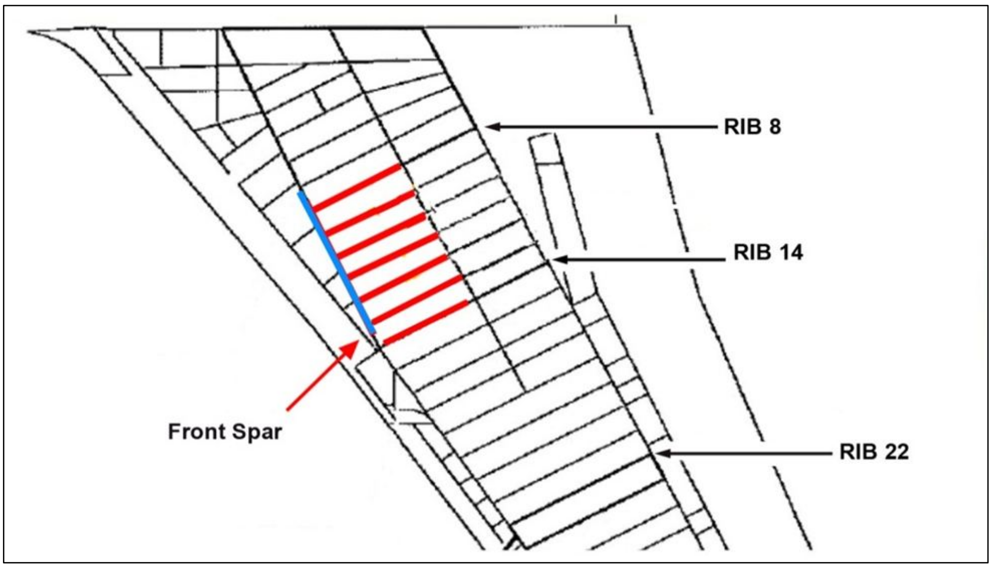


Image source: Airbus

Fuselage butt-straps No. 3 and No. 4

A butt-strap is a drop-forged fitting that overlaps and fastens the wing to the fuselage in conjunction with other structural components. Butt-straps No. 3 and 4 are located at the lower fuselage joint between Frames 46 and 56. These straps sustained impact damage, included gouging over an area of about 76 cm^2 from liberated engine debris between frames 51 and 52 (Figure B21, Figure B22, Figure B23).

Figure B21: Butt-straps No. 3 and No. 4 with damage highlighted

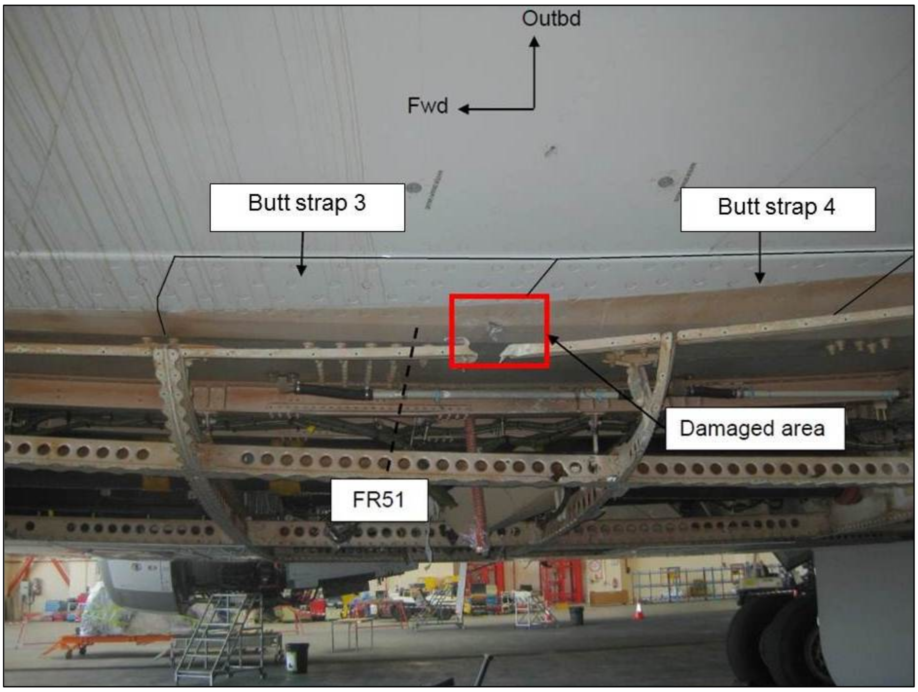


Image source: Airbus

Figure B22: Left butt-straps and damage location between Frame 52 and 51

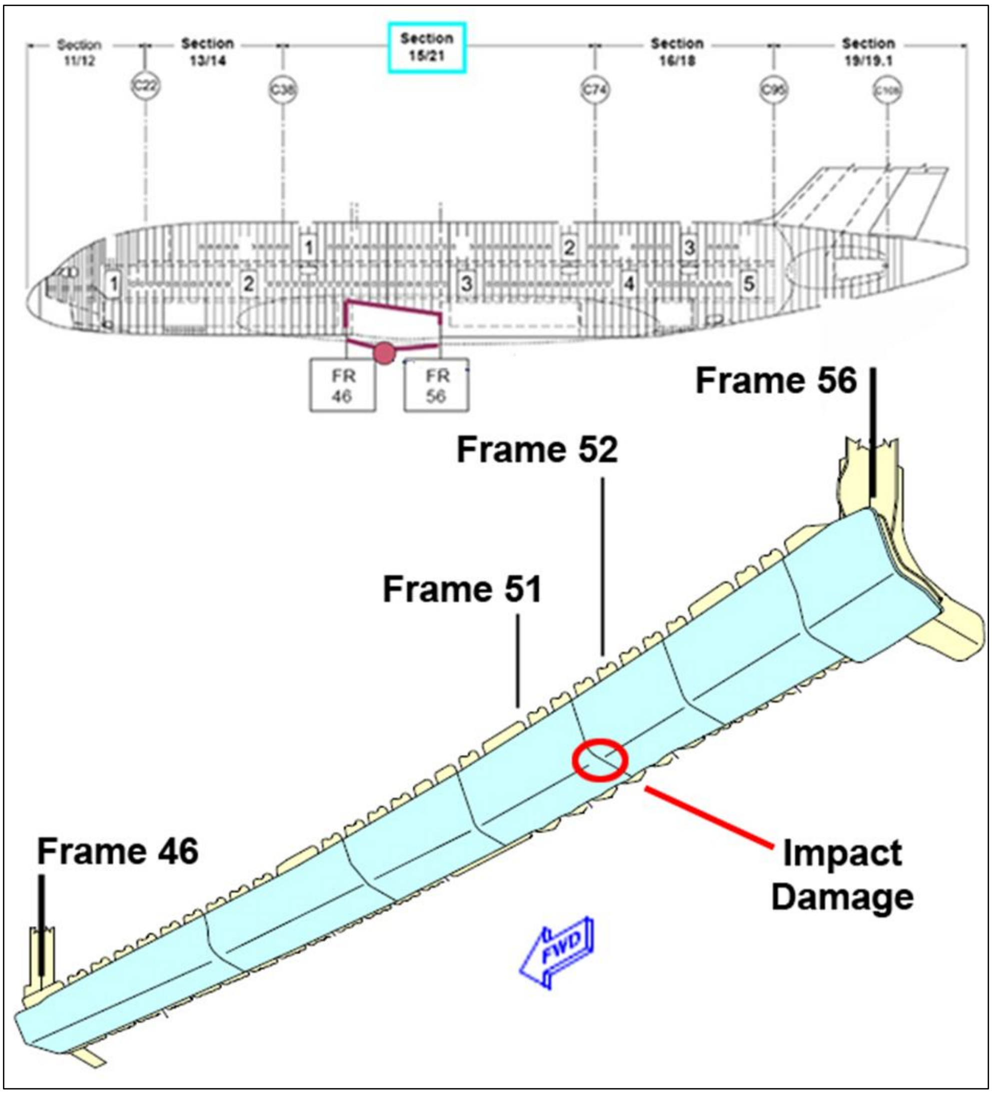


Image source: Airbus

Figure B23: Schematic of the butt-straps damage between Frame 52 and 51

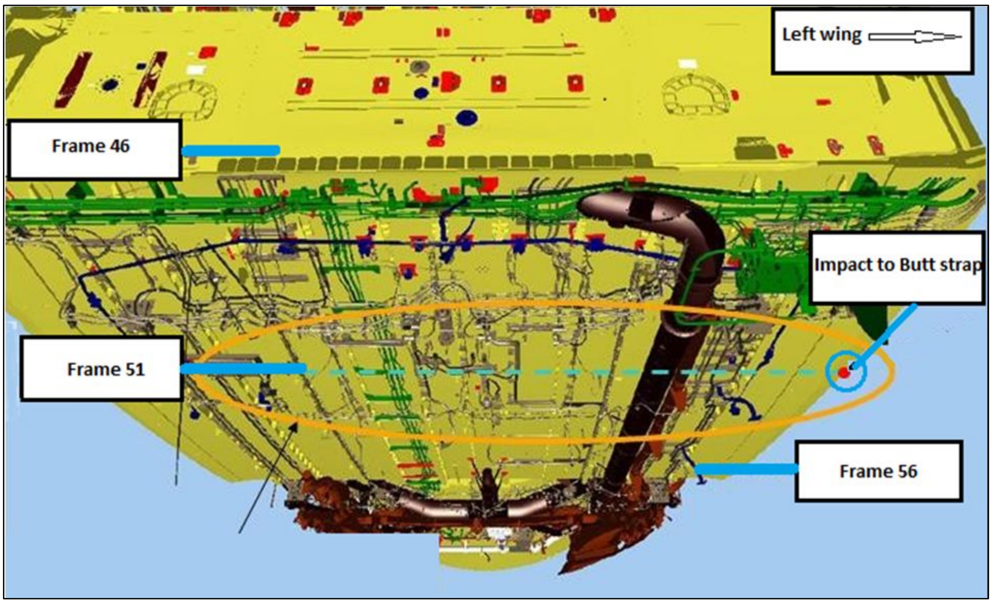


Image source: Airbus

Minor damage

Fuselage damage

The left side of the fuselage (Figures B24 and B25), from passenger door 2 left to the horizontal stabiliser, had impact damage from engine and airframe debris. Although the debris did not penetrate through the fuselage skin, that damage was considered significant and required repair prior to further flight.

Centre fuselage – section 15 damage

- Three fuselage skin panels were damaged in section 15 (Figure B24 and Table B3) that damage included:
- FR46-FR62, STGR6LH-STGR22LH, 39 items of impact damage (scratches, gouges and dents) and 6 windows damaged in that location.
- FR46-FR62, STGR22LH-STGR39LH, 19 items of impact damage (scratches, gouges and dents) and 1 window damaged in that location.
- FR62-FR74, STGR22LH-STGR39LH, 4 items of impact damage (scratches, gouges and dents) and 1 window damaged in that location.

Figure B24: Centre fuselage – section 15 highlighted in orange

Image source: Airbus

Table B3: Fuselage damage along section 15

Type of Damage	Between Frames	Between Stringers
Gouges	54-55	18-21
Gouges	55-56	17-18
Gouges	55-56	22
Gouges	55-56	18-21
Gouges	55-56	18-21
Scratches	55-56	21
Scratches	56-57	18-21
Gouges on Window	56-57	18-21
Gouges	56-57	15-16
Gouges	56-57	21
Gouges	57-58	18-21
Gouges	58-59	18-21
Scratches Window/Frame	67-68	18-21
Gouge	50-51	25
Gouge	50-51	25-26
Gouge	50-51	24-25
Gouge	54-55	24-25
Scratch	54-55	23-24
Gouge	54-55	26-27
Gouge	55-56	25-26
Gouge	55-56	25-26
Gouge	56-57	24-25
Gouge	59-60	22-23
Gouge/Dent	47-48	28-29

Type of Damage	Between Frames	Between Stringers
Gouge	52-53	30-31
Gouge	54-55	33
Gouge	52-53	32-33
Gouge	53-54	27-28
Gouge	55-56	27-28
Gouge	59-60	27-28
Gouge	61-62	35-38
Gouge	63-64	29
Gouge on Window	65-66	35-38
Gouge	68	32-33
Gouge	69	33-34

Fuselage damage along fuselage section 19

Three fuselage skin panels were damaged in section 19 (Figure B25), Damage at that location required repair prior to further flight, that damage included (Table B4):

- FR74-FR89, STGR6LH-STGR22LH, 5 items of impact damage (scratches, gouges and dents) in that location.
- FR74-FR81, STGR22LH-STGR39LH, 5 items of impact damage (scratches, gouges and dents) in that location.
- FR81-FR89, STGR22LH-STGR39LH, 3 items of impact damage (scratches, gouges and dents) in that location.

Figure B25: rear fuselage – section 19 highlighted in orange

Image source: Airbus

Table B4: Fuselage damage section 19

Type of Damage	Between Frames	Between stringers
Gouge	77-78	22-23
Gouge	79-80	18-21
Gouge	79-80	18-21
Gouge/Scratch	81-82	18-21
Gouge	81-82	18-21
Gouge	74-75	23-24
Gouge	75-76	23-24
Gouge	75-76	23
Gouge	81	23-24
Scratch	83	23-24
Gouge	88-89	16-17
Gouge	75-76	30-31
Gouge	81-82	38-39

Trimmable Horizontal Stabiliser

The left side of the trimmable horizontal stabiliser leading edge (Figure B26) was damaged from engine and airframe debris at five locations along the leading edge (scratches, gouges and dents). Damage at that location required repair prior to flight.

Figure B26: Trimmable Horizontal Stabiliser leading edge skin panels highlighted in orange

Image source: Airbus

Vertical stabiliser

The left side of the vertical stabiliser (Figure B27) was damaged from engine and airframe debris at five locations along the leading edge (scratches). Damage at that location required repair prior to flight.

Figure B27: Vertical stabiliser spar box skin highlighted in orange

Image source: Airbus

Wing leading edge structure

The left wing leading edge substructure, droop nose flaps, composite skins and components in that area, between the number two engine and the fuselage (Figures B28-B30) were subject to significant impact forces. Although the damage incurred was significant, it was not considered to be major structural damage.

Figure B28: Wing leading edge sub structure diagram

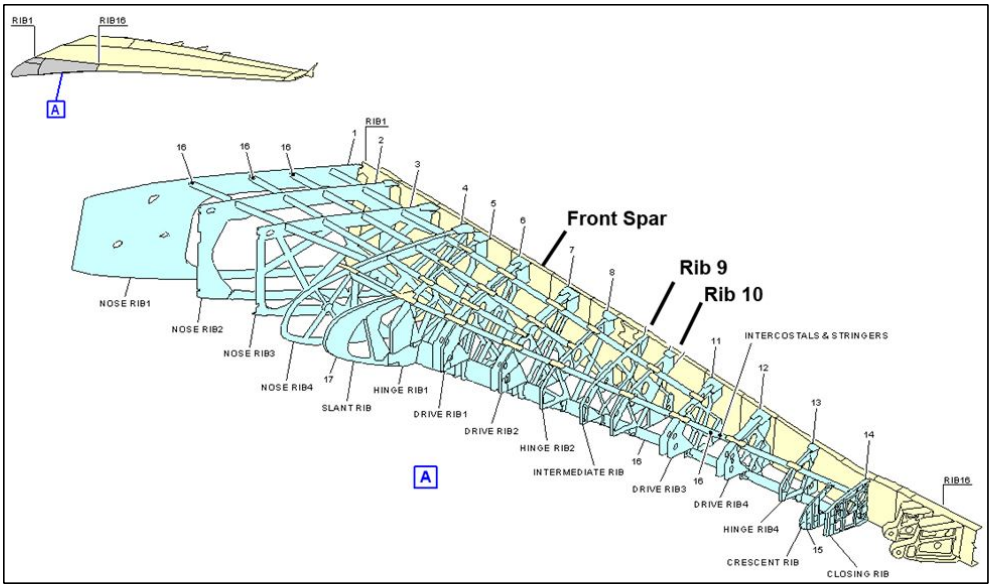


Image source: Airbus

Figure B29: Engine debris impacted the wing droop nose slat, wing structure and components within the leading edge—viewed from the upper wing surface

Figure B30: Damage to the drive motor located at leading edge Drive Rib No. 4

Belly fairing and internal structure

The belly fairing and centre wing box structure (Figure B31) of the aircraft sustained significant damage from engine fragments. A smaller-sized, high-energy disc fragment penetrated the belly region of frame 51 (Figure B32 to Figure B36).

Figure B31: General location of the belly fairing panels on the A380 aircraft highlighted in orange

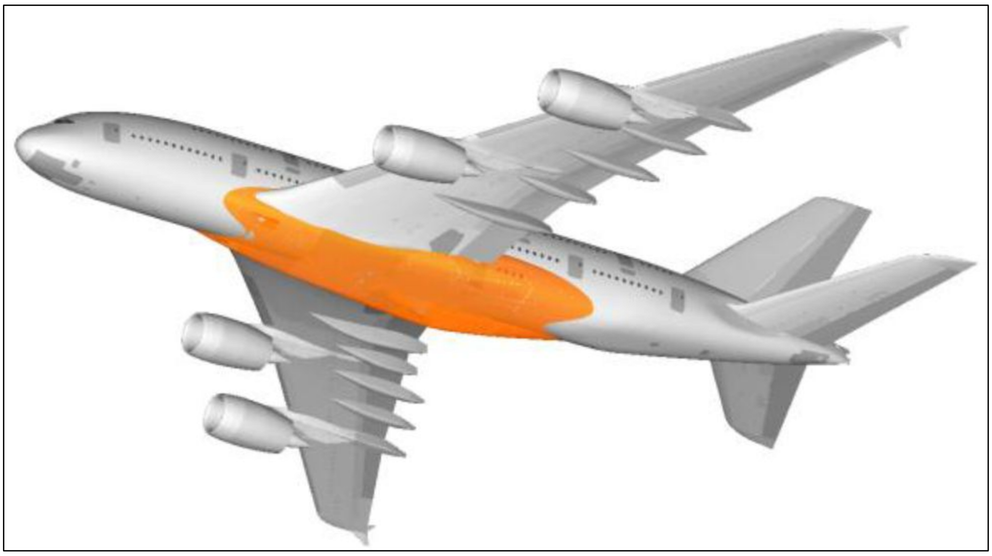


Image source: Airbus

Figure B32: A smaller, high energy fragment of the IP turbine cut a swath directly through the belly fairing of the aircraft—view looking to the right of the aircraft with the No. 3 engine in the background.

Figure B33: Structural arrangement of the fuselage between frames 50 and 54 showing the damage points from engine debris that penetrated the belly fairing

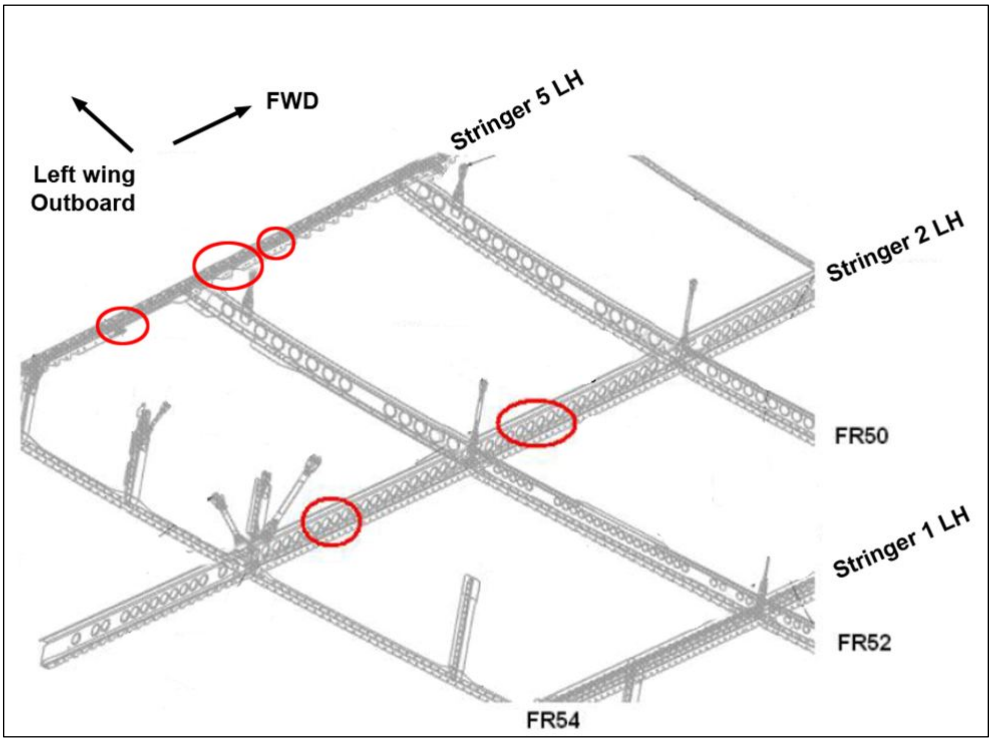


Image source: Airbus

Figure B34: Impact and penetration damage to the fuselage structure between airframe locations FR50 and FR54 from a high-energy fragment of the IP turbine disc

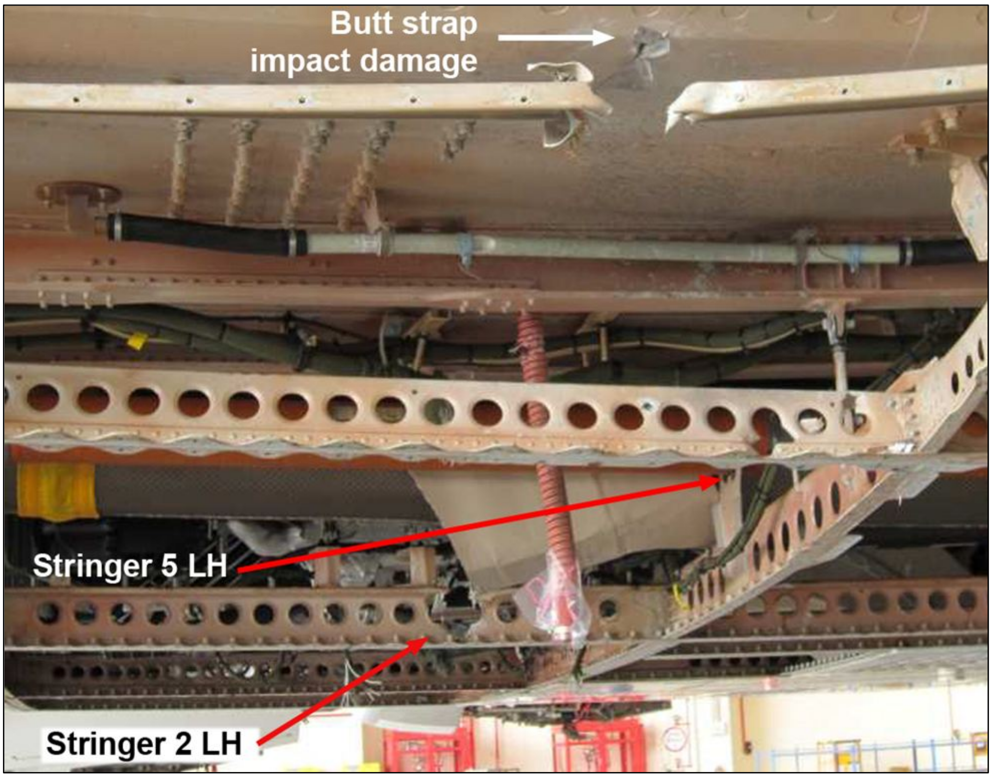


Figure B35: Stringer 2 LH impact damage

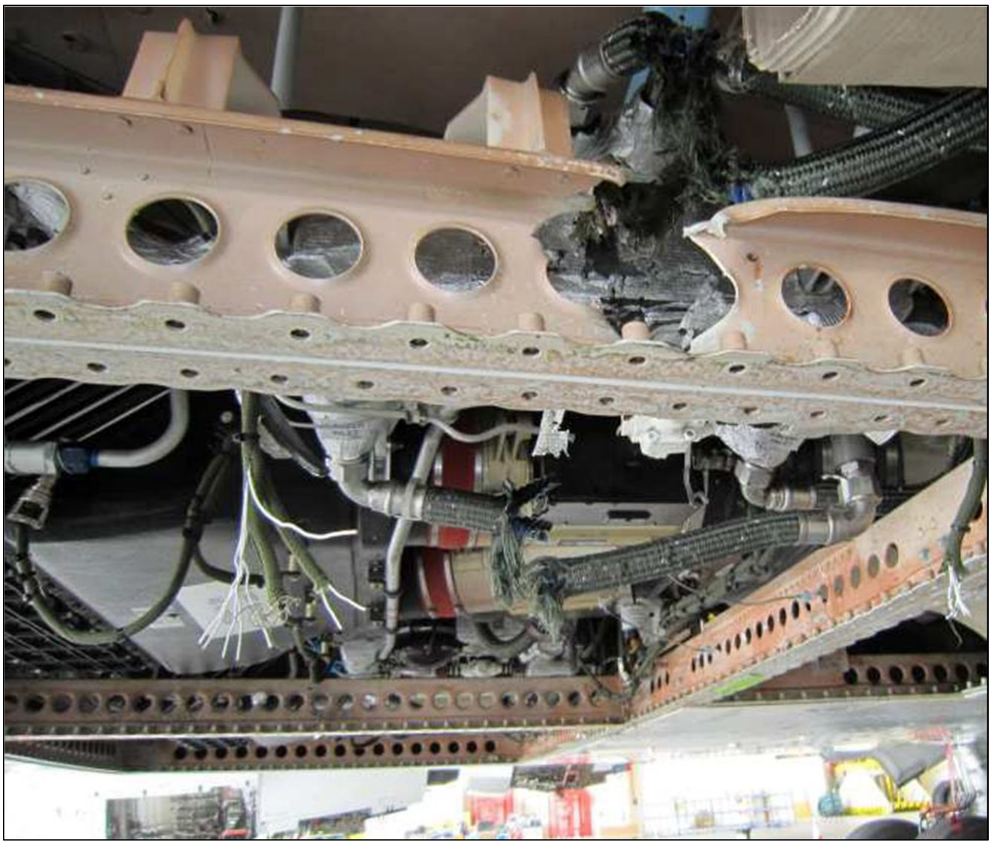


Figure B36: Stringer 5 LH impact damage

Aircraft systems

System effects – wiring damage

The debris from the engine failure directly damaged a number of systems, which in turn affected a number of other systems. Although some systems sustained direct mechanical damage, most of the affected systems were damaged through debris impact to the respective wiring looms. Two main wiring looms were impacted by debris; one running through the leading edge of the left wing, and the other in the belly fairing. This resulted in damage to about 650 wires in total (Figure B37 to Figure B51).

Figure B37: Representation of the smaller, high energy disc fragment's trajectory through the aircraft belly fairing (keel beam) which severed wiring harnesses

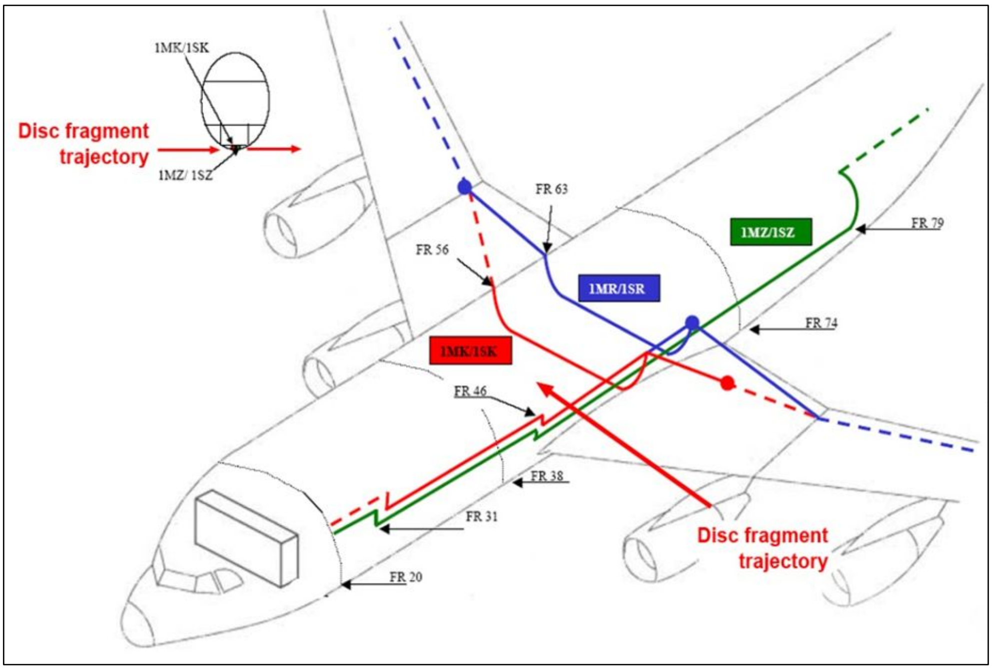


Image source: Airbus

Figure B38: Representation of the disc fragment trajectory through the aircraft left wing leading edge which severed numerous wiring harnesses

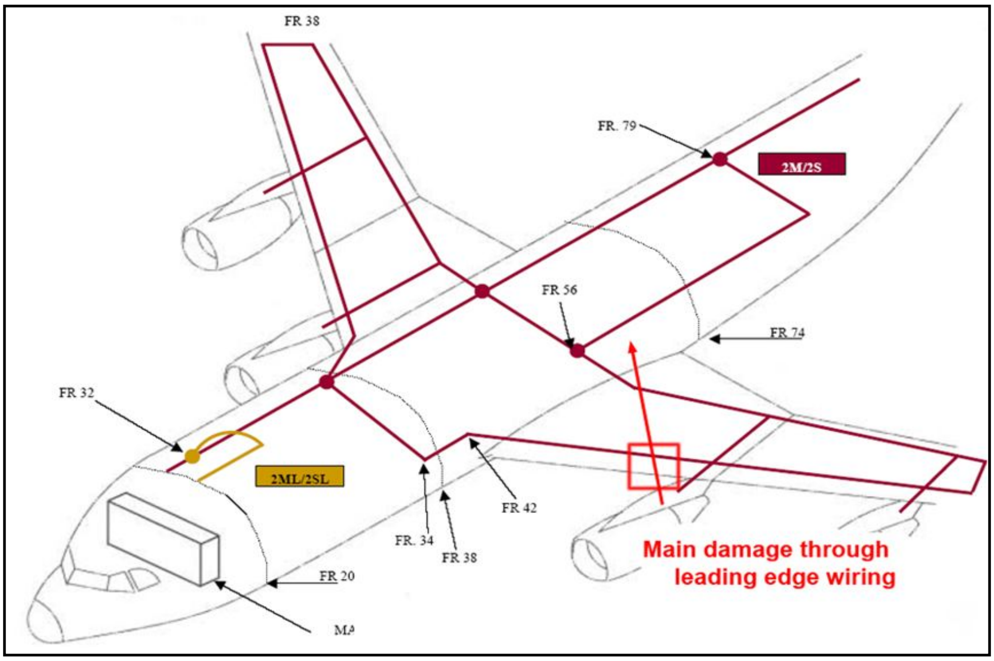


Image source: Airbus

Figure B39: General location of the wiring harness damage within the leading edge of the left wing—red framed area expanded in Figure B40

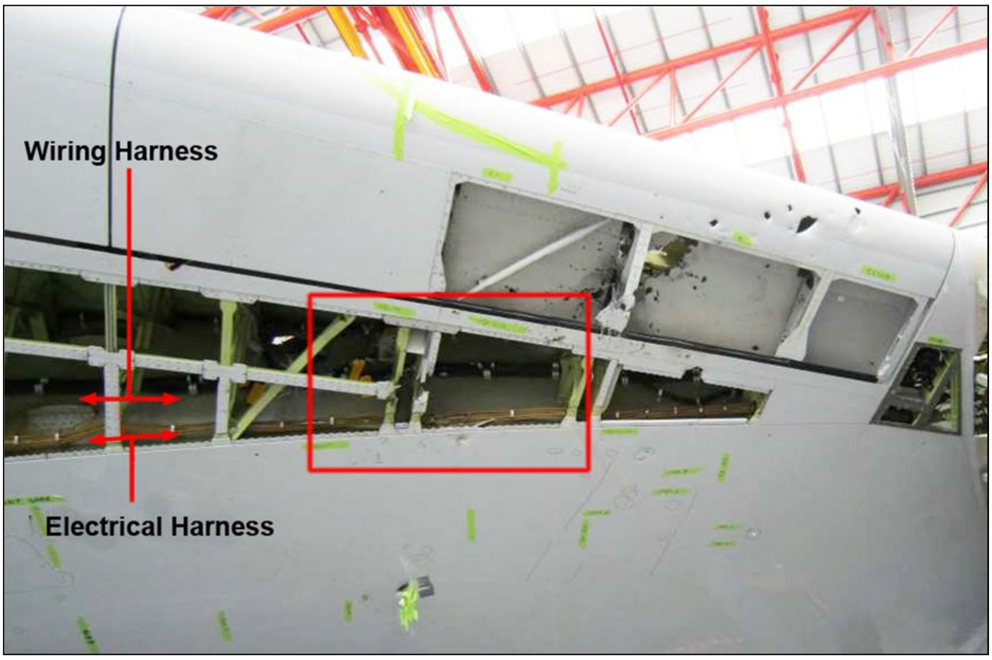


Figure B40: Wiring harness damage at three main locations within the leading edge of the Rib 9 and Rib 10 area

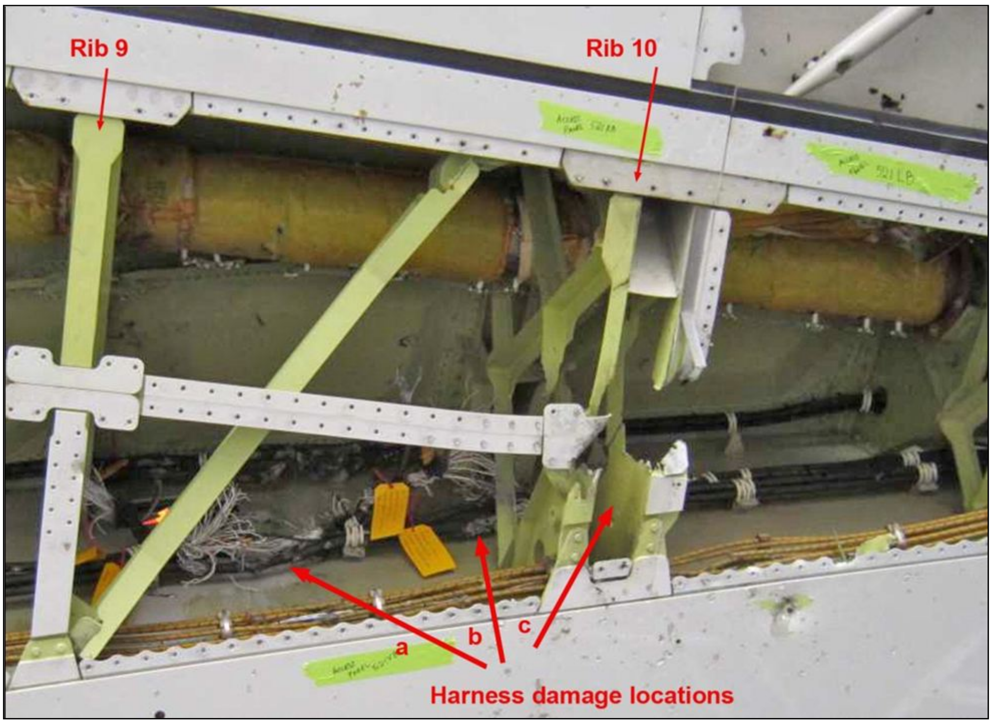


Figure B41: Damage location (a) from Figure B40 showing complete severing of the route 1M, 1S, 2S and 2M wiring harnesses—the entry/exit hole through the front spar and upper wing skin from a high-energy disc fragment is also apparent

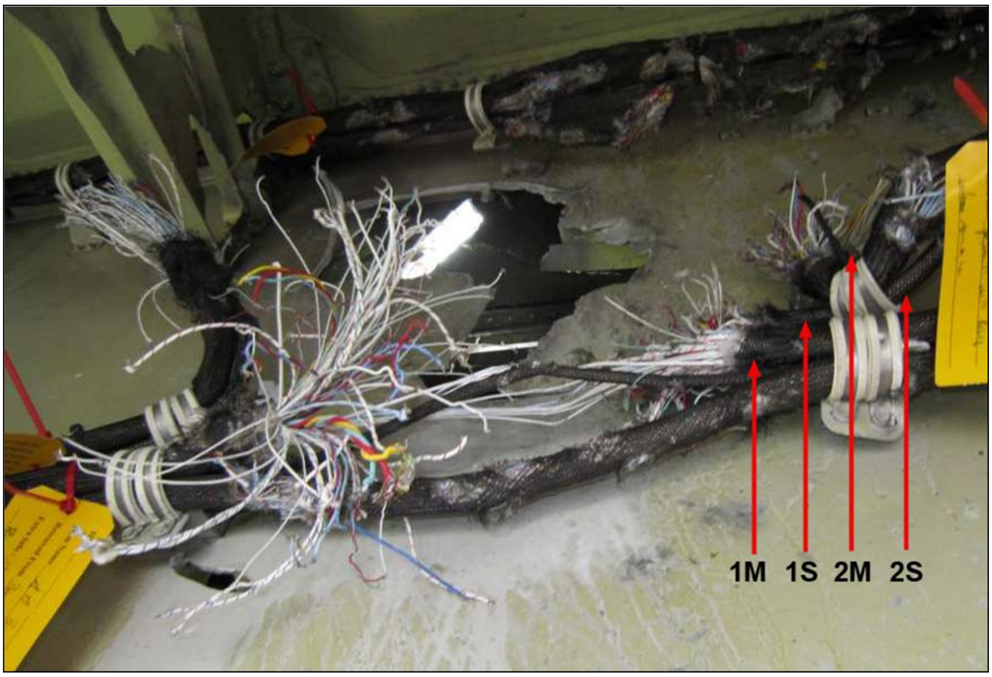


Figure B42: Damage location (b) from Figure B40

Figure B43: Damage location (c) from Figure B40

Figure B44: Main damage to the route 1P route harnesses (looms 3025VB, 3027VB, 3031VB and 3033VB), between Rib 9 and Rib 10 of the left wing lead edge. All wires within the harness were completely severed. The debris entry/exit hole through the front spar is also evident

Figure B45: Detailed locations of damage to the 1G route (electrical harness 3029VB), primarily between Rib 9, 10 and 11 of the left wing; four of the 9 feeder and exciter cables have been severed—red framed area expanded in Figure B46

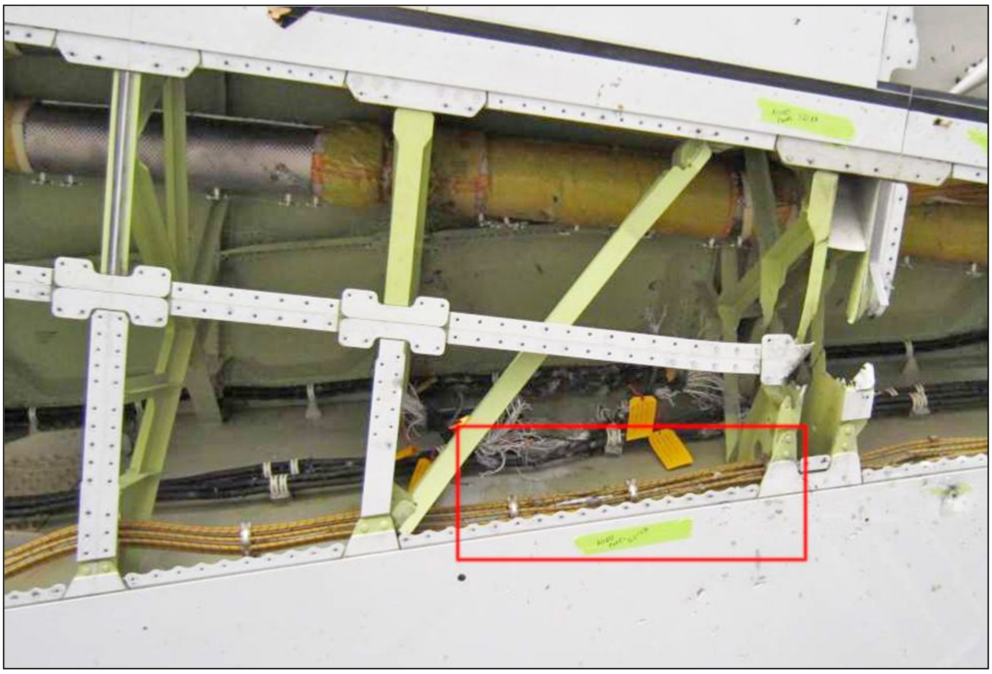


Figure B46: Close detail of damage to route 1G, 3029VB harness, between Rib 10 and Rib 11

Figure B47: Wiring harnesses within the belly region of the aircraft structure were damaged from the liberated engine debris. Shown is harness 1525VB (route 1MK). All wires were severed

Figure B48: Wiring harnesses within the belly region of the aircraft structure were damaged from the liberated engine debris. Shown is harness 1532VB (route 4M and 2MZ). All wires were severed

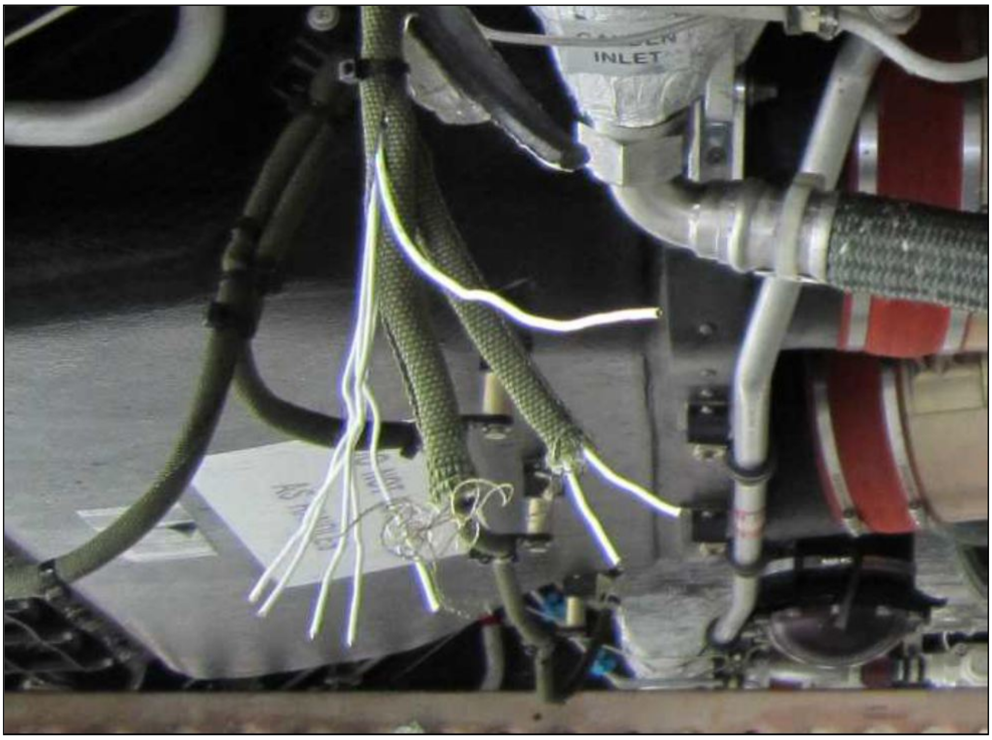
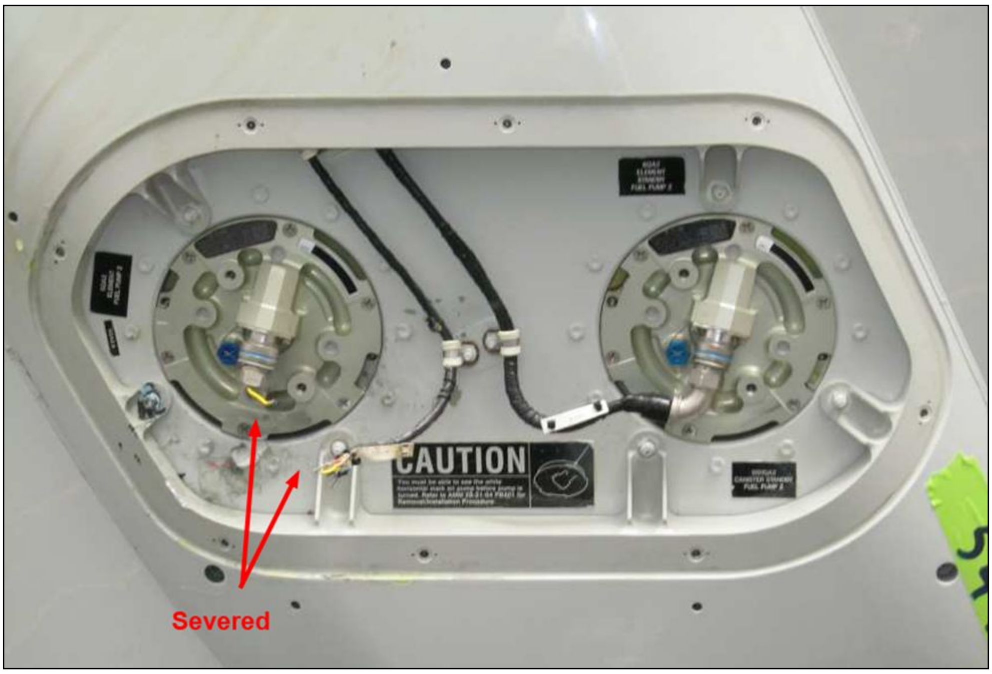


Figure B49: Wiring harnesses within the belly region of the aircraft structure were damaged from the liberated engine debris. Shown is harness 500HW1 (route 1PZ). All wires were severed

Figure B50: Location of the left wing fuel pump harness damage, routes 1M (fuel pump) and 2M (standby fuel pump) were both severed

Figure B51: Damage to the 1M harness that supplied to the left wing forward fuel pump



Depending on their use, several types of electrical wires were fitted within the A380:

- type P (power supply distribution)
- type S (sensitive signals)
- type M (miscellaneous)
- type G (electrical power services – feeders).

The following section outlines the aircraft systems that were affected either as a direct result of impact damage from engine debris, or from damage to electrical wiring that rendered them inoperative, either by design, or as a result of a loss of monitoring function.

Table B5: Airframe systems affected by the wiring damage

Air conditioning	Fire Protection	Flight Controls
Electrical Power	Fuel	Hydraulic Power
Ice and Rain Protection	Landing Gear	Lights
Pneumatic	Water/Waste	Cabin Systems
Engine Fuel and Control	Ignition	Engine Controls
Exhaust	Oil	

Table B6: Airframe wiring harnesses severed by the engine debris

Wiring Route	Harness	Location, route detail
1MK	1525VB	Aircraft belly, between Frame 51 and Frame 52
4M, 2MZ	1532VB	Aircraft belly, between Frame 51 and Frame 52
1PZ	500HW1	Aircraft belly, between Frame 51 and Frame 52
1M	3001VB	Left wing leading edge, between Rib 9 and Rib 10
2M	3007VB	Left wing leading edge, between Rib 9 and Rib 10
1S	3013VB	Left wing leading edge, between Rib 9 and Rib 10
2S	3019VB	Left wing leading edge, between Rib 9 and Rib 10
1P	3025VB 3031VB 3027VB 3031VB 3035VB	Left wing leading edge, between Rib 9 and Rib 10
1M	3033VB	Left wing, lower skin, feed tank 2 fuel pump
2M	3031VB	Left wing, lower skin, feed tank 2, standby fuel pump
1G/1E	3041VB 3029VB	Left wing leading edge, between Rib 9 and Rib 10
3G/3E	3043VB	Left wing leading edge, between Rib 11 and Rib 12

Hydraulic systems

The A380 had two independent hydraulic systems identified as the Green system and the Yellow system (Figure B52). The Green system was powered by hydraulic pumps driven by engines No. 1 and 2 and the Yellow system by hydraulic pumps driven by engines No. 3 and 4. Two hydraulic pumps were fitted to each engine. On the ground without engines running, hydraulic power was provided by electric motor-driven pumps. The Green, Yellow and ground hydraulic systems were independent and hydraulic power could not be transferred from one system to the other.

Figure B52: Hydraulic power generation

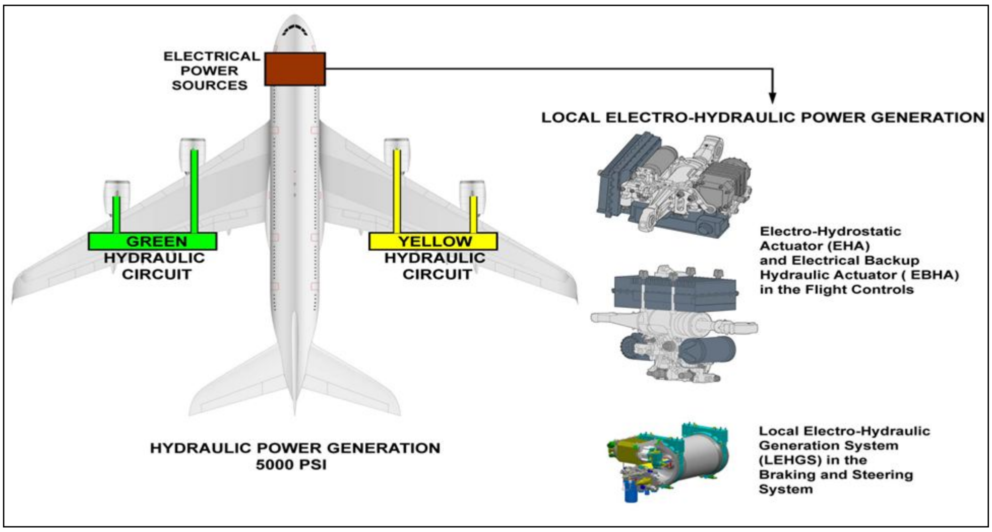


Image source: Airbus

The hydraulic system provided power to the flight controls, the landing gear systems and the cargo doors. In the event of a hydraulic system failure, the following redundant systems were available to provide backup hydraulic power (Figure B53):

- electro-hydrostatic actuators (EHA) and electrical backup hydraulic actuators (EBHA) for use by the flight control systems
- a local electro-hydraulic generation system (LEHGS), which can be used by the landing gear systems.

Figure B53: Green, Yellow and electrical hydraulic back up architecture

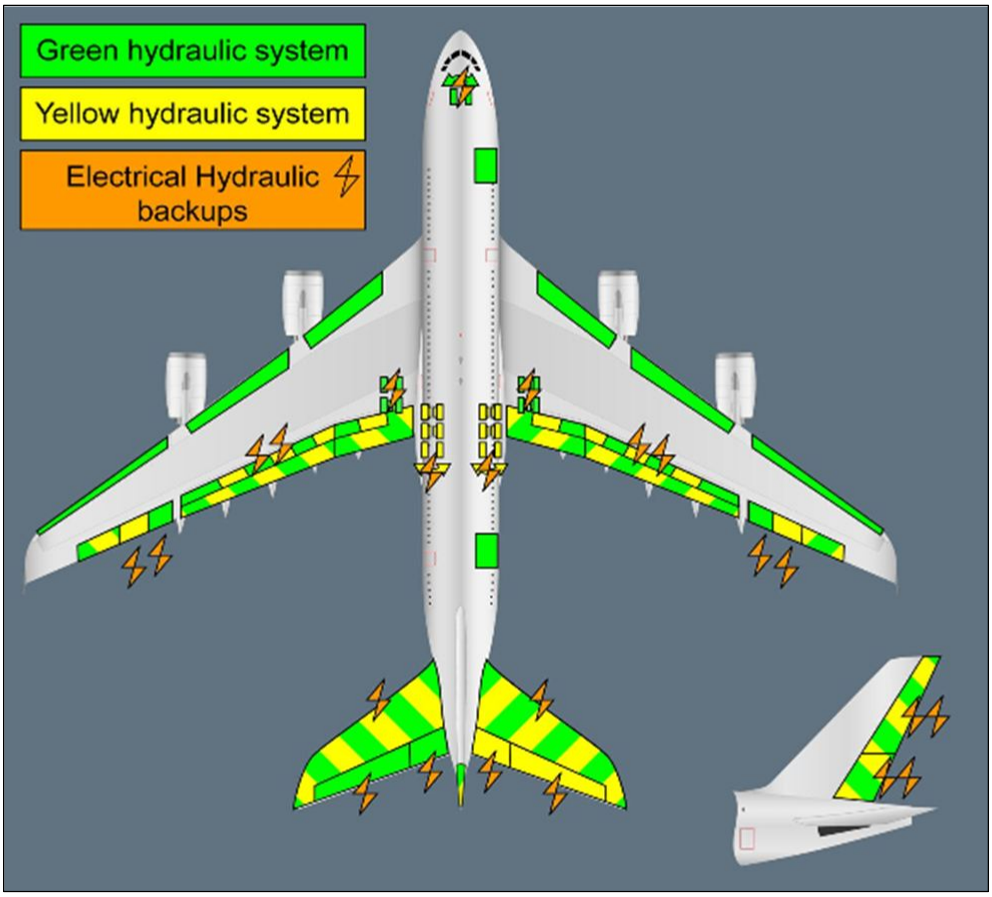


Image source: Airbus

Engine debris damage to the aircraft's wiring resulted in the loss of monitoring capability of the Green hydraulic system fluid level, system pressure and reservoir air pressure. The crew reported that they observed the Green hydraulic system fluid level indicator decreasing. At about the same time, the Electronic Centralised Aircraft Monitoring (ECAM) system displayed messages that required the crew to shut down the Green hydraulic system by disconnecting the No. 1 and 2 engine-driven hydraulic pumps. However, the wiring damage prevented the No. 2 engine hydraulic pumps from being disconnected and, as a result of the No. 2 engine continuing to windmill in flight, a residual hydraulic pressure of about 1,000 psi was maintained. Normal hydraulic system pressure was 5,000 psi. Subsequent examination of the Green hydraulic system found that there was no fluid loss.

Damage to the wiring also resulted in the loss of monitoring capability of the Yellow hydraulic system engine-driven pumps on the No. 4 engine and the crew disconnected both pumps as per the ECAM procedure. The Yellow hydraulic system was powered by the No. 3 engine for the remainder of the flight. The Yellow hydraulic system maintained 5,000 psi for the remainder of the flight and subsequent examination found no fluid loss.

Electrical system

General

The A380 electrical system included alternating current (AC) and direct current (DC) power sources. In flight the AC system was powered by four engine-driven generators and one auxiliary power unit (APU)-driven generator if required. In an emergency situation a ram air turbine[113] was capable of supplying essential electrical loads.

The aircraft's DC system was powered by the AC network via 5 transformer rectifier units. If AC power was not available, limited DC power could be provided from on board batteries.

The APU had two AC generators labelled 'A' and 'B'. In ground mode, APU generators A and B were available. In flight, only APU generator A was available. Therefore, on the ground the aircraft could draw AC power from either or both APU-driven AC generator or from external power.

During normal operations, AC power was distributed to applicable systems via four independent AC busbars (Figure B54).[114] In emergency situations, the distribution of AC power was via one essential and one emergency busbar.

DC power was normally distributed to applicable systems via two independent DC busbars. In emergency situations, there was one DC essential busbar.

Figure B54: A380 electrical network architecture

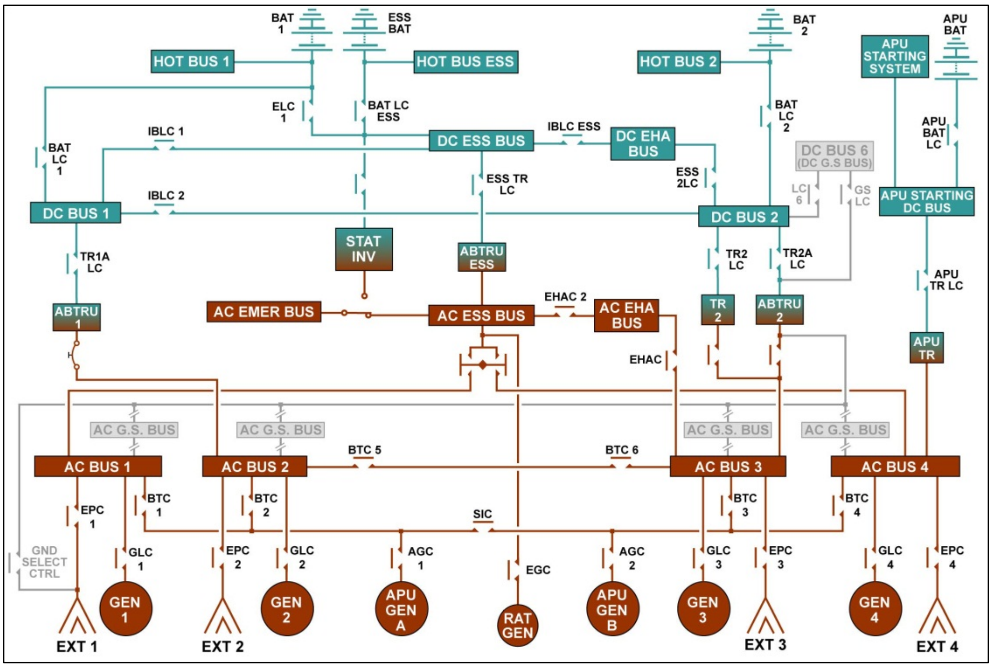


Image source: Airbus

113 An air-driven turbine that is coupled to an AC generator and can be deployed into the oncoming air flow to provide emergency electrical power.

114 A busbar is used to distribute electrical power to multiple outlets.

Effect of the damage to the aircraft on power supply

The AC power generation feed lines from generators No. 1 and No. 2 were severed by engine debris, removing the power supply to AC busbar 1 (AC BUS 1) and AC BUS 2 (Figure B55). The system automatically reconfigured AC BUS 1 to obtain its supply from AC BUS 4. When the crew started the APU, AC BUS 1 was reconfigured to be supplied with AC power from APU generator A.

In the event of a loss of power to AC BUS 2, the system would normally reconfigure to receive supply from AC BUS 3. Due to residual current being sensed from the No. 2 engine generator, the Generator and Ground Power Control Unit (GGPCU) considered the No. 2 generator was still online and isolated AC BUS 2. This resulted in AC BUS 2 being unable to be reconfigured and it was not powered.

The loss of AC BUS 2 resulted in the loss of Air Data Inertial Reference Unit (ADIRU) 3. One of ADIRU 3's functions is to provide a speed signal to the electrical power system. This speed signal is used by the electrical power system to determine which APU generators are available to supply power. When power was lost to ADIRU 3, its speed signal output was frozen in the 'speed higher than 50 knots' configuration (that is, in the air mode).

As the electrical system was locked in the air mode, only APU generator A was available to supply power following the shutdown of engines No. 3 and 4. This resulted in only AC BUS 1 and the essential bus bar continuing to be supplied with AC power. This limited power source resulted in some flight deck display screens being lost and only the VHF 1 radio being available for use by the flight crew.

Figure B55: Engine feeder routing and the main damage location

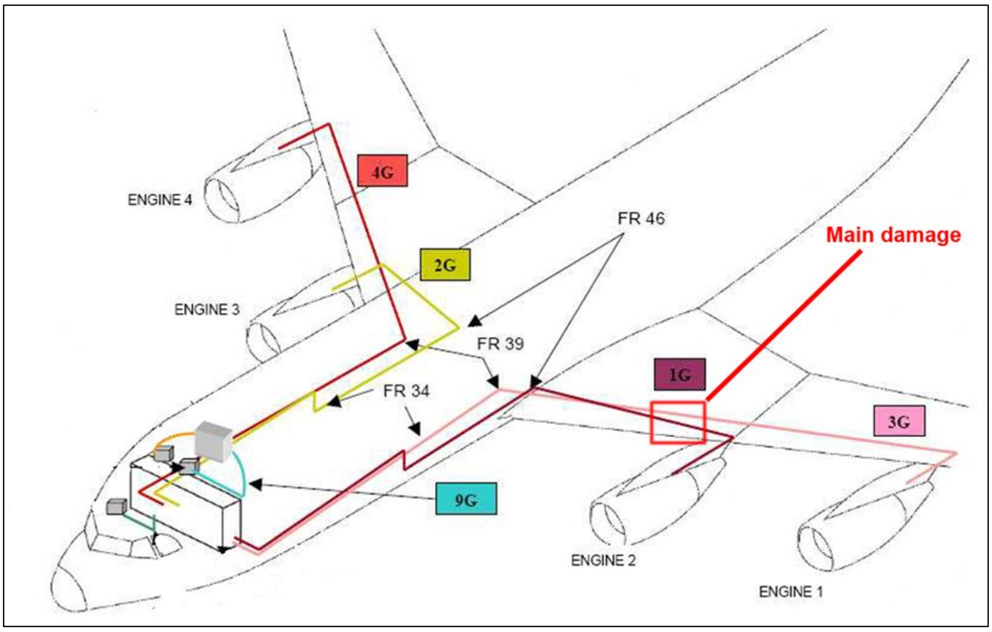


Image source: Airbus

Flight controls

Immediately after the uncontained failure, the following flight controls were inoperative:

- all slats

- all droop nose flaps
- the left mid aileron
- spoilers No. 4 and 6 on the left wing.

As a result of the inoperative slats, the flight law changed from normal law to alternate law.[115] Approximately 10 minutes later, coincident with the depressurisation of the Green hydraulic system, the following additional flight controls were unavailable:

- the left and right outboard ailerons
- spoilers No. 2 and 8 on the left wing
- spoilers No. 2 and 8 on the right wing
- spoiler No. 4 on the right wing.

All trailing edge flaps, the elevators and the trimmable horizontal stabiliser and rudder control surfaces remained available for the duration of the flight.

The availability of the primary and secondary flight control surfaces following the uncontained engine failure and subsequent ECAM actions by the crew are depicted in Figure B56.

Figure B56: Flight control surface status following the engine failure and ECAM actions by the crew (unavailable control surfaces highlighted in red)

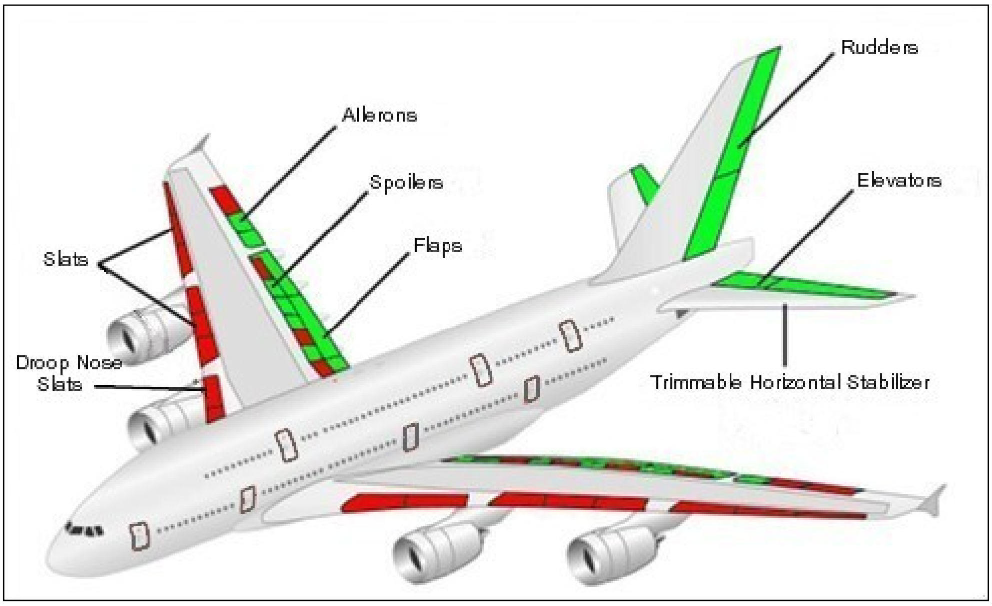


Image source: Airbus

Trailing edge flaps

Flap extension and retraction was powered by two hydraulic motors. One motor was supplied with hydraulic pressure from the Green hydraulic system and the other from the Yellow hydraulic system. The flaps were operated through gearboxes, torque shafts and six rotary actuators (two for each flap).

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115 In normal law, all protections are available. In alternate law, some protections are reduced or lost.

No wiring damage affected the flap system; however the flaps operated at half normal speed due to the loss of the Green hydraulic system.

Leading edge lift augmentation devices

The slat and droop nose extension and retraction mechanism was normally powered by one hydraulic motor supplied by the Green hydraulic system, and one electric motor. During normal operation both drives were active. The slats and droop noses were operated through gearboxes, torque shafts and 16 rotary actuators—two for each slat and two for each droop nose (Figure B57).

Wiring looms to the asymmetry position pickoff units and the wingtip power-off brakes were severed by engine debris. This damage affected the control and monitoring of the left wing slats. In addition the engine debris severed the left wing slat transmission. As a result the left wingtip power-off brakes automatically secured the slats and droop nose in the retracted position.[116]

Figure B57: Leading edge lift augmentation mechanism

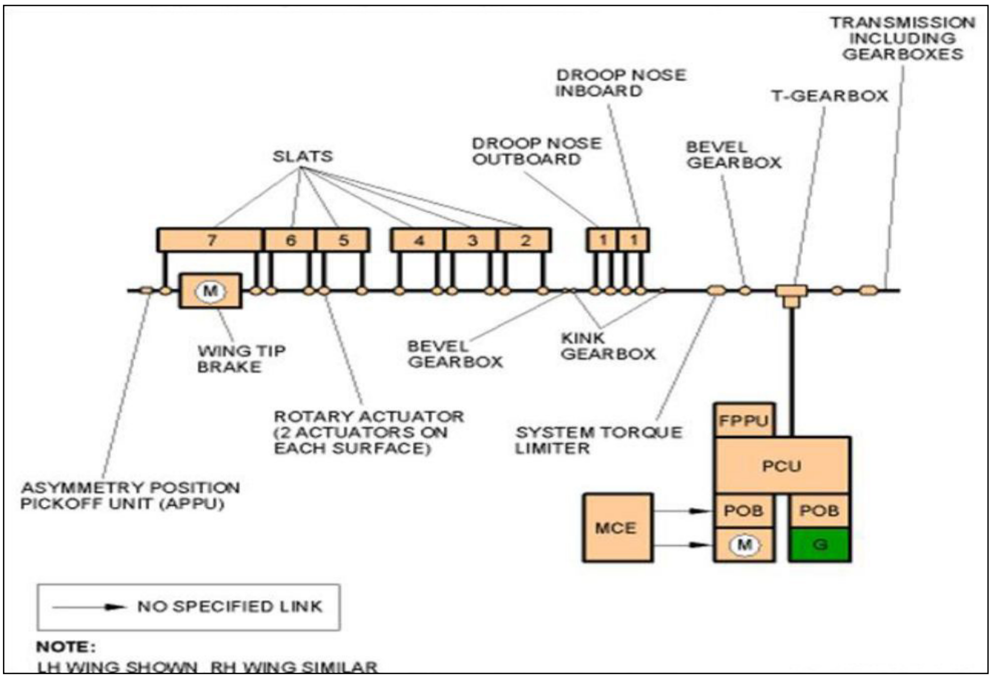

Image source: Airbus

Primary flight controls actuation systems

The primary flight control actuation systems in the A380 were powered by four independent systems; two hydraulic systems and two electrical systems (Figure B58). The aircraft was able to fly with only one hydraulic or one electrical system available. The control surfaces were actuated by the following mechanisms:

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116 The asymmetry position pickoff units determine any asymmetry between the left and right wing leading edge lift augmentation devices. The wingtip power-off brakes automatically secure the left and right wing leading edge lift augmentation devices when commanded, or if power is lost to the brakes.

- Conventional servocontrols[117]:
  - four at the elevators (one inner and one outer per side)
  - two at the pitch trim
  - eight at the ailerons (two outer, one mid, one inner per wing)
  - one at spoilers 1, 2, 3, 4, 7, and 8.
- Electro-hydrostatic actuators (EHA):
  - four at the elevators (one inner and one outer per side)
  - one at the pitch trim
  - four at the ailerons (one mid and one inner per wing).
- Electrical backup hydraulic actuators (EBHA)
  - four at the rudders (two per surface)
  - spoilers 5 and 6.

EHAs contained their own electro-hydraulic generation system that received commands from the fight control computers. EBHAs operated like a conventional servo-control, however, a specific mode called backup enabled them to operate like EHAs.

Figure B58: Primary flight controls actuation system (a red cross indicates the loss of actuator operation after the engine failure)

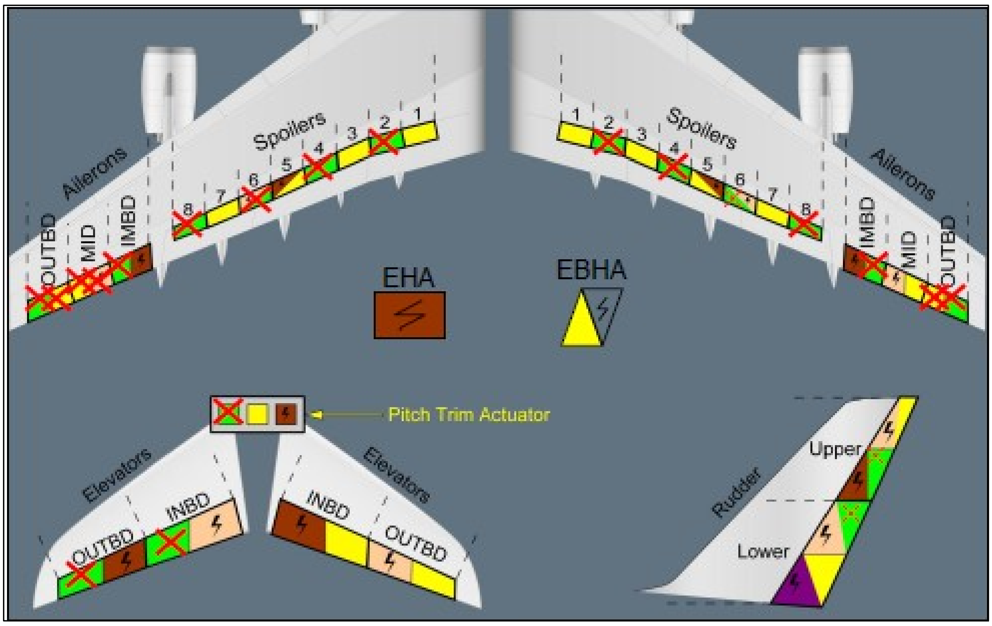

Image source: Airbus

[117] The servocontrols in the A380 were powered by the aircraft's Green or Yellow hydraulic systems and controlled electrically by the aircraft's flight control system.

Ailerons

Each wing had an outboard, mid and inboard aileron. Each aileron had two independent actuators per control surface for redundancy (see the previous discussion at section B.3.3.3 titled *Primary flight controls actuation systems*).

The loss of the Green hydraulic system and impact damage to the aircraft's wiring combined to affect the availability of the left aileron system for the remainder of the flight. As a result, the outboard aileron Yellow and Green servocontrols, mid aileron EHA and Yellow servocontrol and the inboard aileron Green servocontrol were unavailable. As a result, the left outer and mid ailerons were disabled. (Figure B58).

On the right wing, the inboard Green and both outboard Yellow and Green servocontrols were unavailable. As a result, the right outboard aileron was disabled. The aileron droop function of the remaining operative surfaces was not affected.[118]

Spoilers

Each wing had eight spoilers, numbered 1 to 8 outwards from the aircraft centreline **(**Figure B58). The loss of the Green hydraulic system and wiring damage resulted in the unavailability of spoilers 2, 4, 6 and 8 on the left wing and of spoilers 2, 4 and 8 on the right wing.

Elevators

Each horizontal stabiliser had an inboard and outboard elevator, each with two independent actuators for redundancy (Figure B58). Although the Green hydraulic system servocontrols were lost on the left inboard and outboard elevators, all elevators remained operable due to the availability of redundant actuators.

Trimmable horizontal stabiliser

Pitch trim operation was unaffected following the uncontained engine failure. The trimmable horizontal stabiliser had three independent actuators, numbered 1 to 3 from left to right, for redundancy (Figure B58). The loss of the Green hydraulic pressure rendered the No. 1 actuator inoperative, while actuators 2 and 3 continued normal operation.

Rudder

The vertical stabiliser had an upper and lower rudder, each with two EBHA actuators. These actuators were numbered lower/upper 1 and 2. Despite the loss of Green hydraulic system pressure, system redundancy meant that there was no loss of any rudder actuators.

Landing gear

The A380 landing gear comprised of the left and right wing landing gear, the left and right body landing gear and the nose gear (Figure B59).

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118 The droop function augments the high lift function of the slats and flaps during approach. When the flaps are extended all ailerons droop downwards to increase the camber of the wing.

Figure B59: A380 landing gear configuration

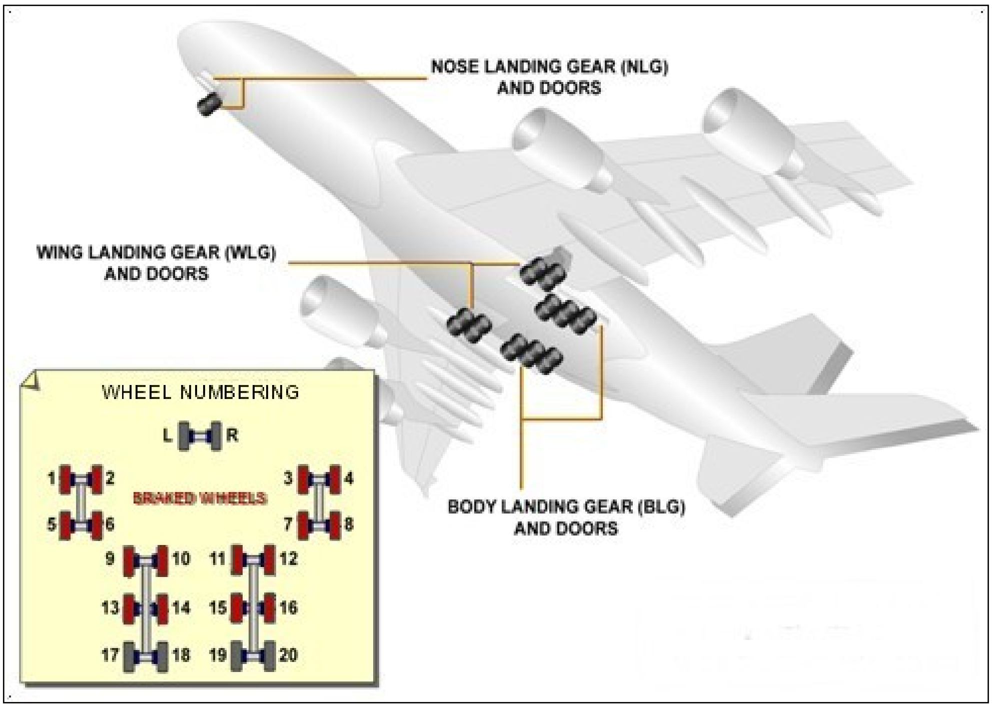

Image source: Airbus

Landing gear extension

The Green hydraulic system provided the hydraulic power supply for the nose and wing landing gear and their respective gear doors. The Yellow hydraulic system provided the hydraulic power supply for the body landing gear and their respective gear doors.

In the event that Yellow or Green hydraulic power was not available to extend the landing gear, the crew were required to perform a gravity extension procedure. The depressurisation of the Green hydraulic system necessitated the extension of the landing gear by the flight crew using this procedure.

Braking

All of the A380's wing landing gear wheels were equipped with hydraulically actuated brakes. Similarly, all of the wheels on the body landing gear (except for wheels 17 to 20) were equipped with hydraulically actuated brakes. Wheels 17 to 20 are used for body gear steering (Figure B59).

The A380's braking control system included reconfiguration logic, which was based on a philosophy of maintaining performance requirements despite associated equipment failures that result in changes to the braking mode. A number of braking modes were available in the A380, including NORMAL, ALTERNATE, EMERGENCY and ULTIMATE.[119] The logic took account of the availability of the respective operating modes.

Engine debris damage to the wiring looms affected the aircraft's braking system as follows:

- the body landing gear brakes remained in NORMAL mode
- the right wing landing gear brakes reconfigured to ALTERNATE mode but without anti-skid protection
- the left wing landing gear brakes lost pressure. As a result of this pressure loss there was no braking capability on the left wing landing gear.[120]

During the landing, the initially heavy braking by the crew was reduced as the landing roll progressed. Asymmetric braking during the landing resulted from the unavailability of the left wing landing gear brakes and caused the left body landing gear brakes to absorb significant energy. The maximum brake temperatures recorded on the left body landing gear, wheel 10 brake, exceeded 900 °C.

The left main body gear wheel fuse plugs[121] melted as brake temperature increased after the aircraft had come to a complete stop. As a result, the left body gear wheels 9, 10, 13 and 14 deflated.

Steering

The depressurisation of the Green hydraulic system set the nosewheel steering to alternate mode. In this mode, the nosewheel steering used the local electro-hydraulic generator system and there was no operational impact.

Bleed air

The bleed air system supplied high-pressure air for the following:

- air conditioning and cabin pressurisation
- wing and engine anti-ice
- engine starting
- hydraulic reservoir pressurisation.

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[119] Normal braking provided anti-skid protection and the crew were able to select the 'auto brake' function. There were no restrictions on the use of the system in normal mode. Alternate braking was similar to normal braking; however, hydraulic power was supplied by the local electro-hydraulic generation system. Emergency braking did not provide anti-skid protection and the auto brake function was not available in this mode. Braking pressure was reduced in the emergency mode. In the event that pedal braking was unavailable, ultimate braking to the BLG was available when the crew set the park brake handle to the on position. In the ultimate mode braking was automatically applied to the WLG when the ground spoilers were deployed.

[120] The display to the crew of the associated ECAM alert was inhibited by flight phase logic.

[121] Fuse plugs were designed to deflate wheels that were exposed to temperature beyond a specified limit.

The bleed air system received compressed air from the engines and distributed this air at the correct pressure, temperature and flow rate. This air could be sourced from the engines, the APU or from ground air sources.

An Overheat Detection System (ODS) protected and monitored the pneumatic system and was in continuous operation during flight. If a pneumatic system duct was damaged, the ODS would identify the location and isolate that area to prevent potential damage to the surrounding structure and components from the leaking hot air. The overheat detection loops were installed adjacent to the pneumatic hot air ducts in the fuselage, the belly, the forward cargo hold, the wing and the engine pylons.

The bleed air ducts in the left wing leading edge and centre fuselage were damaged from impact by the engine debris, affecting the distribution of bleed air from engines No. 1 and 2 and the APU (Figure 60). The damage was detected by the ODS and the affected pneumatic systems were isolated in less than 10 seconds. There was no operational impact.

Figure 60: Bleed air duct damage

Fuel system

The A380 carried fuel in 11 integral fuel tanks, including an outer, mid and inner tank in each wing. Two engine feed tanks per wing supplied fuel directly to their respective engines (for example, Feed Tank 2 supplied fuel exclusively to the No. 2 engine). A trim tank was also located in the horizontal stabilizer (Figure B61).

Figure B61: A380 fuel tanks

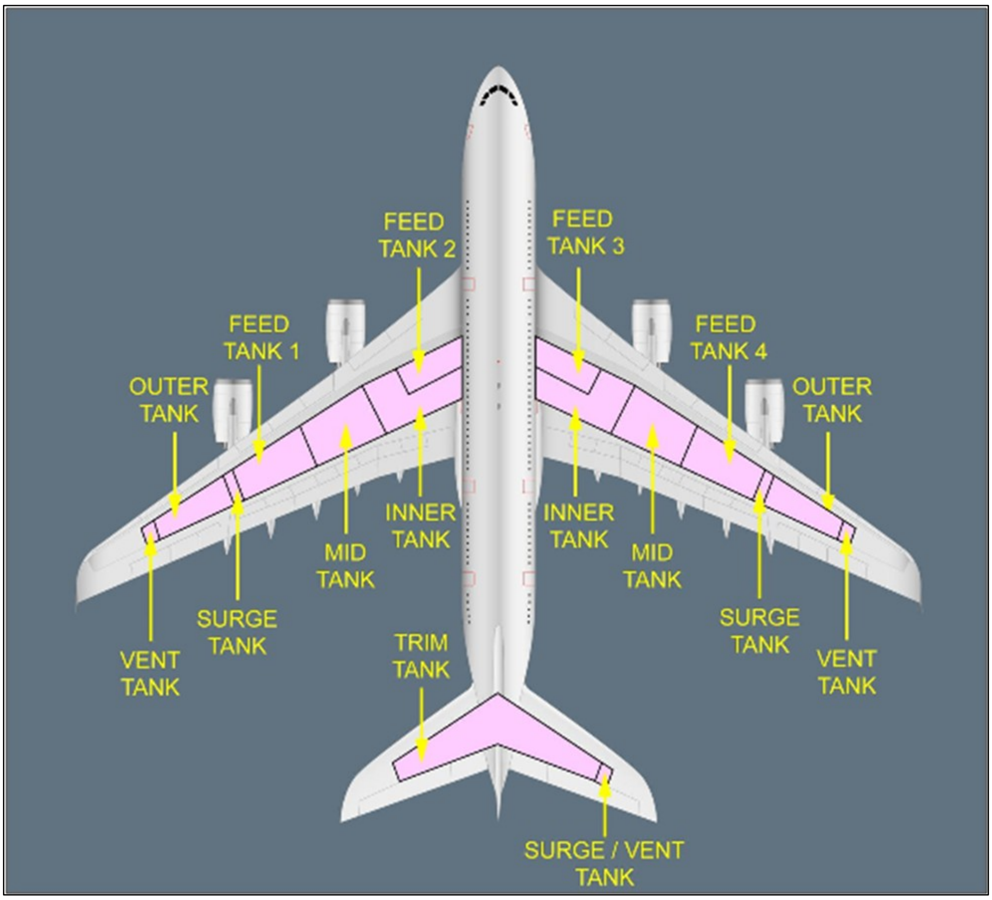

Image source: Airbus

The aircraft's fuel quantity management system (FQMS) automatically controlled and monitored the transfer of fuel within the fuel system. Manual control is also available to the flight crew. The FQMS operated to:

- supply fuel to the engines and APU
- maintain the aircraft's centre of gravity within limits
- reduce the structural loads on the aircraft through the load alleviation function
- control the refuelling and defueling of the aircraft
- enable fuel jettison if necessary.

In this accident, the operation of the fuel system was degraded as a result of the damaged wiring looms and the loss of the AC BUS 2. In addition, the Left Inner Tank (LIT) and Feed Tank 2 were punctured by engine debris, resulting in fuel leaking from that damage. The recorded data showed that Feed Tank 2 leaked about

5,000 kg of fuel in flight and a further 2,600 kg of fuel once on the ground. The fuel quantity records for the LIT did not show a marked reduction of fuel quantity following the uncontained engine failure, indicating that the hole in that tank was most likely located above the fuel level. However, evidence of the loss of anti-corrosion protection at the left wing front spar was consistent with some fuel leakage.

Damage to the fuel system resulted in:

- the FQMS being disabled, with the consequential loss of the fuel jettison function.
- fuel in the Left and Right Inner Tanks being unusable. The Right Inner Tank contained fuel and was physically capable of its transfer to a feed tank. However, system logic inhibited the manual transfer of fuel due to wiring damage to the LIT fuel pumps. The transfer inhibition was designed to prevent a fuel imbalance between wing tank pairs.
- the Left and Right Mid Tanks being unusable; however, these tanks were empty for the entire flight.
- the inability to transfer fuel from the Left Outer Tank using a manual Emergency Outer Tank Transfer function,[122] fuel was transferred from the Right Outer Tank while fuel remained trapped in the Left Outer Tank. This function was controlled without any logic to prevent the asymmetric operation of outer tank transfer valves.
- the restriction of the trim tank fuel transfer capability to the manual transfer of fuel to the inner feed tanks after the provision of a low fuel warning to the crew. Had the flight crew initiated this procedure, approximately half of the fuel in the trim tank would have been lost due to its transfer to the damaged Feed Tank 2.

Low pressure shut off valve

Low pressure shut-off valves (LPSOV) were installed in the pylon of each engine. Their main function was to isolate the engine from the hydraulic fluid supply (fuel and oil) in case of an engine fire. The LPSOVs were actuated by twin motors, one motor was powered by the normal electric route along the front spar, and the second motor was powered by a diverted route along the keel, entering the wing via the shroud box Figure B62). When commanded, a LPSOV isolates its engine from the fuel supply. Control of each LPSOV was through the associated:

- ENG MASTER switch, which opened and closed the valve
- engine FIRE PUSH switch, which closed the valve.

Severing of signal wires in the left wing leading edge and belly fairing of the aircraft contributed to the loss of control of the No. 1 engine LPSOV. This resulted in the inability to shut down that engine after the aircraft had landed at Changi Airport, Singapore. Although the leading edge wing damage was from major disc fragments, the damage to the belly fairing and its internal wiring was established to be from a smaller fragment.

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[122] The Emergency Outer Tank Transfer function allowed fuel to be transferred from an Outer Tank to the respective outboard feed tank. Left Outer Tank transferred to Feed Tank 1 and the Right Outer Tank transferred to the Feed Tank 4.

Figure B62: Schematic showing the trajectory of the disc fragments as they exited the No. 2 engine case and impacted the left wing and airframe. A smaller, high energy disc fragment penetrated the belly fairing and severed the LPSOV control wires for the No. 1 and No. 2 engines

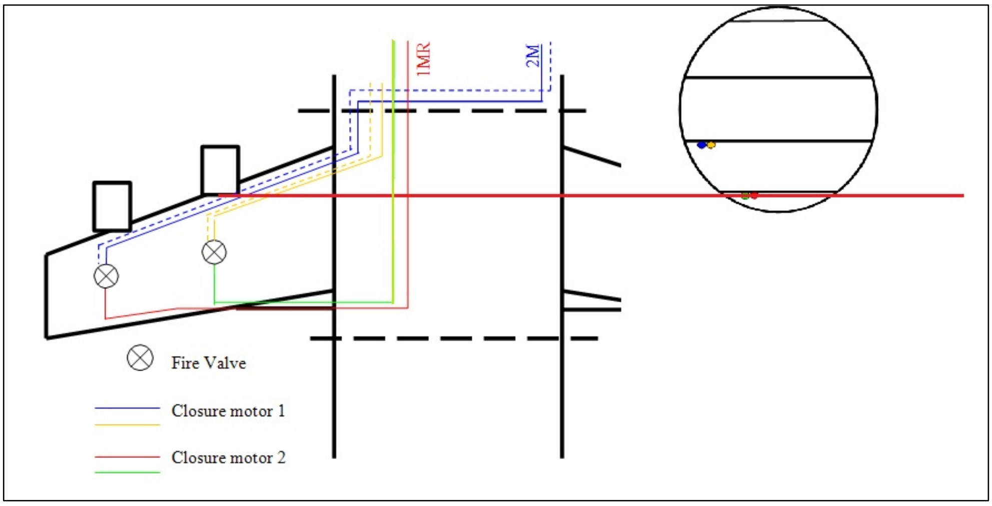


Image source: Airbus

Fire protection

The No. 2 engine fire push button was activated shortly after the crew were alerted to the uncontained engine failure. After the aircraft had landed back at Changi Airport, the crew commenced shutting down the remaining engines. Despite switching the final engine master switch to OFF, the crew were alerted by Airport Emergency Services (AES) that the No. 1 engine remained in idle operation. In response, the crew pressed the engine fire push-button in an attempt to shut down the still-running engine. That process was ineffective, with the No. 1 engine eventually being 'drowned' using retardant by AES.

Each engine was fitted with two fire extinguishing bottles installed within the pylon, and each bottle was controlled by two individual discharge cartridges (squibs). The design of the aircraft incorporated redundant wiring routes; one along the wing leading edge and the other routed along the centre of the fuselage and then along the wing trailing edge. Airbus reported that during their inspection of the fire extinguishing system, neither of the fire bottles for the No. 1 engine had discharged. Only one bottle had discharged for the No. 2 engine.

As detailed in Table B7, the physical state of the fire bottles and the extent of wiring damage to the fire extinguishing system was established. One of the bottles (Bottle-1) to the No. 2 engine had discharged while the second bottle (Bottle-2) remained full and had not discharged when commanded. Inspection of the bottles for the No. 1 engine revealed that both bottles were full and had not discharged when commanded.

Conductivity testing of the wiring looms for the fire extinguishing system revealed significant damage had occurred as a result of the disc burst. Many of the signal wires to the system had been severed. The signal wire to Bottle-1 Squib-1 from the No. 1 engine was found to be intact. That bottle should have discharged its contents as commanded when the crew attempted to shut down that engine. The investigation was unable to determine why that fire bottle did not function.

Table B7: Fire extinguishing system – bottle discharge status and the electrical continuity of the signal/control wires (whether they were intact or had been severed)

Engine No. 1	Bottle-1	Squib-1	Severed	Did not discharge
		Squib-2	Intact	
Engine No. 1	Bottle-2	Squib-1	Severed	Did not discharge
		Squib-2	Severed	
Engine No. 2	Bottle-1	Squib-1	Severed	Discharged
		Squib-2	Intact	
Engine No. 2	Bottle-2	Squib-1	Severed	Did not discharge
		Squib-2	Severed	

APPENDIX C: FLIGHT RECORDERS AND DATA

SUMMARY

The aircraft was fitted with crash-protected flight recording systems and non-mandatory data recording systems. The engine monitoring and control units also had a recording capability. Recorded data from all of these sources was utilised by the Australian Transport Safety Bureau (ATSB) to determine a sequence of events and understand the engine failure mechanisms.

The recorded data showed that the aircraft departed Changi Airport, Singapore, runway 20C, at 01:56:47 UTC on 4 November 2010. Approximately 3½ minutes later, during climb through 5,000 ft, some No. 2 engine air and oil parameter values began to gradually diverge from the values shown by the other engines. At 02:00:58, the No. 2 engine rotor speeds, temperatures and vibrations started to fluctuate, followed by rapid increases in turbine cooling air temperatures. No. 2 engine temperature and speed exceedances resulted in a *Master Warning* activation at 02:01:09. Two seconds later at 7,250 ft, the intermediate-pressure (IP) turbine disc burst and was liberated from the aircraft. Fault indications from many aircraft systems were recorded subsequently, including electrical power, flight controls, hydraulics, brakes and bleed air.

Recorded data showed that after the failure, the aircraft levelled at 7,400 ft. Over a 70-minute period, it completed nine orbits over the ocean before commencing a descent into Singapore. During the descent, the flaps extended at half the normal rate, and two *Low Energy* warnings were recorded during the final approach. A stall warning sounded just before touchdown on runway 20C at 03:46:47.

The aircraft remained on runway 20C with the right engines (No. 3 and No. 4) being shut down by 03:49:08, shortly after the aircraft came to a stop. The No. 1 engine was unable to be shut down immediately. All passengers had disembarked by 05:40:05. The flight recordings ceased about 20 minutes later, with the No. 1 engine still running. Fuel loss on the ground at Singapore was recorded as 2,600 kg, as a result of a fuel leak from the aircraft's No. 2 feed tank.

During the investigation, historical recorded data obtained from the operator was utilised. Maximum engine pressures experienced during previous service (especially on takeoff) were used to determine the limitations on the return-to-service of the operator's other A380 aircraft.

FACTUAL INFORMATION

Recorded flight data

The aircraft was fitted with both crash-protected (mandatory) and supplementary (non-mandatory) data recording systems. The aircraft system, engine monitoring and control units also had a recording capability. Recorded data from all of these sources was utilised during the investigation.

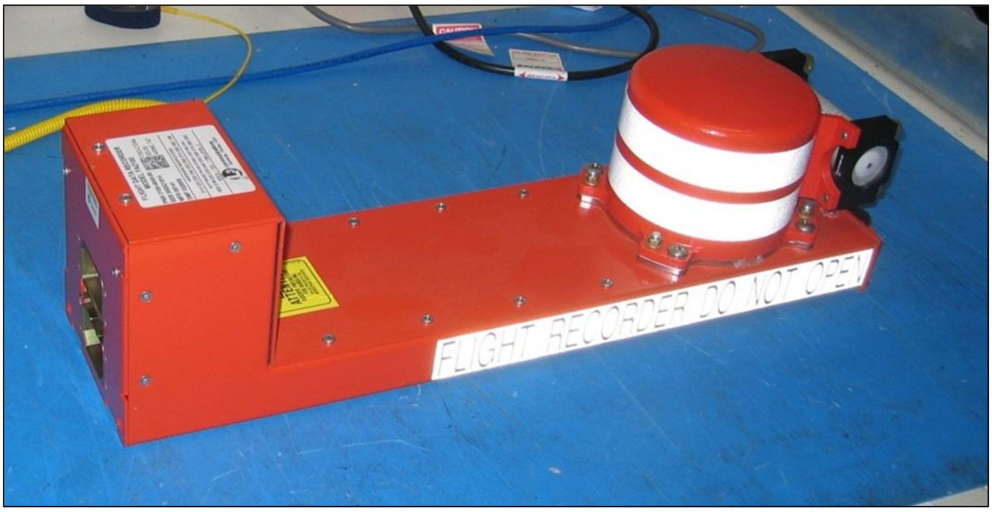
Crash-protected flight recorders

As required by Australian legislation and international certification standards, the aircraft was fitted with two crash-protected flight recorders; a flight data recorder (FDR) and a cockpit voice recorder (CVR).

Flight data recorder

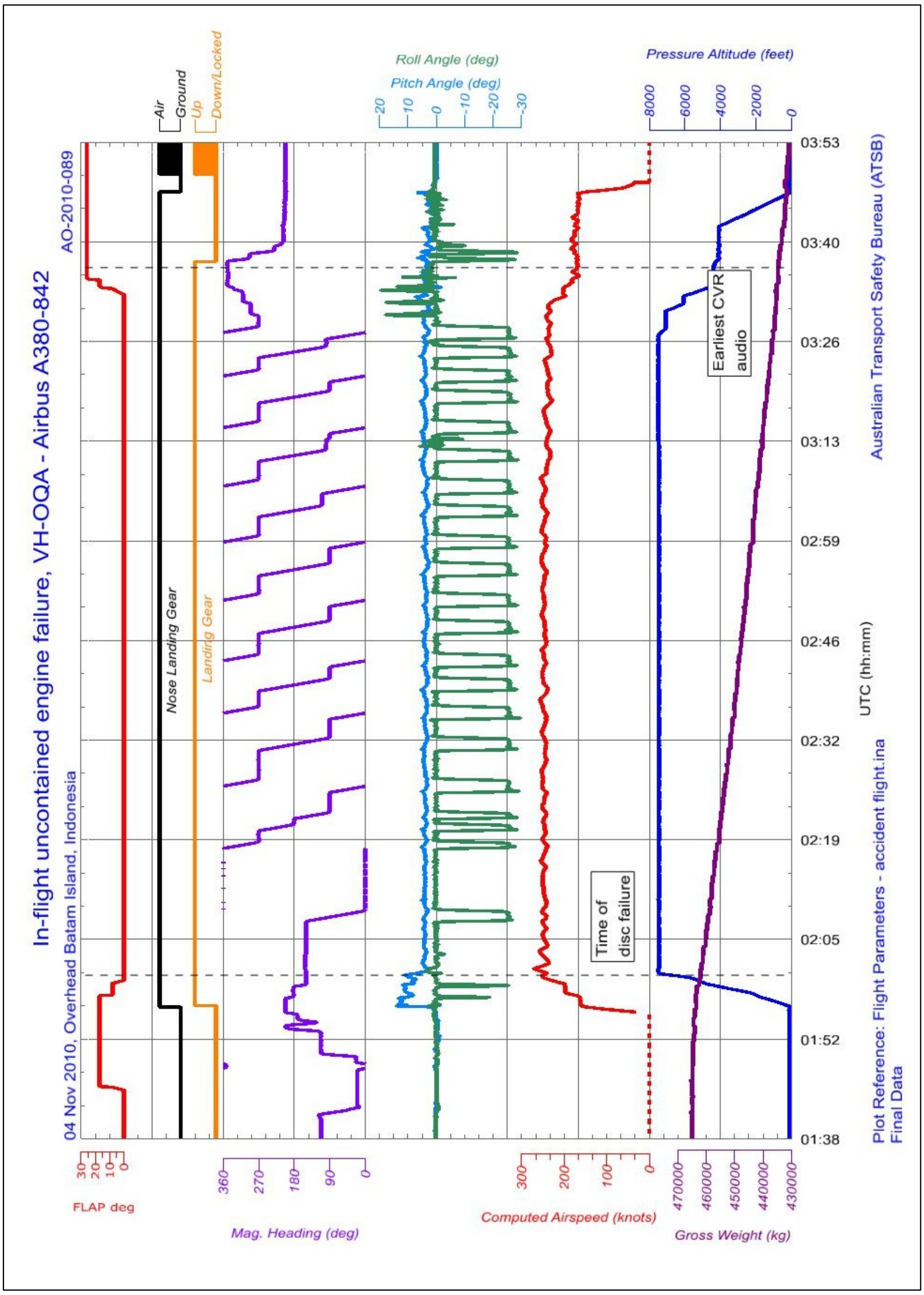
The FDR was an L3 Communications Model FA2100 solid state flight data recorder (Figure C1). The FDR was required to record as a minimum, the previous 25 hours of flight data. The FDR was removed from the aircraft under the supervision of the Air Accident Investigation Bureau (AAIB) of Singapore and transported to the aircraft operator's Sydney base. The ATSB supervised the download of the recorded data from the FDR on 5 November 2010.

Figure C1: Flight Data Recorder



The FDR was undamaged and contained 77 hours 16 minutes of recorded aircraft operation. The data was of excellent quality and included the accident flight and five previous flights. The FDR recorded over 2,000 aircraft and flight parameters (Figure C2).

Figure C2: Occurrence flight summary using selected FDR parameters



Cockpit voice recorder

The CVR was an L3 Communications Model FA2100 solid state cockpit voice recorder (Figure C3). The CVR was manufactured to record, as a minimum, the previous 2 hours of flight crew conversation and the cockpit aural environment. The CVR was removed from the aircraft and returned directly to the ATSB's technical facilities in Canberra for download and analysis.

Figure C3: Cockpit Voice Recorder

The CVR was undamaged and contained over 2 hours of excellent quality cockpit audio, comprising separate channels for the captain, first officer, observer position[123] and cockpit area microphone. Due to the continued operation of the No. 1 engine after the aircraft landed, electrical power remained available and the CVR continued operating, overwriting the audio recorded at the time of the uncontained engine failure. The CVR system is designed so that the recording unit stops when power is removed, either 5 minutes after the last engine is shut down or by manually operating the CVR circuit breaker.

The recorded CVR audio commenced during the aircraft's descent in the landing approach and included touchdown and the subsequent ground operations. As a consequence, the audio associated with the engine failure and initial flight crew response was overwritten. The CVR contained pertinent information regarding the approach, landing and roll-out. Of particular value to the ATSB were the communications between the flight crew, ATC and emergency services following the aircraft coming to a stop, communication by the flight crew with the cabin crew and passengers and the coordination with ground services during passenger disembarkation, as well as the attempts to shut down the No. 1 engine. A copy of the CVR audio was made and retained by the ATSB before the recorder was returned to the operator.

Aircraft operations and flight path

Information was obtained from the FDR data regarding the flight and engine operations leading up to the engine failure. The recorded data also provided information on aircraft and system performance leading up to, and following, the engine failure.

Following the engine failure, the aircraft levelled off at 7,400 ft and continued on track. The aircraft then made a left turn and flew over Bintan Island, Indonesia. Nine left orbits were then conducted over the ocean north of Bintan Island (Figure C4). Approximately 90 minutes after the engine failure, the aircraft commenced an approach and descent into Changi Airport. The CVR audio

[123] The public address (PA) audio was recorded on the channel of the crew that initiated the PA call.

commenced during the descent after the flaps and before the landing gear were extended.

Figure C4: Aircraft flight path

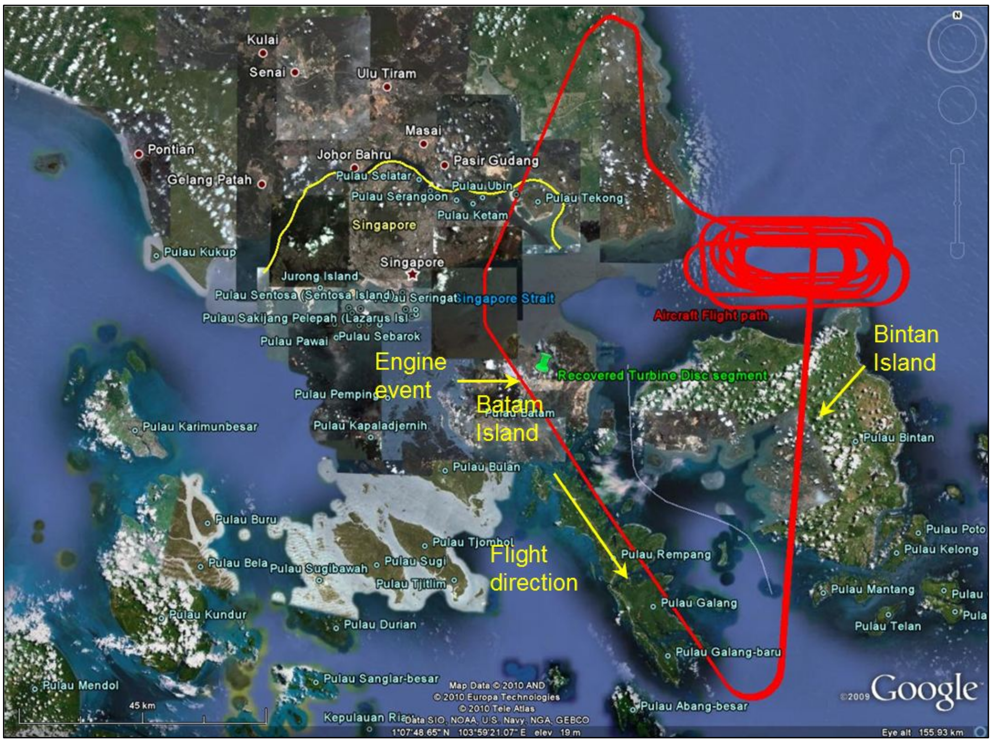


Image source: Google Earth

Non-mandatory data recording systems

The operator had incorporated a number of supplementary data recording systems in its fleet of A380 aircraft. Those recording systems provided additional engine data (above that available from the FDR) that facilitated an early analysis of the specific engine failure mechanism.

The operator used a number of other devices to store information that originated from the Aircraft Condition Monitoring System (ACMS) and the Centralised Maintenance System (CMS). That information differed from that captured by the mandatory, crash-protected FDR, because the ACMS programming was customised to provide enhanced flight and systems monitoring information.

Wireless digital ACMS recorder

The wireless digital ACMS recorder (WDAR) was a Teledyne P/N 2243800-81 unit (Figure C5) that stored a continuous record of over 1,000 different aircraft and engine system parameters. These parameters were stored in a customised format that was defined by the operator in conjunction with the aircraft and engine manufacturers.

The WDAR contained additional aircraft system and engine parameters that were recorded at a greater resolution and sampling rate than the FDR.

The WDAR's normal mode of operation was to store the ACMS information on an internal memory device, and upon aircraft arrival in an Australian port (or when remotely triggered) to transmit recorded data to the operator using available cellular telephone networks.

Figure C5: Wireless DAR unit

Aircraft Network Server Units

The Airbus A380 uses computer network architecture to monitor and control the aircraft's systems. The aircraft was fitted with main and backup aircraft network server units (ANSU-OPS1 and ANSU-OPS2), which contained duplicate recorded information sets from the ACMS and CMS.

Duplicate Flight Data/Digital ACMS Recording

During normal aircraft operations, the recorded information stored in the two ANSU-OPS units was fully redundant (that is, identical).

Each ANSU provided a virtual quick access recorder (VQAR) functionality, in which a complete copy of the FDR information was stored. In a similar way, a copy of the DAR information was stored on each ANSU, as a virtual digital ACMS recorder (VDAR). The VDAR normally stored the same information as the separate WDAR unit (Figure C6).

Figure C6: Schematic of the data acquisition and recording system

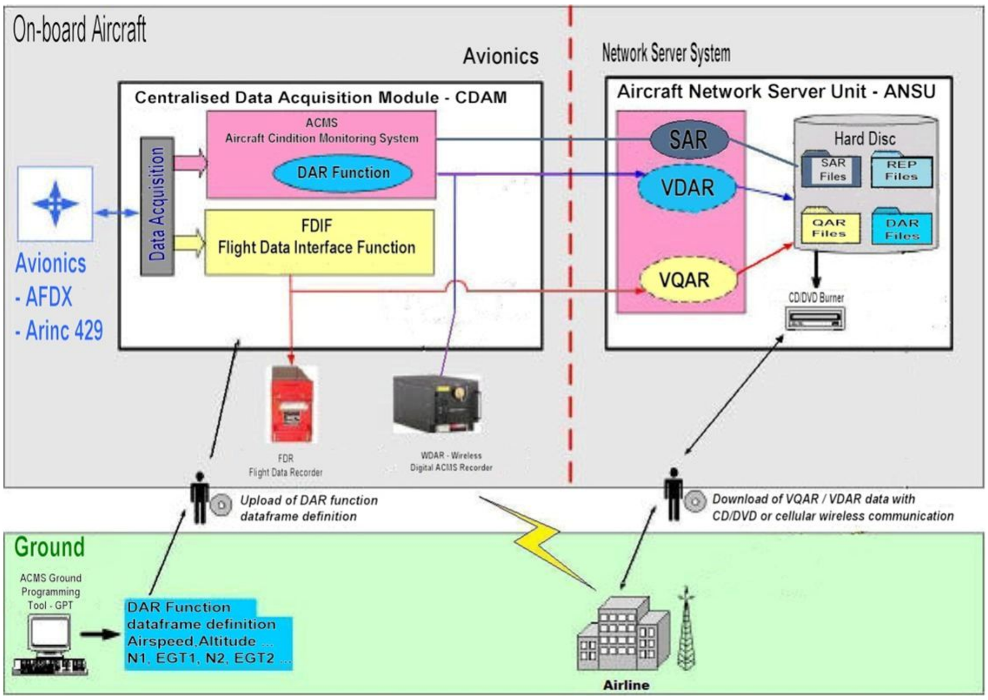


Image source: Qantas

Smartaircraft condition monitoring system recorder

The Smartaircraft condition monitoring system recorder (SACMSR) provided a high capacity (defined by the number of parameters recorded and their sampling rate), continuous recording of up to 256 parameters around pre-defined trigger events. The SACMSR parameter information for the engine failure provided high-resolution (8 Hz) engine parameters for 180 seconds leading up to the uncontained engine failure, and a further 180 seconds following the occurrence.

Aircraft system reports

Aircraft system reports (REP) were also recorded in ANSU-OPS1 and 2.

The REP folder contained data that may originate from an aircraft system or be calculated according to pre-defined logic. Up to 1,000 different reports may be defined, and can be transmitted from the aircraft via the aircraft communication addressing and reporting system (ACARS).

An example of information provided by a REP was a snapshot of brake temperatures for all wheels following the touchdown (Figure C7).

Figure C7: REP 208 from ANSU-OPS1

```
    ACID     ACTYP     ENG | REP  CODE  FMT  CNT   | DATE        UTC       FP
H01 .VH-OQA  A380-800   RR | 208  2081   82  00308 | 04.NOV.10   03:49:05  12

    FROM TO  FLT           | SW     AI-DB  DATABASE   | MOD
H02          ----------    | S0385  S0385  AWC110909A | 00.XXX.00

H03 REASON: Snapshot of brake temperatures for phase 12

                BRAKE TEMPERATURE FOR WHEEL 1 to 16

            BRTEMP.1    BRTEMP.2    BRTEMP.3    BRTEMP.4
              31          32          350         530

            BRTEMP.5    BRTEMP.6    BRTEMP.7    BRTEMP.8
              40          37          534         523

            BRTEMP.9    BRTEMP.10   BRTEMP.11   BRTEMP.12
              831         1000        469         421

            BRTEMP.13   BRTEMP.14   BRTEMP.15   BRTEMP.16
              858         831         547         448

            TAT :  28.9
```

Image source: Airbus

Centralised Maintenance System

The ANSU-OPS also stored information from the CMS including the post flight report, fault message and ECAM warning histories. This information provided the ATSB with a timeline of fault messages generated from the engine failure and subsequently displayed to the flight crew (Figure C8).

Figure C8: ECAM warning data around the time of engine failure from the CMS recorded on ANSU-OPS

FAULT_CODE	TITLE	SINCE_TIME	UNTIL_TIME	STATUS	ALERT_LEVEL	ALERT_TYPE
3160G184	Amber crosses nac temp eng3	04/11/2010 01:42:48	04/11/2010 01:45:29	DISPLAYED	UNKNOWN	UNKNOWN
3160G185	Amber crosses nac temp eng4	04/11/2010 01:42:48	04/11/2010 01:45:29	DISPLAYED	UNKNOWN	UNKNOWN
3160G162	Red line oil pressure eng1	04/11/2010 01:42:50	04/11/2010 01:43:43	DISPLAYED	UNKNOWN	UNKNOWN
3160G164	Red line oil pressure eng3	04/11/2010 01:42:50	04/11/2010 01:44:53	DISPLAYED	UNKNOWN	UNKNOWN
3160G165	Red line oil pressure eng4	04/11/2010 01:42:50	04/11/2010 01:44:55	DISPLAYED	UNKNOWN	UNKNOWN
7722W120	ENG 2 TURBINE OVHT	04/11/2010 02:01:08	04/11/2010 02:01:08	CALCULATED	WARNING	INDEPENDENT
7722W120	ENG 2 TURBINE OVHT	04/11/2010 02:01:08	04/11/2010 02:01:30	DISPLAYED	WARNING	INDEPENDENT
7100W420	ENG 2 STALL	04/11/2010 02:01:09	04/11/2010 02:01:09	CALCULATED	CAUTION	INDEPENDENT
7100W420	ENG 2 STALL	04/11/2010 02:01:09	04/11/2010 02:01:17	DISPLAYED	CAUTION	INDEPENDENT
7720W120	ENG 2 EGT OVER LIMIT	04/11/2010 02:01:09	04/11/2010 02:01:09	CALCULATED	CAUTION	INDEPENDENT
7720W120	ENG 2 EGT OVER LIMIT	04/11/2010 02:01:09	04/11/2010 02:01:13	DISPLAYED	CAUTION	INDEPENDENT
7932W220	ENG 2 OIL TEMP HI	04/11/2010 02:01:09	04/11/2010 02:01:17	DISPLAYED	CAUTION	INDEPENDENT
7932W220	ENG 2 OIL TEMP HI	04/11/2010 02:01:09	04/11/2010 02:01:09	CALCULATED	CAUTION	INDEPENDENT
2230W040	AUTO FLT A/THR OFF	04/11/2010 02:01:13	04/11/2010 02:01:13	CALCULATED	CAUTION	INDEPENDENT
2230W040	AUTO FLT A/THR OFF	04/11/2010 02:01:13	04/11/2010 02:01:30	DISPLAYED	CAUTION	INDEPENDENT
2780N030	SLATS	04/11/2010 02:01:13	04/11/2010 02:01:13	CALCULATED	OTHERWISE	INOP
2780N030	SLATS	04/11/2010 02:01:13	04/11/2010 02:01:30	DISPLAYED	OTHERWISE	INOP
2780W030	F/CTL SLAT SYS 1+2 FAULT	04/11/2010 02:01:13	04/11/2010 02:01:13	CALCULATED	CAUTION	INDEPENDENT
2780W030	F/CTL SLAT SYS 1+2 FAULT	04/11/2010 02:01:13	04/11/2010 02:01:30	DISPLAYED	CAUTION	INDEPENDENT
2910N270	Y ENG 4 PMP A+B	04/11/2010 02:01:13	04/11/2010 02:01:13	CALCULATED	OTHERWISE	INOP
2910N270	Y ENG 4 PMP A+B	04/11/2010 02:01:13	04/11/2010 02:01:30	DISPLAYED	OTHERWISE	INOP

Image source: Airbus

Recovery of the non-mandatory recorded information

The WDAR was downloaded by the operator under ATSB supervision using another of the operator's A380 aircraft. The occurrence flight recording was incomplete, ending about 50 seconds prior to the engine failure. Examination of the WDAR revealed that the recorder operation ceased as a result of disruption to the electrical system during the occurrence, and that some data was lost due to internal buffering.

On 10 November 2010, ANSU-OPS2 was downloaded in Singapore with assistance from the aircraft and unit manufacturers. When the recovered VDAR data was examined, it was found that the recording had been interrupted and ceased prior to the event. However, a further 14 seconds of VDAR information was available before the recording terminated. The additional information obtained from the VDAR was able to clarify the understanding of the engine failure sequence. SACMSR, REP and CMS files recorded on ANSU-OPS 2 were also incomplete due to the power disruption.

The ANSU-OPS1 was downloaded on 12 November 2010. Upon examination, it was evident that the complete history of the occurrence flight, including the engine failure had been retained by the VDAR, SACMSR and REP folders, with no premature recording termination and data loss. The CMS history data was also complete. The data downloaded from ANSU-OPS1 was provided to all parties to the investigation.

Engine recorded data

The electronic engine controller (EEC) and engine monitor unit (EMU) fitted to each engine contained non-volatile recordings that were able to be accessed by the engine manufacturer .The EEC data contained a history of fault message information from the flight and a snapshot of a relevant data-set at the time each fault was registered (once-per-second data recording) (Figure C9). Upon download and examination, it was evident that fault messages had been recorded for the No. 2 engine for up to 12 minutes after the engine failure.

Figure C9: No. 2 engine EEC data around time of engine failure

	A	B	C	D	E	F	G	H	I	J	K	
1	ID	Date	Time	FWS	Class	Message	FM	Side	Status	Priority	FDCE	Comp_Data
2	1	4-Nov	01:44:08		5	N1Turbine Speed Probe 2	461	2	Detected Failure	High	FWS	0005000003021C0A2532051903670008156451CD2042000000001000
3	1	4-Nov	01:44:08		5	N1Turbine Speed Probe 2	461	3	Detected Failure	High	FWS	0005000003021C0A2532051903670008156461CD2042000000001000
4	3	4-Nov	02:01:07	ENG x TURBINE OVHT	1	Propulsion System - Turbine Ovht	436	2	Detected Failure	High	FWS	0001112A47485D565C614D192D1D2F5E156451B440420000050100
5	3	4-Nov	02:01:07	ENG x TURBINE OVHT	1	Propulsion System - Turbine Ovht	436	3	Detected Failure	High	FWS	0001112A47485D565C614D1A2D1D2F5E156461B440420000050100
6	6	4-Nov	02:01:08	ENG x OIL TEMP HI	1	Oil System - High Oil Temp	434	2	Detected Failure	High	FWS	0001112A0F483936635F0F190B672F5E156451B2404200006001000
7	6	4-Nov	02:01:08	ENG x OIL TEMP HI	1	Oil System - High Oil Temp	434	3	Detected Failure	High	FWS	0001112A10483B3B615F10190A672F5E156461B2404200006001000
8	2	4-Nov	02:01:08	ENG x STALL	1	Propulsion System - Surge	435	2	Detected Failure	High	FWS	0001112A26484B555C622919294E2F5E156451B3404214C08000458
9	2	4-Nov	02:01:08	ENG x STALL	1	Propulsion System - Surge	435	3	Detected Failure	High	FWS	0001112A26484B555C6229192D4E2F5E156461B340420634800045
10	4	4-Nov	02:01:08	EGT Amber Line 31	1	Propulsion System - EGT Amber	480	2	Detected Failure	High	CDS	0001112A0E48352F635F0E180B672F5E156451E0404200000701000
11	4	4-Nov	02:01:08	EGT Amber Line 31	1	Propulsion System - EGT Amber	480	3	Detected Failure	High	CDS	0001112A0E48352F635F0E180B672F5E156461E0404200000701000
12	5	4-Nov	02:01:09	ENG x EGT OVER LIMIT	1	Propulsion System - EGT Over Limit	428	2	Detected Failure	High	FWS	0001112A0D483128635F0D180A672F5E156451AC40420000470100
13	5	4-Nov	02:01:09	ENG x EGT OVER LIMIT	1	Propulsion System - EGT Over Limit	428	3	Detected Failure	High	FWS	0001112A0D483128635F0D180A672F5E156461AC40420000470100
14	7	4-Nov	02:01:12	ENG x MINOR FAULT	1	EIPM	2094	2	Detected Failure	Low	FWS	0002122A0648251C635F061605672F561564582E20429244010042B
15	8	4-Nov	02:01:12		4	EIPM	2095	2	Detected Failure	High	FWS	0003122A0648241B625E061705672F561564582F2042FBC2010042E
16	8	4-Nov	02:03:25		5	TCA	2077	3	Detected Failure	High	FWS	0005122A01020913281F021401520011156468 1D204200000001810
17	9	NCD	02:03:25		5	TCA	2077	2	Detected Failure	High	FWS	0005122B010207133100011500520011FFFF581D203281084007000
18	7	4-Nov	02:03:25	ENG x MINOR FAULT	1	EIPM	2094	3	Detected Failure	Low	FWS	0002122A01020913281F0214015200111564682E20424729010041F
19	9	4-Nov	02:03:27	ENG x SENSOR FAULT	1	P20T20 Probe	2098	3	Detected Failure	High	FWS	0002122B01020A13291D02180152001115646832204 28E084300000
20	10	4-Nov	02:03:29	ENG x TURBINE OVHT	1	Propulsion System - Turbine Ovht	436	3	Detected Failure	High	FWS	000112250 1020A13291B0218005200101564 61B4204200000001000
21	14	4-Nov	02:03:29		5	EEC	2115	3	Detected Failure	High	FWS	0005122A01020913281F0214015200111564684320428A1143008A
22	12	NCD	02:03:29		5	EEC	2115	2	Detected Failure	High	FWS	0005122B010207133100011500520011FFFF584320328A1143008A
23	15	4-Nov	02:03:29		5	EEC	2116	3	Detected Failure	High	FWS	0005122A01020913281F0214015200111564684420428A1143008A
24	13	NCD	02:03:29		5	EEC	2116	2	Detected Failure	High	FWS	0005132C010207133100011701520011FFFF584420328A1143008A
25	11	4-Nov	02:03:29	ENG x REV MINOR FAULT (Engine positions 2 & 3)	1	CH. A Proximity Sensor of Tertiary Lock	3106	3	Detected Failure	High	FWS	0002122B01020A13291D021601520011 15646C22204284018300840
26	10	NCD	02:03:29	ENG x REV MINOR FAULT (Engine positions 2 & 3)	1	CH. A Proximity Sensor of Tertiary Lock	3106	2	Detected Failure	High	FWS	0002132C010207133100014A01520011FFFF5C22203284048300840
27	12	4-Nov	02:03:29	ENG x REV MINOR FAULT (Engine positions 2 & 3)	1	CH. B Proximity Sensor of Tertiary Lock	3107	3	Detected Failure	High	FWS	0002122B01020A13291D0216015200111 5646C2320428404830084
28	11	NCD	02:03:29	ENG x REV MINOR FAULT (Engine positions 2 & 3)	1	CH. B Proximity Sensor of Tertiary Lock	3107	2	Detected Failure	High	FWS	0002132C010207133100014A01520011FFFF5C23203284048300840
29	13	4-Nov	02:03:31	ENG x REV INOP	1	ETRAC	3121	3	Detected Failure	High	FWS	0001122B01020A13291D02160152001115646C312042010004000 1

EEC2_CSV

Image source: Rolls-Royce plc

The EMU provided continuous, high-resolution engine data (5 Hz) of about 100 specific engine parameters. The No. 2 engine EMU data was available for the flight duration, until 3.4 seconds after the engine failure. The EEC and EMU data accessed by the engine manufacturer was provided to all relevant parties to the investigation.

Use of historical WDAR data

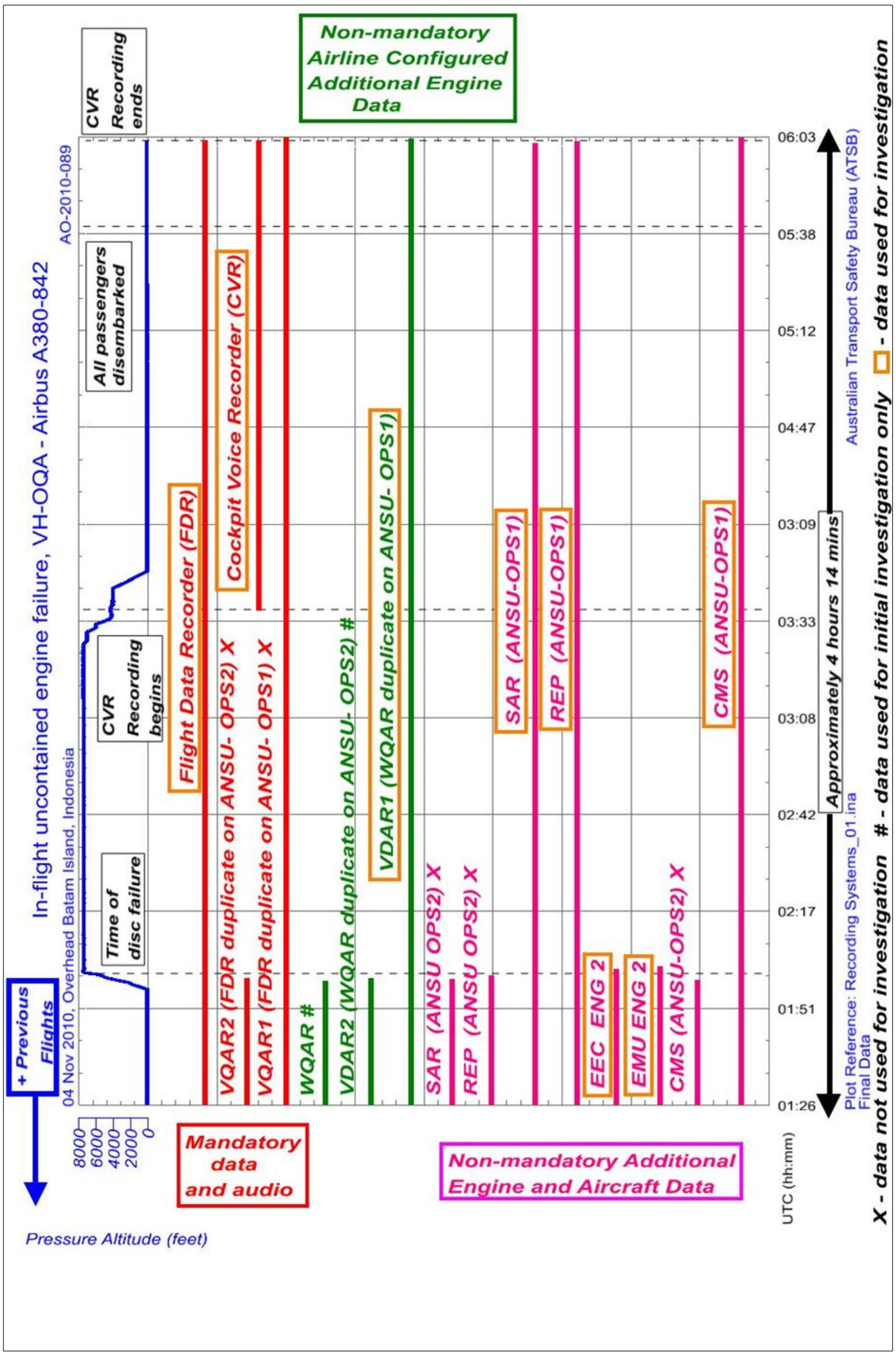
During the investigation, WDAR data from aircraft's previous flights was requested from the operator. This data was used to examine engine parameters during takeoff from various airports. Examination of the maximum high pressure (HP) compressor delivery pressure (P30) achieved during these takeoffs was used by the Civil Aviation Safety Authority (CASA) to determine the limitations for the return to service of the operator's other A380 aircraft following the voluntary grounding of their fleet.

Analysis

Recorded data analysis

The analysis of data downloaded from the aircraft's mandatory and non-mandatory recording devices provided the ATSB with a concise sequence of events. Recorded engine and aircraft parameters from multiple sources were combined to provide a sequence leading up to, and following, the No. 2 engine intermediate-pressure turbine (IP turbine) disc liberation. A summary of data available to the investigation and used is shown in (Figure C10).

Figure C10: Summary of recorded data



Sequence of events leading up to and relating to the IP turbine disc liberation

Engine parameters from the various sources of available data[124] were integrated to define a sequence of events leading up to and shortly after the IP turbine disc liberation. The CVR audio was not available for this period.

Table C1 tabulates the graphical representations of the engine parameters shown in Figure C11 and Figure C12.

Table C1: Sequence of events leading up to the No. 2 engine IP turbine disc liberation

UTC[125] (hh:mm:ss)	Event	Comment
01:43:24 01:44:36 01:44:39	No. 1 and No. 2 engine starts No. 3 engine starts No. 4 engine starts	Aircraft gross weight 464.8 t at Singapore. Engine and Warning Display (THR, N1, EGT, thrust mode and thrust rating limit) and ENG SD Page displayed to flight crew (N2, N3, fuel flow, oil quantity, oil pressure, oil temperature, N1, N2, N3 vibrations and nacelle temperature)
01:55:00	Aircraft commences takeoff on runway 20C	No. 2 Eng. N1[126] 79%, flex temp 59 °C
01:56:47	Airborne at Changi Airport,	No. 2 Eng. Turbofan Power Ratio (TPR) 72%. No. 2 Engine Oil Pressure 8% lower than average of four engines No. 2 Eng. N1 78-79%
01:57:55	System Display page automatically changes to cruise page at 1,500 ft.[127]	Engine indications displayed to flight crew are now Engine and Warning Display (THR, N1, EGT, Thrust mode and Thrust Rating Limit) plus fuel flow on CRUISE SD page
01:58:07 - 01:58:20	Engine thrust increased	No. 2 Engine Turbofan Power Ratio (TPR) increased to 90%, No. 2 Engine N1 86%
02:00:07	No. 2 engine TCAR[128] starts to increase	TCAR not displayed to flight crew. Indicative of oil feed stub pipe breach, passing barometric-corrected altitude of 4,980 ft. No. 2 Engine N1 87% TCAF = 557°°C, TCAR = 480 °C

[124] Refer to (Figure C7) for sources used from the aircraft.

[125] Timing from FDR data.

[126] N1 – Engine low pressure (LP) assembly shaft speed – measured as a % rpm of a reference speed (LP reference speed (100%) is 2,900 RPM for an RB211-Trent 972-84 engine. N1 max continued 97.2% max takeoff (5 minute limit) 97.2%. Source: EASA Type-Certificate Data Sheet E.012).

[127] The Cruise Page includes engine fuel flow information in addition to Engine and Warning Display parameters (THR, N1, EGT, Thrust mode and Thrust Rating Limit)

UTC[125] (hh:mm:ss)	Event	Comment
02:00:15	No. 2 engine oil temperature starts to increase	Oil temperature not displayed to flight crew at this time
02:00:16	No. 2 engine oil pressure starts to decrease	Oil pressure not displayed to flight crew at this time
02:00:22	No. 2 engine oil temperature and pressure values begin to diverge from the recorded values for the other engines	No. 2 engine oil temperature increasing from 179 °C No. 2 engine oil pressure decreasing from 68 psi Altitude 5,330 ft No. 2 engine N1 speed 87.4% Vibration not displayed to flight crew at this time
02:00:58	No. 2 engine N3 vibration[129] increases No. 2 engine N3[130] fluctuation	Altitude 6,620 ft No. 2 engine N1 87.4%
02:01:00-02:01:01	No. 2 engine N1 and N2 [131] starts to decrease No. 2 engine N3 starts to increase Rapid increase in No. 2 engine TCAR and TCAF commences	N1 from 87.6% to 86.6%, N2 from 94.5% to 93.3% N3 from 94.3% to 96.7% TCAF = 572 °C, TCAR = 535 °C N2, N3 not displayed to flight crew at this time. TCAF and TCAR not displayed to flight crew
02:01:06	No. 2 engine N3 on limit exceedance message	TCAF = 821 °C, TCAR = 649 °C N1 at 86.8%, N3 at 97.7%
02:01:07	No. 2 engine N3 reaches 98.4% and starts to decrease No. 2 engine fuel flow decreases Rapid No. 2 engine N1, N2 and P30[132]decrease No. 2 engine TGT rise	Indicative time of drive arm separation (turbine disc separation from the shaft)

[128] TCAR – Turbine Cooling Air Rear – cooling air temperature measured at the rear of the IPT.

[129] N3 vibration is the vibration level of the high-pressure (HP) spool (comprising the HP compressor and HP turbine). Similarly, N1 vibration is the vibration level of the LP spool (comprising the LP compressor and LP turbine). N2 vibration is the vibration level of the IP spool (comprising the IP compressor and IP turbine).

[130] N3 – Engine HP rotor speed – measured as a % rpm of a reference speed (HP spool reference speed (100%) is 12,200 RPM for an RB211-Trent 972-84 engine). N3 max continued 97.7% max takeoff (5 minute limit) 97.8%. Source: EASA Type-Certificate Data Sheet E.012).

[131] N2 – Engine intermediate pressure (IP) rotor speed – measured as a % rpm of a reference speed (IP reference speed (100%) is 8,300 RPM for an RB211-Trent 972-84 engine. N2 max continued 97.8% max takeoff (5 minute limit) 98.7%. Source: EASA Type-Certificate Data Sheet E.012).

[132] P30 – HP compressor delivery pressure.

UTC[125] (hh:mm:ss)	Event	Comment
02:01:08	No. 2 engine turbine overheat parameter activates and oil temp over limit message	TCAF = 840 °C, TCAR = 611 °C
02:01:09	Master Warning and Master Caution activate No. 2 engine core surge exceedance message	
02:01:10	No. 2 engine oil temp over limit	
02:01:11	No. 2 engine N2 to 40% and oil pressure starts to drop Loss of No. 2 engine TGT and TCAR signal No. 2 engine P30 surge Fault indications commence from multiple systems	Indicative of time of turbine disc failure Altitude 7,250 ft TGT = 1,018 °C

Figure C11: Graphical representation of selected engine parameters leading up to the engine failure (30 seconds)

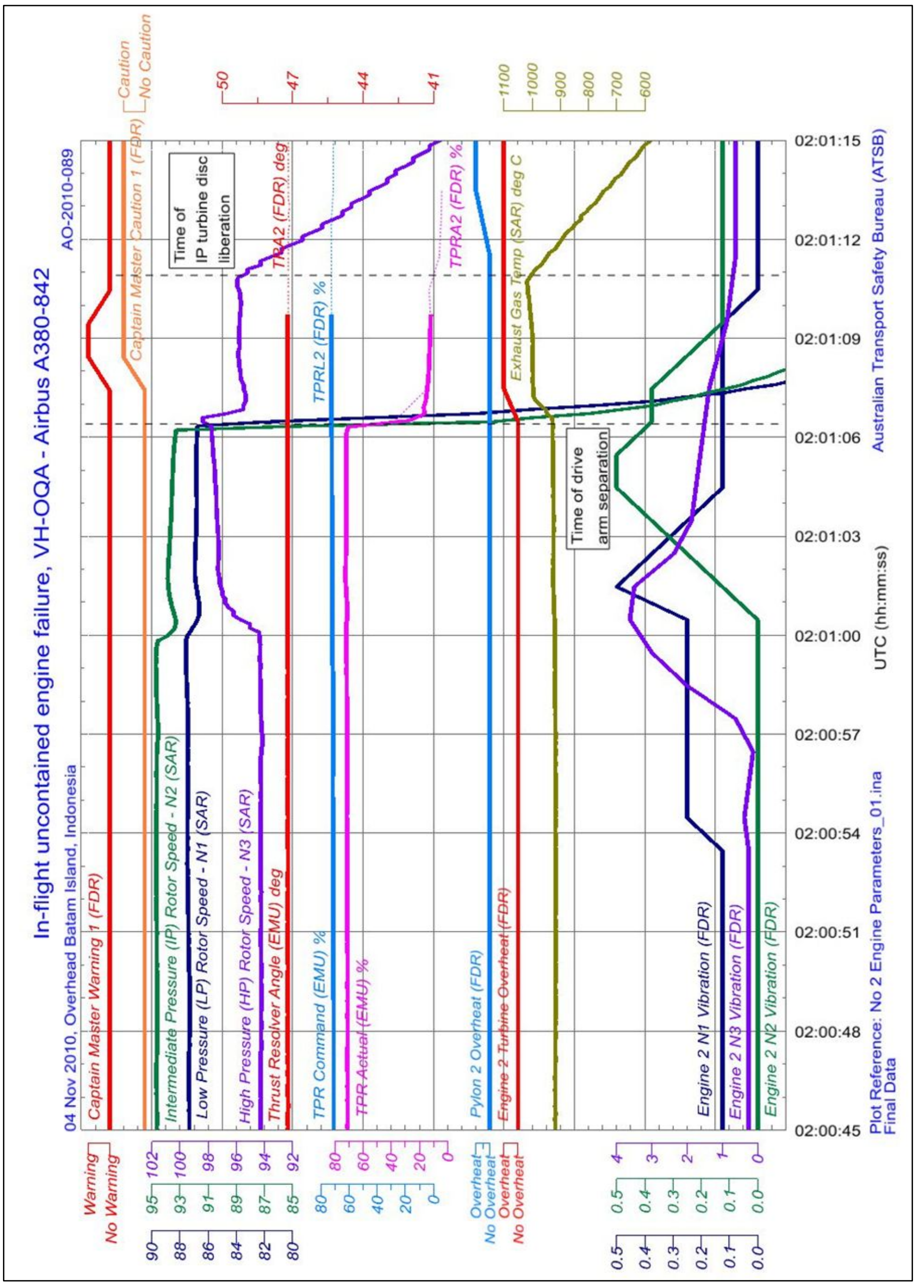
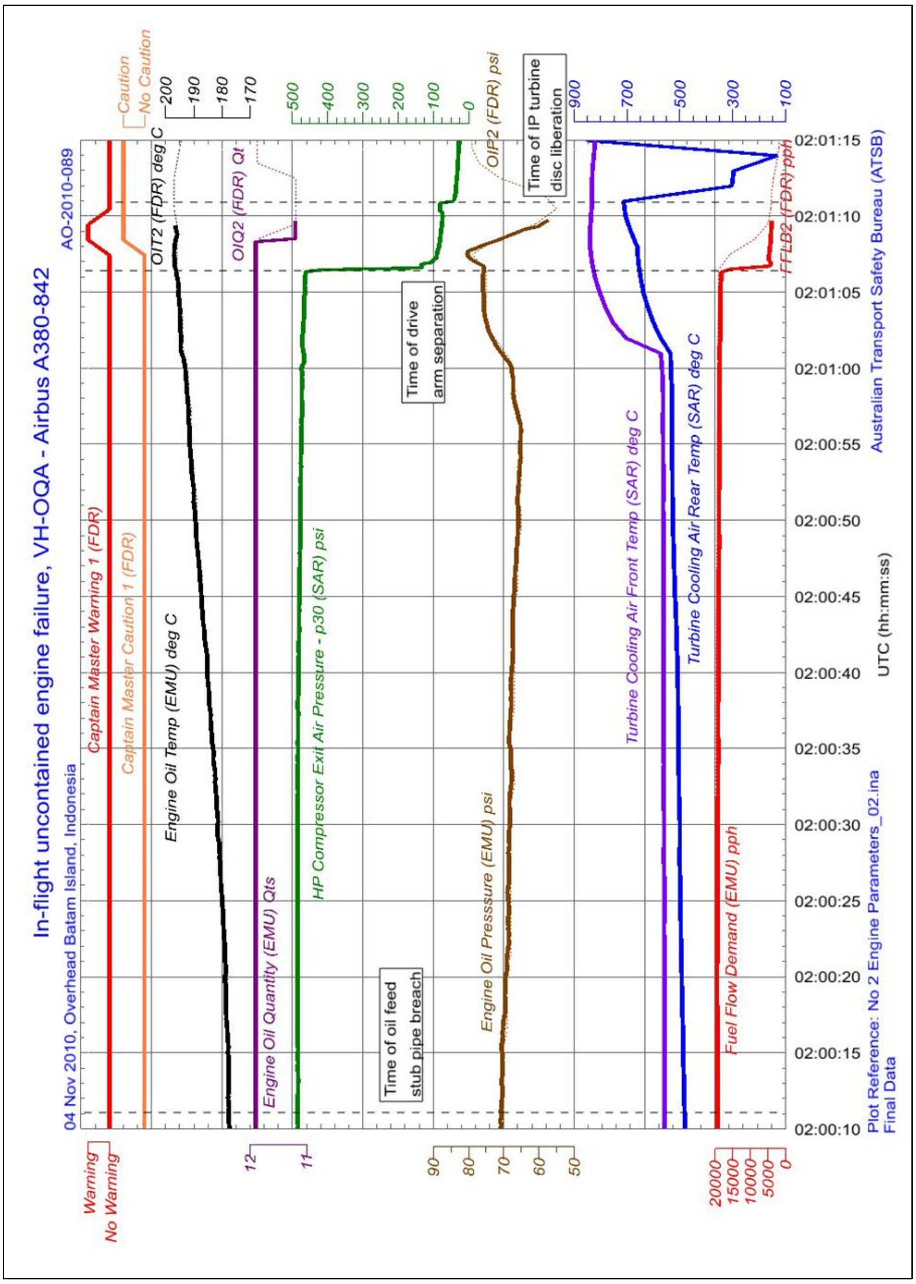


Figure C12: Graphical representation of engine temperature, pressure and oil parameters leading up to event



Sequence of engine events following the IP turbine disc liberation

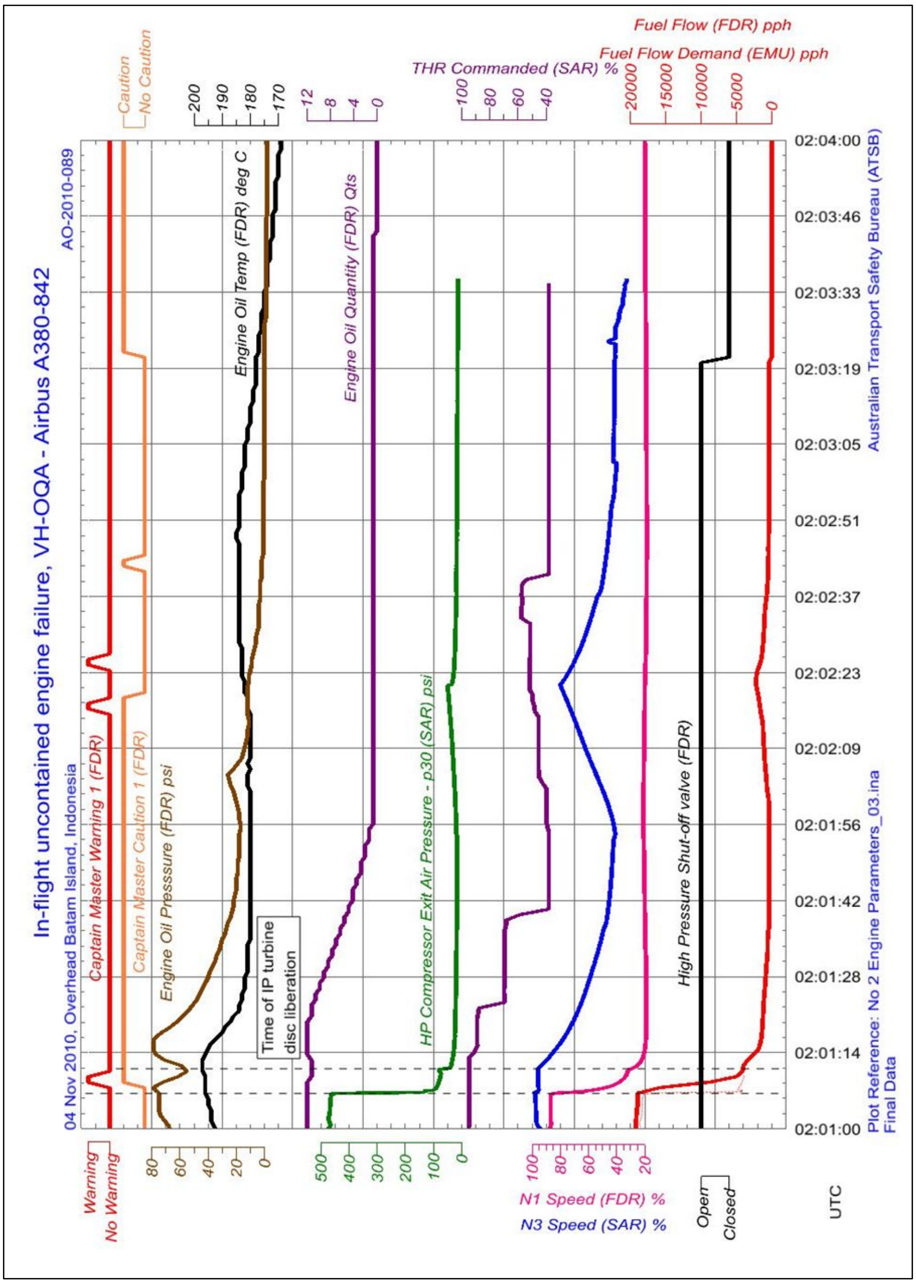
Table C2 tabulates the graphical representation of the engine parameters after the disc separation as shown at Figure C13.

Table C2: No. 2 engine events following the No. 2 engine IP turbine disc liberation

UTC[133] (hh:mm:ss)	Event	Comment
02:01:17	No. 2 engine flameout exceedance message	CMS only
02:01:38	No. 2 engine thrust lever reduced to idle	
02:01:51	No. 2 engine low oil quantity	ENG SD page called
02:01:53	No. 2 engine thrust lever progressively advanced to half range	
02:02:08	P30 rises	
02:02:20	P30 drops	Indicative of compressor stall
02:02:25	Fire warning	
02:02:39	No. 2 engine thrust lever retarded to idle position	
02:03:21	No. 2 engine high pressure shut-off valve closed	

133 Timing from FDR data.

Figure C13: Engine parameters following the IP turbine disc liberation



Aircraft and system events following the IP turbine disc separation

The IP turbine disc liberation resulted in structural and wiring damage to the aircraft. The recorded data provided information about multiple system degradations and aircraft performance issues following the disc separation. The FDR, DAR, CVR, EMU and CMS data was integrated to compile a sequence of events (summarised in Table C3, Figure C14 and Figure C15). Detailed analysis of the effects of the damage on the various aircraft systems is provided in the relevant sections of the main report and appendices.

Table C3: Aircraft system events following the disc separation

UTC[134] (hh:mm:ss)	Event	Comment/Source
02:01:11	Fault indications commence from multiple systems including: Flight controls - Left mid-aileron, spoiler No. 4, slats Hydraulics - green Anti-skid Electrical power – AC2 Bleed air- pylon and wing overheat	Indicative of time of turbine disc failure Altitude 7,250 ft FDR/ VDAR
02:10:59	Hydraulic pressure green decreased. Loss of outboard aileron actuator function and spoilers 8 and 2	
02:18:05	Aircraft commences left orbits over the ocean north of Bintan Island, Indonesia	Altitude 7,400 ft[135]
03:28:51	Aircraft completes ninth orbit and commences descent into Singapore	
03:35:08	Flaps extended to Flap 3 Extension at 1/2 speed	
03:36:39	CVR recording begins during descent	
03:37:52	Landing gear down and locked	FDR
03:38:38	PA announcement to secure cabin for landing	CVR
03:39:58	Cabin confirmed secure	CVR
03:40:58	Landing checks completed – flight crew brief carried out for ‘aircraft abnormal configuration’	CVR
03:41:58	Discussion of commit/go-around options	CVR
03:45:36	Low energy audible warning (‘speed speed speed’)[136] during final approach when passing 1,000 ft RA[137]	CVR and FDR
03:45:55	A/P off	CVR and FDR

134 Timing from FDR data.

135 Barometric corrected altitude

136 Triggered to inform the pilot that the aircraft’s energy has become lower than a threshold under which, to recover a positive flight path angle through pitch control, thrust must be increased.

137 Radio altitude - height above terrain.

UTC[134] (hh:mm:ss)	Event	Comment/Source
03:46:17	Low energy audible warning during final approach when passing 362 ft RA	CVR and FDR
03:46:43	Stall warning at 3 ft RA	CVR and FDR
03:46:47	Touchdown at Changi Airport on runway 20C	CVR/ FDR
03:46:48	Max braking	CVR
03:48:35	Feed Tank 2 fuel quantity 12,640 kg	FDR
03:49:05 03:49:08 03:49:15	No. 3 engine shutdown No. 4 engine shutdown Attempted shutdown of No. 1 engine	CVR/ FDR
03:51:01	First radio contact between the flight crew and Singapore Rescue and Fire-fighting Service (AES)[138]	CVR
03:53:57	Fuel leak in the left wing confirmed by AES	CVR
03:54:34	AES advises flight crew that they are covering leaked fuel with foam	CVR
03:57:46	FDR and CVR data has some interruptions from this time onwards	FDR/CVR
04:39:00	Passenger disembarkation begins	CVR
05:41:05	All passengers disembarked	CVR
06:01:55	Feed Tank 2 fuel quantity 10,064 kg	FDR
06:02:44	Last FDR data recorded	FDR
06:02:53	CVR recording ends	CVR

[138] The Airport Emergency Services (AES) Singapore was part of the Changi Airport Group (Singapore) Pty Ltd (CAG).

Figure C14: Graphical representation of system fault indications at around the time of the event

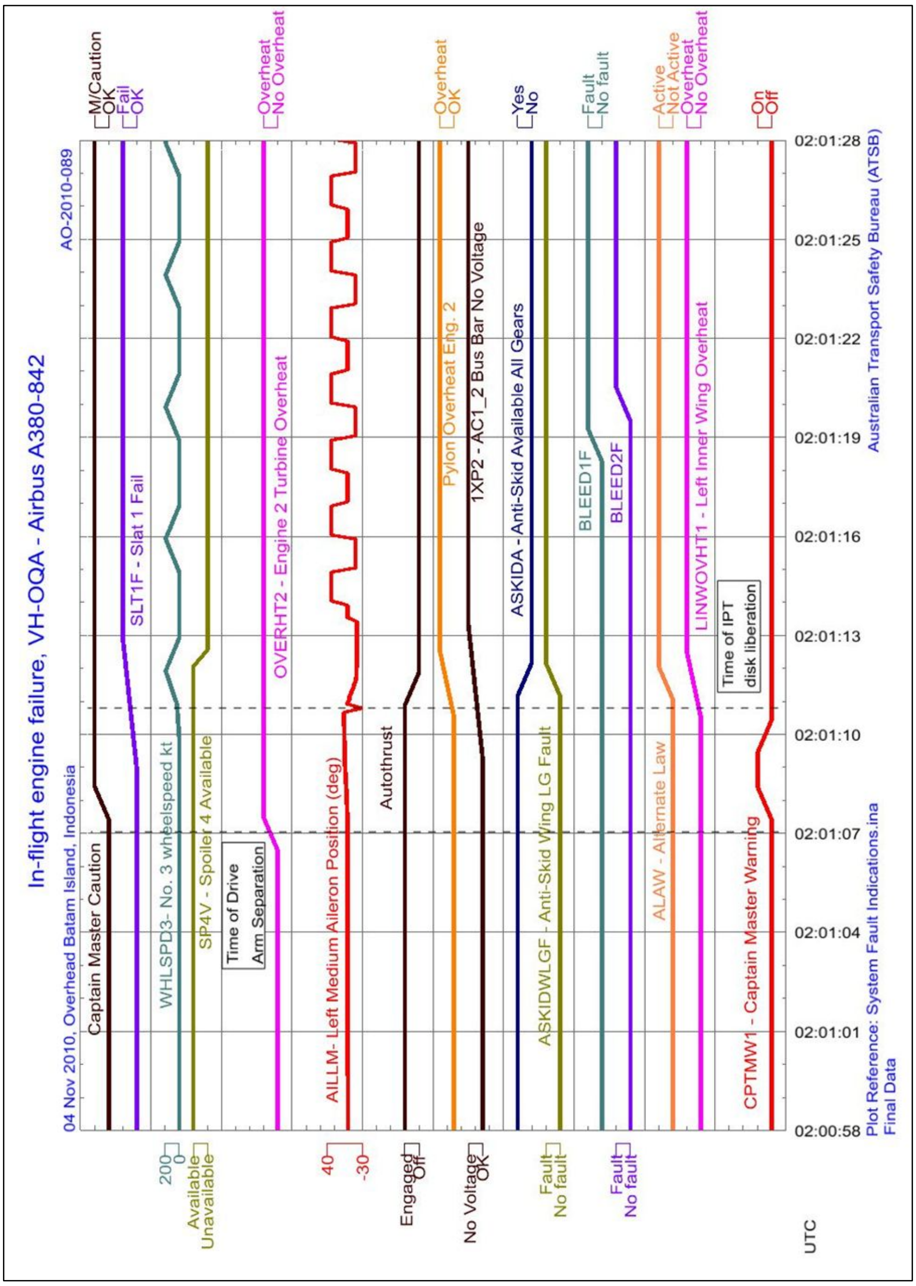
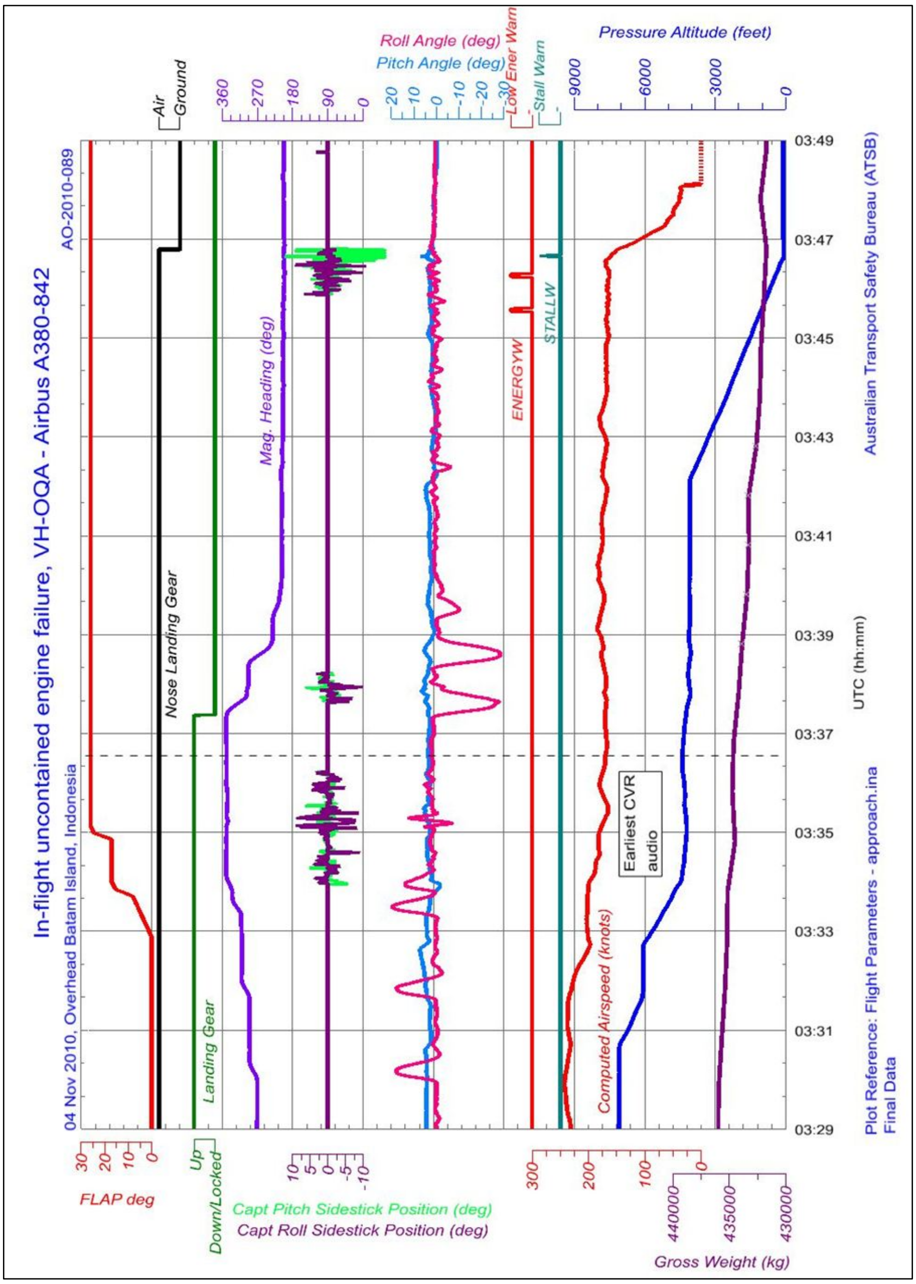


Figure C15: Graphical representation of selected aircraft parameters during the approach and landing



Isolation of power to the CVR

The CVR automatically records when the A380 is in one of the following configurations:

- in flight with engines running or stopped
- on the ground with at least one engine running
- on the ground during the first 5 minutes following the application of power to the aircraft's electrical network
- on the ground up to 5 minutes after last engine shutdown.

To prevent the CVR from recording and therefore overwriting the audio from the previous 2 hours, it would be necessary to isolate electrical power to the unit, by opening the relevant circuit breakers (located remote from the cockpit).

Analysis of the sequence of events provided the timing for the CVR key events during the occurrence flight as shown in Table C4.

Table C4: Timing of key events

Time UTC (hh:mm:ss)	Event
02:01:07	Uncontained engine failure
03:36:39	CVR recording began
03:48:00	Aircraft stopped on runway
03:49:05-03:49:15	Engines No. 3 and 4 shut down and unsuccessful engine No.1 shutdown attempt
03:57:46	CVR power interruptions of about 22 minutes from this point
04:39:00	Disembarkation of passengers commenced
05:41:05	Disembarkation of passengers ended
06:02:53	CVR recording ends

For the CVR to have recorded the engine failure (within its 2-hour recording duration), it would have needed to have been shut down (power removed) by 04:01:00 UTC – 13 minutes after the aircraft came to rest on the runway.

At 04:01:00 UTC, circumstances were such that the aircraft was not yet in a safe condition, with the potential remaining for subsequent safety-related events (such as the outbreak of fire from leaking fuel). At that time, the No. 1 engine was still operating and the passengers were yet to disembark.

From an investigation perspective, deactivation of the CVR at this time would have been inappropriate, considering that any subsequent events would have been unrecorded. The ATSB is satisfied that there was no justification for the flight crew to prioritise the isolation of the CVR above their activities in continuing to manage the emergency response and the safety of the aircraft.

The earliest reasonable time to have powered-down the CVR was about 04:41 UTC, when the risks had been reduced and the disembarkation commenced. This would have increased the flight-time recording duration, but there were many other sources of in-flight information also available. However, once the aircraft was powered-down on the ground, CVR audio was one of the few sources of continuing recorded information. Existing flight data has allowed the ATSB to develop a good understanding of the technical issues leading to the engine failure. CVR audio prior to and around the time of the engine failure would have been very useful in this investigation to examine human factors associated with this event (ECAM, workloads and the like). It would also have been able to provide information on the event, ECAM sequences and crew workload and resource management, as well as provide a correlation with other sources of flight information, allowing the ATSB to more quickly develop a full understanding of the engine failure and the flight crew's response.

CVR audio recorded on the ground was useful in evaluating the emergency response by AES, air traffic control, other ground personnel and the crew.

APPENDIX D: WING FIRE

SUMMARY

On 4 November 2010, an Airbus A380 aircraft, registered VH-OQA, departed from Changi Airport Singapore, for Sydney, Australia. Following a normal takeoff, the crew reported that while maintaining 250 kt and climbing through 7,000 ft above mean sea level, they heard two almost coincident 'loud bangs', followed shortly after by indications of the failure of the No. 2 engine.

A subsequent examination of the aircraft found that the No. 2 engine had sustained an uncontained failure of the intermediate pressure (IP) turbine disc. Sections of the liberated disc penetrated the left wing and the left wing-to-fuselage fairing, resulting in structural and systems damage to the aircraft.

A passenger reported seeing a fire in the region of, and extending from the left upper wing surfaces. Photographs supplied by the witness showed a dark, soot-like residue on the upper wing surface behind a large perforation. Photos taken by the Australian Transport Safety Bureau (ATSB) during the investigation showed similar external residue and a similar residue on the internal upper surfaces of the left inner fuel tank (LIT). Testing of the residue from the internal surfaces of the LIT characterised it as a carbonaceous material, typical of combustion soot.

While it appeared that the general conditions within the LIT, in a steady-state (static) vapour environment, were outside the flammability range of Jet A-1 fuel, a literature review indicated that combustion was possible, due to factors that drove conditions in the tank into the dynamic vapour flammability range.

A number of potential sources of fuel vapour ignition were explored, but the passage of the hot IP turbine disc fragment through the tank was considered the most likely.

The ATSB found that the impact and passage of the large disc fragment through the wing tank likely created fuel agitation and misting in the tank vapour space (ullage), dynamically raising the vapour concentration. That effect, combined with the fragment surface temperatures (significantly hotter than the auto-ignition and hot-surface ignition temperatures of Jet A-1 fuel) led directly to the ignition event. Friction between the liberated fragment and the tank skin was also likely to have provided an additional localised heat source. There was no evidence of sustained excessive temperatures (fuel tank deformation or heat distress), which indicated the fire had likely been a short, unsustained event.

There were a number of possible reasons why the left inner tank fire was not sustained:

- The flash fire initially consumed the oxygen inside the tank; reducing the oxygen concentration level to below the threshold required for sustained combustion of Jet A-1.
- The general temperature rise associated with the initial combustion was low and insufficient to increase the temperature of the fuel inside the tank to at least the lower flammability limit of Jet A-1. Quenching of the flame by the cool fuel tank walls (evident from the sooting) and the entry of cooling air through the tank perforations may have promoted this.

FACTUAL INFORMATION

Witness reports

Immediately following the engine failure, a passenger who had been seated on the upper left deck of the aircraft reported seeing a fire burning on the wing. The passenger reported hearing two bangs approximately 5 minutes after takeoff. Subsequent to this, the passenger observed a 'couple of holes' in the upper surface of the left wing with a fire burning immediately out of one of the holes (Figure D1) The witness described the fire as extending straight up to a height that was about the same height as the diameter of the hole in the wing and was above the wing for 5 to 6 minutes. Although it had reduced in severity after this, the passenger mentioned that it remained visible for approximately 10 minutes. The witness described the fire as bright yellow in colour and intermittent; 'at times it was gone for a few seconds and then it came back'. During the ATSB interviews, the flight crew did not make mention of a fire in the left wing. Subsequently, the flight crew reported that, during the flight, they were not aware of such a fire.

Figure D1: View from the window on the upper deck, left side showing the presence of smoke/dark residue on the aft side of the hole (arrowed)

Image supplied by a passenger

Photographs supplied by the witness, those taken after the aircraft had landed and visual confirmation by the ATSB all showed the presence of smoke/dark residue on the aft side of the hole in the upper wing skin (Figure D1 and Figure D4).

A review of the damage observable from the passenger's seat on the upper deck indicated that the engine mechanicals and structure could be seen through the arrowed hole in Figure D1.

A380 wing tank detail

The A380 wing fuel tanks are integral to the aircraft's wing structure (Figure D2). The LIT is an L-shaped chamber and is located close to the fuselage. The tank is bound by wing ribs 1and 13 behind the centre spar, and ribs 8 and 13 in front of the centre spar (Figure D3). The inner tanks are capable of holding a total fuel weight of 36,220 kg (79,850 lb). Information recorded on the flight data recorder (FDR) showed that at the time of the engine failure, the LIT was approximately 1/4 full, holding approximately 4,300 kg (9,480 lb) of fuel. The FDR data showed that the weight of the fuel in the LIT remained constant during the period following the failure of the IP turbine disc.

Figure D2: Fuel tanks onboard the A380

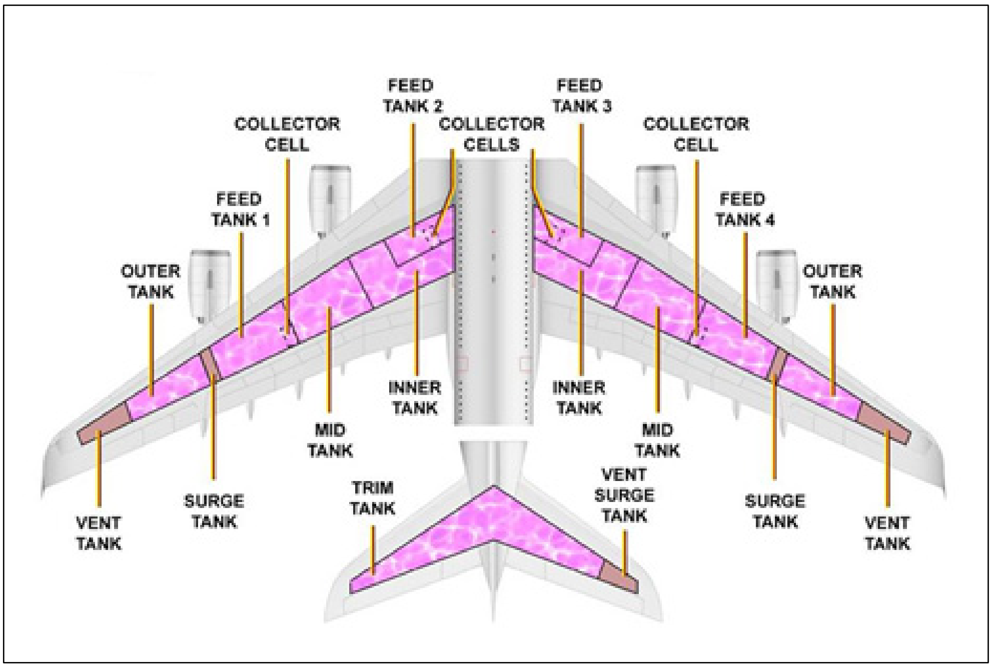


Image courtesy Airbus training manual, Level 1, ATA 28

Damage to the left inner tank

The LIT structure was considerably damaged by debris from the No 2 engine failure. The damage resulted when a large segment of the liberated IP turbine disc struck the wing bottom skin and passed through the front spar and LIT structure, before exiting through the wing upper skin. As a result of the disc's trajectory, the most significant damage to the LIT occurred to the front spar between ribs 9 and 10, and the upper wing skin in the region of rib 9 (Figure D3 to Figure D5).
Table D1 provides the dimensions of the through-penetrations that were identified in the LIT.

Table D1: Summary of damage in the left inner tank

Location	Dimensions (mm)
Front spar, Rib 9 – 10	350 x 350
Front spar, Rib 9 – 10	82 x 20
Front spar, Rib 10 – 11	150 x 250

Front spar, Rib 12 – 13	80 x 50
Wing bottom skin, Rib 10 – 11, Stringer 6	140 x 110
Wing upper skin, Rib 9 -10, Front spar – stringer 1	460 x 120

Figure D3: Schematic of the LIT (green) and feed tank 2 (blue), showing the location of the major front spar and top skin damage

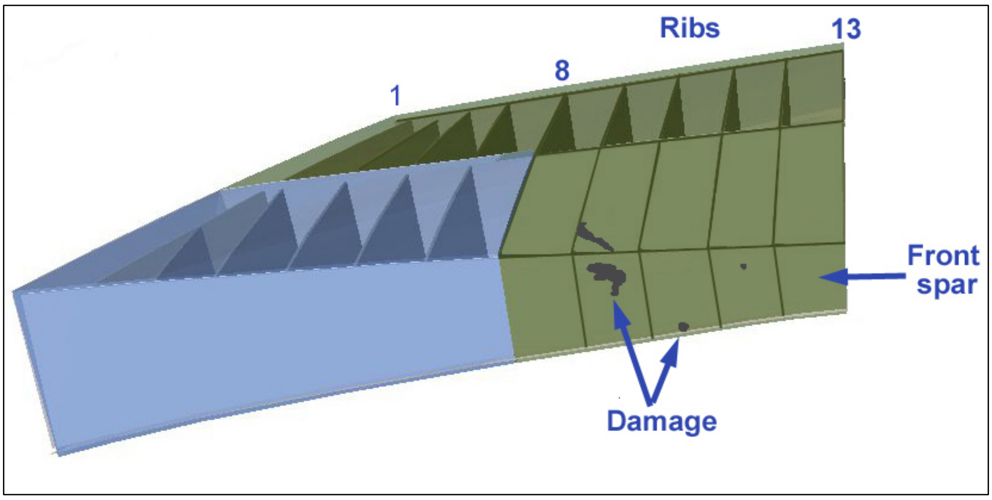
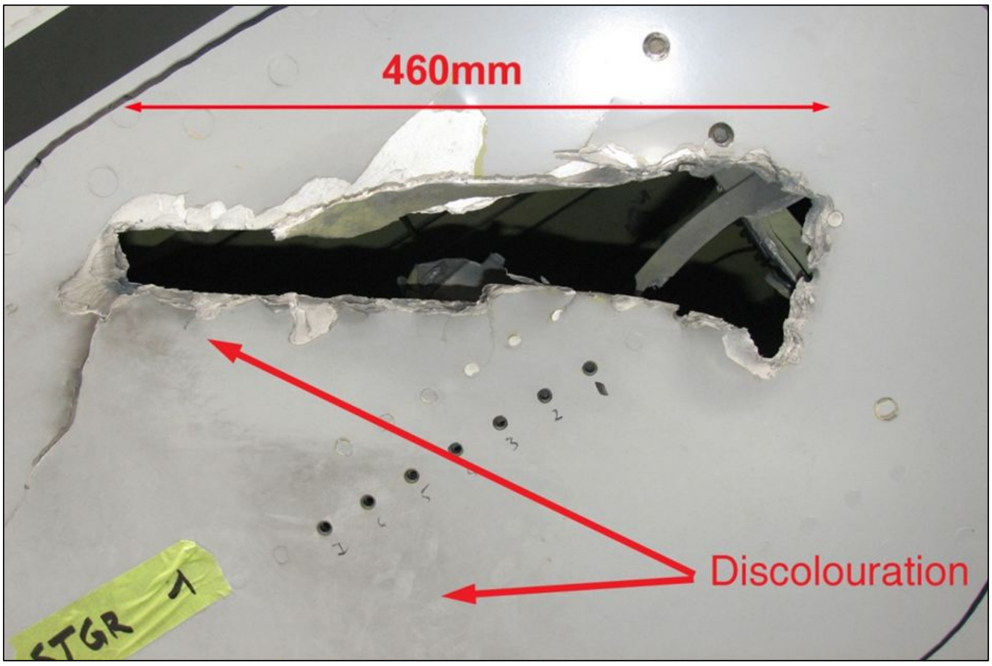


Image modified from an Airbus-supplied model.

Dark coloured residue

During the aircraft damage assessment, a dark coloured residue was identified on some surfaces within the LIT. Minor discolouration was also observed on the fractured surfaces of the large hole in the upper wing skin and on the painted surface of the skin towards the rearward side of the disc exit hole (Figure D4).

Figure D4: Major damage on the LIT upper wing skin



An examination of the internal surfaces of the LIT showed the residue to be principally located on the upper regions of the front spar (above the large hole), and along the underside of the top skin on the surfaces of the stringers (Figure D5 and Figure D6). The residue was also observed within the vent pipe shown in Figure D6. No evidence of dark residue was observed on either the wing underside, or on the front droop-nose structure.

Figure D5: Damage and residue on the front spar (viewed forward from inside the LIT)

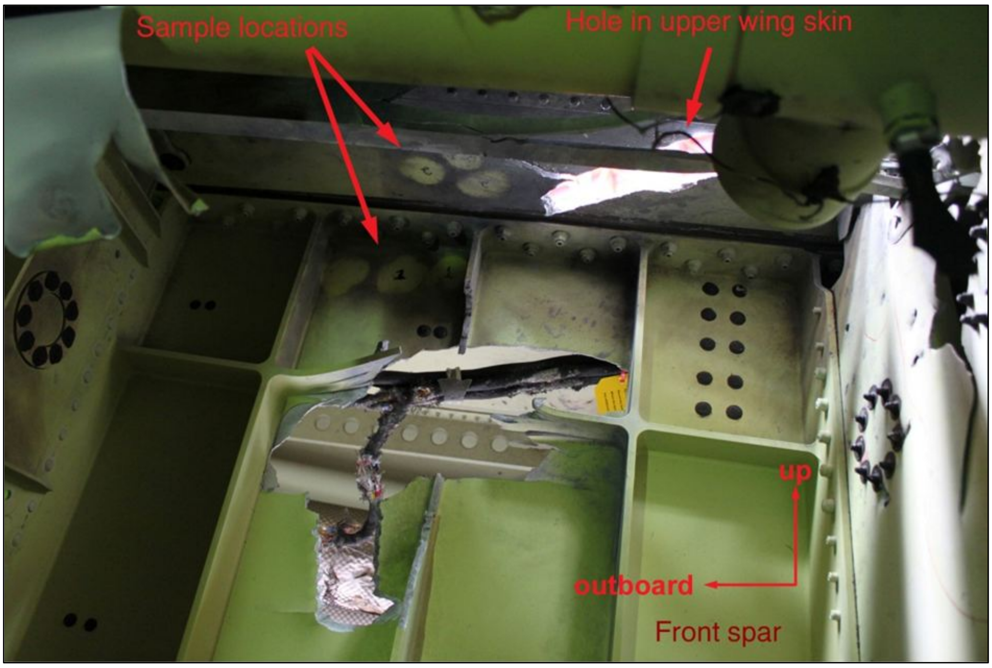
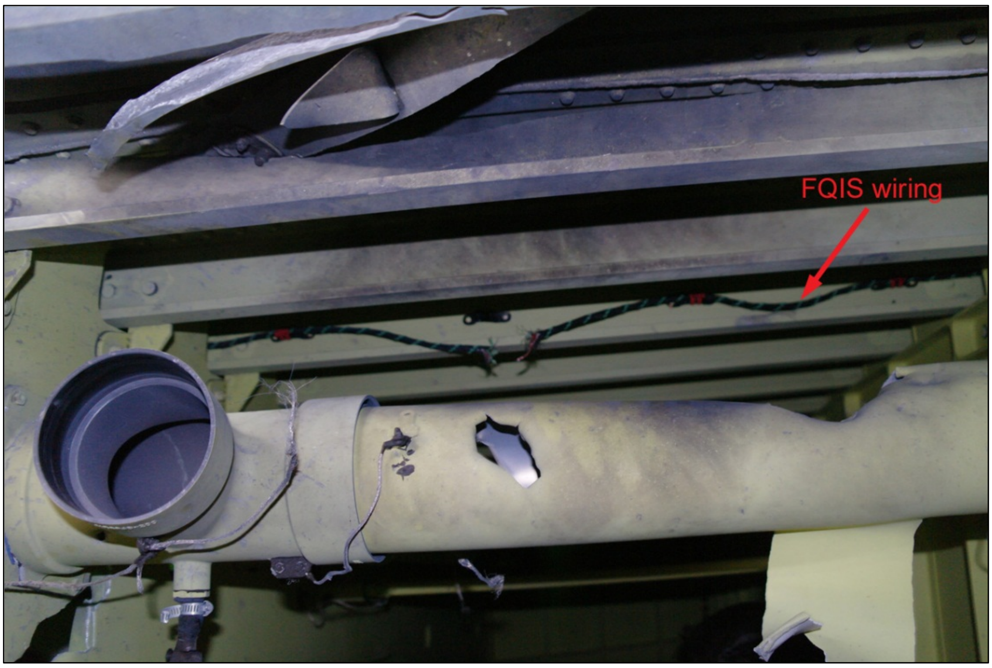


Figure D6: Residue observed along the top of the LIT



Residue swab samples were taken from a number of locations within the LIT for further characterisation Figure D5. The painted surfaces immediately underneath the residue did not exhibit any damage, discolouration or blistering. Airbus advised that the paint used within the LIT had undergone testing at between 100 °C and 180 °C with limited colour change noted.

Analysis of dark residue/debris

The samples from within the LIT were forwarded to the Australian Federal Police Chemical Criminalistics laboratory in Canberra for examination. In order to establish the composition of the dark residue, the samples were analysed using a number of techniques, including: microscopic examination, Raman spectroscopy, scanning electron microscopy, energy dispersive x-ray spectrometry and gas chromatography-mass spectrometry. The main findings of that analysis were that the:

- dark residue was amorphous carbon, or soot
- soot was heavily enriched with fine (spherical) particles of aluminium
- dark residue did not contain heavy petroleum distillate hydrocarbons, such as engine oil
- dark residue was not bacterial in nature.

Examination of wiring

The severed and damaged wires in the areas surrounding the perforations in the LIT presented a potential ignition source for the fire that developed within the tank. As such, the wiring looms and cables were removed from the area surrounding the main damage sites in the left wing structure and forwarded to the ATSB's Canberra facilities for further examination. The examined wiring originated from the:

- fuel quantity indicator system (FQIS)
- No.1 engine variable frequency generator (VFG) feeder cable
- No.2 engine VFG feeder cable
- wiring loom harnesses 1M, 1S, 2M, 2S and 2P.

The results of those examinations are described in the following sections.

Fuel Quantity Indicator System

The FQIS wiring was the only wiring located inside the LIT (shown in Figure D6 and Figure D7).

Figure D7: Location of the FQIS wiring within the LIT (green)

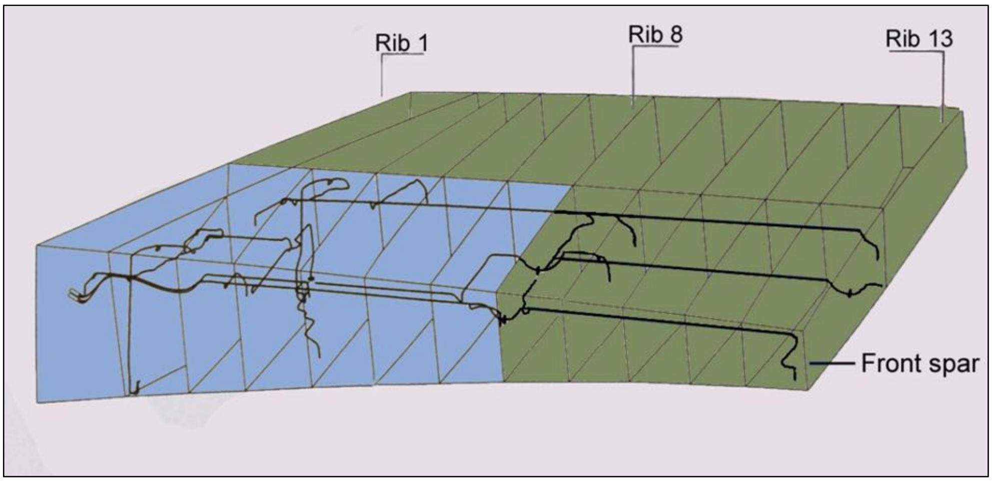
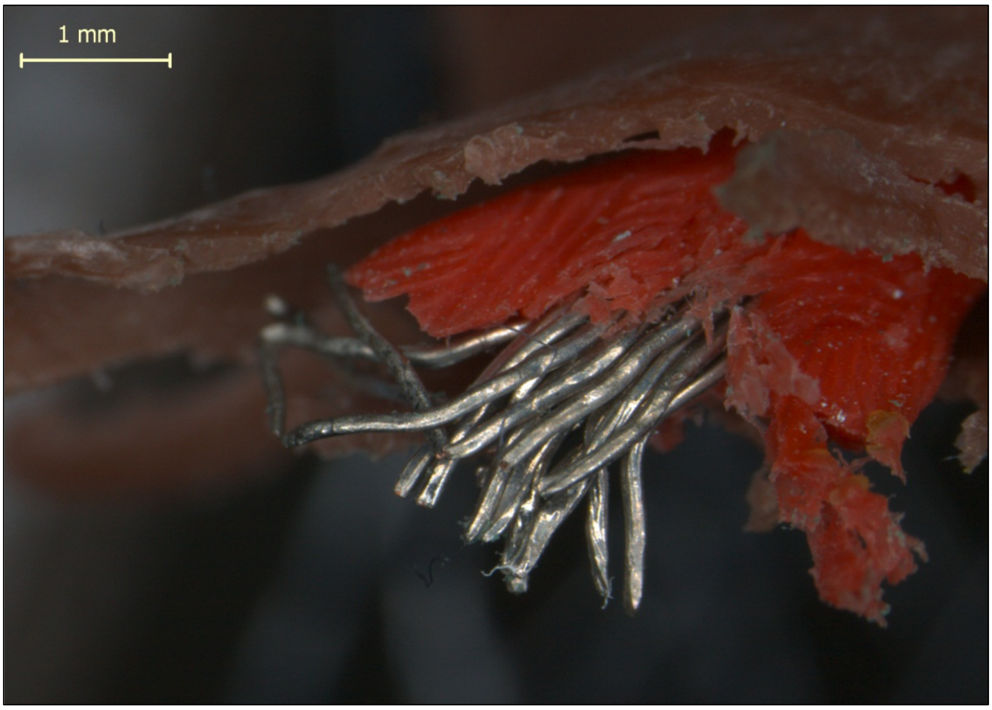


Image: Modified from A380 Aircraft Maintenance Manual

The section of FQIS wiring supplied was a y-shaped piece, approximately 1.2 m long, which had been severed during the engine failure. A small area of additional damage was observed towards the end of one of the sections. A visual inspection of the exposed wires showed severe mechanical damage; many of the conductors exhibited significant deformation as a result of contact with the liberated engine debris. The inspection identified a number of ‘cup and cone’ and angular shear fractures, all consistent with tensile overstress failure (Figure D8). No evidence of any discolouration or arcing was observed on the fractured ends of the wires, and the plastic coating did not exhibit any discolouration, burning or charring.

Figure D8: Magnified view of the severed end of the FQIS wiring



Variable Frequency Generator feeder cables

The VFG feeder cables for the No. 1 (VFG1) and No. 2 (VFG2) engines were located in the bay forward of the front spar (Figure D9). The VFG1 and VFG2 feeders had been severed at the left wing leading edge. An examination by Airbus reported that there was no visible trace of arcing, short circuit or overheats on any cable.

Following removal from the aircraft, sections of both feeder cables were sent to the ATSB's Canberra facilities for further examination. The VFG1 cable section had been completely severed, while the section of VFG2 cable had some remaining wires intact.

Figure D9: Location of VFG1 and 2 feeder cables along the wing front spar

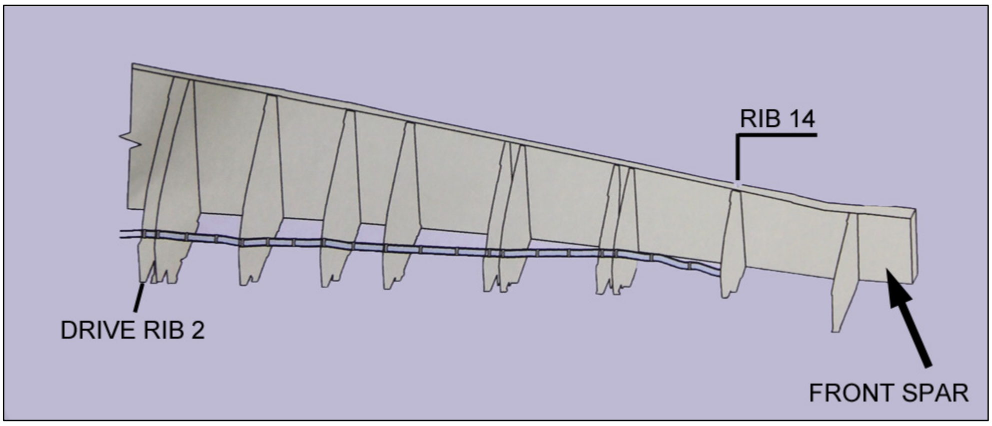


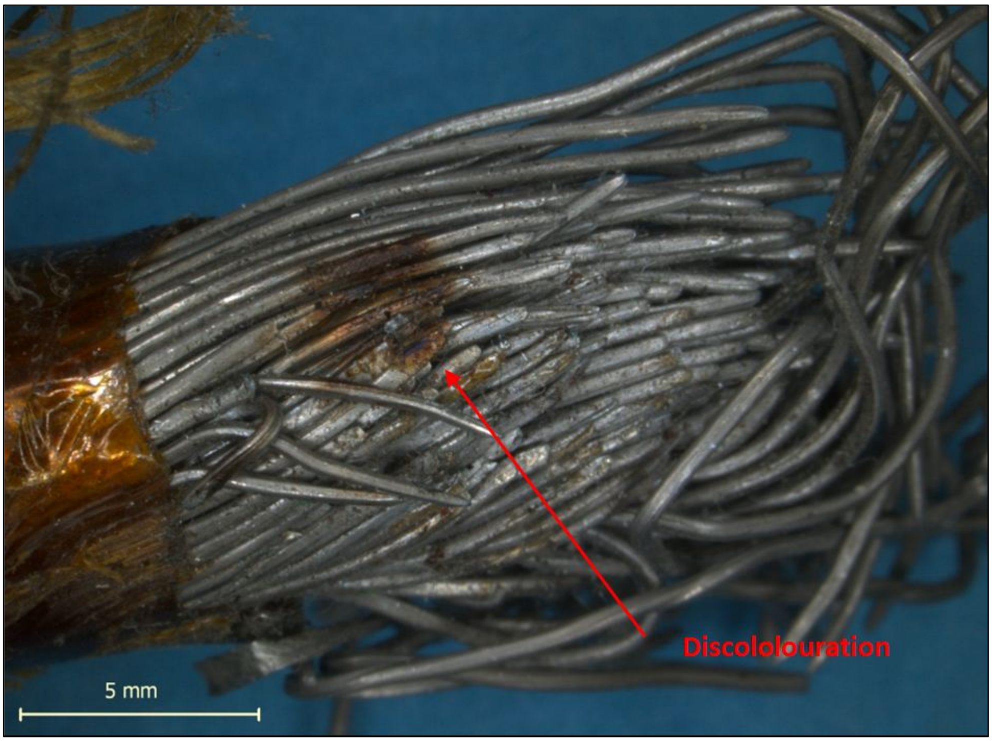
Image: Modified from A380 Aircraft Maintenance Manual

The severed wires exhibited deformation and markings consistent with the mechanical damage that was observed on the FQIS wiring. The fractured strands exhibited 'cup and cone' features that were indicative of an overstress failure (Figure D10).

Figure D10: Magnified view of the severed end of VFG2 wiring

A small discoloured spot was identified on one of the VFG2 cables supplied (Figure D11). While the source of the discolouration was not identified, there did not appear to be any associated melting that would indicate that it was a result of arcing.

Figure D11: VFG2 cable showing small area of discolouration



Wiring harnesses

The following VFG monitoring harnesses were provided to the ATSB for examination:[139]

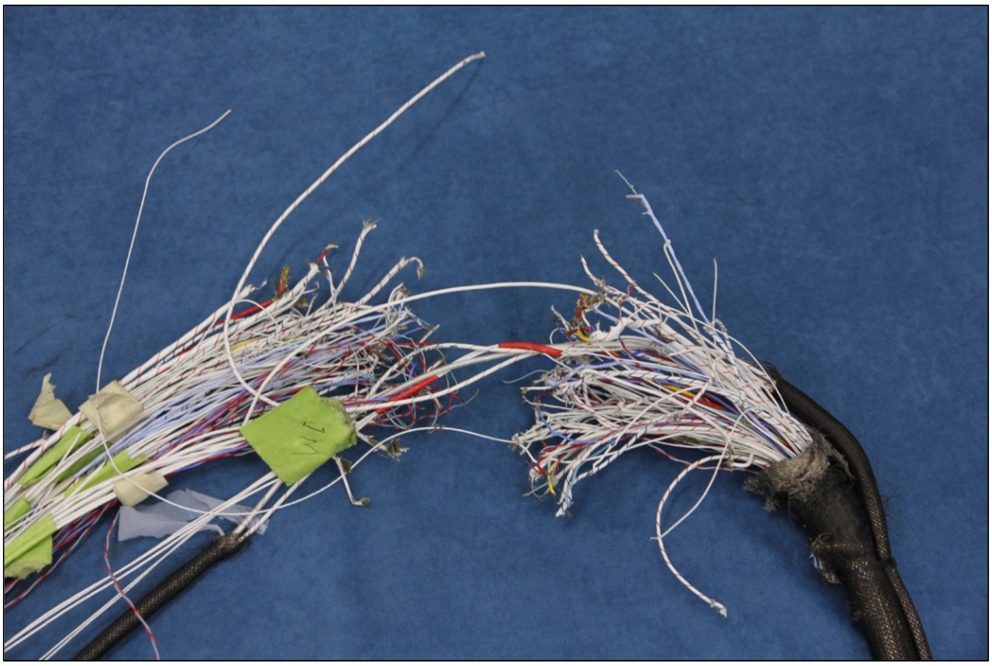
- Harness 1M 3001 VB
- Harness 1S 3013 VB
- Harness 2M 3007 VB
- Harness 2S 3019 VB
- Harness 1P – 3031VB, 3033VB, 3035VB and 3025VB

The sections of harness comprised of a number of smaller wires encased in an outer sheath. Damage to the harnesses varied from complete separation to minor damage to the outer sheath and a small number of wires (Figure D12 and Figure D13).

[139] The harnesses were a set of electrical wires identified by four digits and the letters VB. The first two identifiers denoted the harness route, where the number designates the system and the latter the route category. For example, 'M' meant miscellaneous and route 'P' included power lines.

Figure D12: Overview of severed harnesses in-situ, located in the left wing leading edge (also showing the front spar, looking rearward)

Figure D13: Magnified view of Harness 1M 3001VB



All of the exposed severed wires were examined, with no evidence of any arcing noted. The fractured ends of the wires were consistent with overstress failures as a result of localised mechanical impact.

Fuel tank flammability

A review of the current literature on aircraft fuel tank fires was conducted to ascertain the conditions under which combustion can occur within an aircraft fuel tank. A further study was undertaken to look at the factors that might limit the sustainability of any such fire, and conversely, what factors might support the continuation (and possible growth) of a fire under similar conditions.

A discussion on the recent activity by the various worldwide aviation safety and regulatory groups to minimise the inherent risk of fuel tank susceptibility to fire is also included below.

Combustion theory

Fire is a complex process controlled by chemical and physical factors. Fire is initiated by achieving an appropriate heat balance,[140] and can only be maintained if the associated chemical reactions can sustain themselves.

The *fire triangle* concept states that three components - fuel, an ignition source, and an oxidiser - must be simultaneously present for a fire to occur. To begin with, the fuel and oxygen (the oxidiser) are required to be present in flammable proportions. Heat is then added locally (the ignition source) until a large enough population of free radicals (unstable chemical species) is developed to ignite the mixture. The temperature must be high enough to provide the energy required to break chemical bonds in fuel molecules and enable various reactions to occur. The chemical reaction is then propagated until an equilibrium flame temperature is reached. Usually, this process is vigorous enough for thermal radiation to be released at a visible wavelength – thus the appearance of a flame.

Combustion of pre-mixed fuel vapours is subsonic in speed and called a *deflagration*. The deflagration can be ignited by an external source or self (auto) ignition. External ignition sources can be flames, sparks or hot surfaces. The flame itself can be a diffusion flame (laminar, for example a candle, or turbulent, as in the case of most fires) or pre-mixed (laminar, for example a Bunsen burner, or turbulent, such as in an internal combustion engine or a fuel vapour fire inside a fuel tank).

Deflagration events can also be transient (flash) or sustained (fire). This will be determined by a number of factors, but essentially stems from the heat generation rate compared with the heat loss rate and the production rate of radicals compared with their termination rate.[141]

The factors that influence whether combustion occurs and how long it will last include the proximity of the fuel to an oxidiser, any ventilation, and the temperature and pressure. For hydrocarbon fuels, the principal chemical reaction occurs through gas-phase radical formation, whereby thorough mixing is required between the fuel and the oxidiser in the gaseous phase. In this context, ventilation is important, as the gas ingredients (fuel and oxygen) must be incorporated (if not pre-mixed). It follows that any physical confinement will hinder gas travel (and hence fuel/oxygen

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[140] Heat balance is the establishment of a condition of thermal equilibrium in a space, where the heat gains equal the heat losses.

[141] BlazeTech, *Aircraft fire and explosion in accidents, combat, and terrorist attack: Course notes*, Nov 2011.

mixing), while too much ventilation can lead to cooling or vapour dilution sufficient to extinguish a flame.

Temperature is important as it controls the vapour pressure and initial vapour flammability. The temperature must be high enough to initiate a reaction, and must remain so to continue the process. Pressure also affects the flammability limits, with increasing pressure widening the temperature range at which combustion will occur.

A transition from deflagration to detonation (supersonic) combustion can occur in gaseous mixtures due to flame acceleration by mechanisms such as precompression of the gasses ahead of the flame, turbulence generation from obstacles and solid boundaries, and flow speed increases at physical constrictions along the flame path.

Properties of aviation fuels

The aircraft's fuel tanks were filled with Jet A-1 fuel, which is a kerosene fuel that has essentially the same specification limits as Jet A fuel. The only property that differs between the two fuel types is that Jet A-1 has a lower maximum freezing point of -47 °C. International fuel standards[142] specify a number of properties, including distillation temperature, flash point temperature and net heat of combustion. Jet A-1 has a specified boiling range of 149 to 290 °C, a flash point minimum of 37.8 °C (100 °F) and an auto-ignition temperature in the range of 200 to 260 °C.[143]

The Jet A-1 properties of relevance for fire predictions (such as flammability limits, ignition energy, electrostatic charging tendency, auto-ignition temperature and rate of fire spread) are not included in the fuel specifications and are not generally measured during fuel inspection.[144]

The *flash point* of a fuel is described as the minimum liquid temperature giving sufficient vapour to form a flammable vapour-air mixture at the liquid/air interface. The mixture will ignite when subjected to a small flame, but the fire may not sustain and will therefore 'flash' and then self-extinguish. The *CRC Handbook of aviation fuel properties* gives the average flash point of Jet A-1 as 42.2 °C, but different methods used to determine the flash point do not always give equivalent results.

In general, the flash point is useful for qualitative ranking of fuels. However, it has been shown that liquid fuels present in tanks and pools will propagate flames at temperatures below the established flash point.[145]

Another relevant property of a fuel is known as the *fire point*. This determines the temperature at which a flame will be sustained after the ignition source is removed.

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[142] There are three major specifications in civil use worldwide, ATSB D1655, the British Def Stan 91-91 and the Russian specification GOST 10227.

[143] Coordinating Research Council report No. 635, *"Handbook of Aviation Fuel Properties"*, 2004 Third Edition.

[144] Fuel flammability task group, DOT/FAA/AR-98/26, *A Review of the Flammability Hazard of Jet A Fuel Vapor in Civil Transport Aircraft Fuel Tanks*, Federal Aviation Administration, June 1998.

[145] Ibid.

Fire points are generally higher than flash points by about 4 to 5 ^{o}C for most hydrocarbon fuels.

The *vapour pressure* of a fuel is the pressure of a vapour in equilibrium with the liquid and is temperature dependant. If the vapour pressure at a particular temperature and the volume of space is known, calculations can predict the amount of fuel existing in the vapour. The Reid Vapour Pressure is a common measure of the volatility of a fuel and is defined as the vapour pressure exerted by a liquid at 100 °F (37.8 °C). This test method results in an extremely low value for Jet A/A-1 (less than 1 psi/6.9 kPa), with the effect that, for low volatility fuels such as Jet A/A-1, the results are of limited application for predicting flammability.

Flammability envelope

The standard flammability envelopes shown in Figure D14 were derived from various literature sources and contain a number of assumptions, including a static (unmoving) tank, a well-mixed ullage[146] and thermodynamic equilibrium with the atmosphere. The graph indicates that Jet A is mainly non-flammable, with a small exception in tropical atmospheres at low altitude.

Figure D14: Typical flammability limits for common aviation fuels

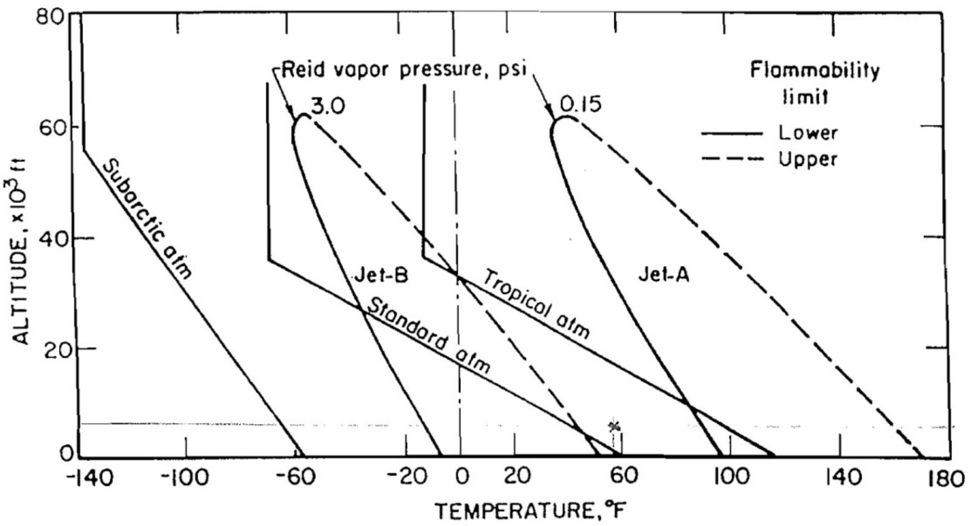

Image: Courtesy of Fuel flammability task group, DOT/FAA/AR-98/26, *A review of the flammability hazard of Jet A fuel vapour in civil transport aircraft fuel tanks*, Federal Aviation Administration, June 1998.A380 Aircraft Maintenance Manual.

It should be noted that the static flammability envelope has not been used as a tool in fuel tank design or flight operations. That is, designers have always assumed a flammable vapour can exist in fuel tanks, and accordingly, have adopted standards to preclude ignition sources from within such tanks.[147]

Importantly however, flammability limits can be extended, and the tank ullage can become flammable under the influence of a number of variables. These variables can include fuel properties, fuel tank design, and the amount of fuel in the tank, as well as external factors such as heat transfer from surrounding equipment.

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[146] Ullage is the volume, or free space, above the liquid level in a tank.

[147] Ibid.

It is well documented within the literature that dynamic processes (conditions of movement or agitation) can significantly broaden the flammability envelope over that present in the static case (Figure D15). The rich limit[148] will remain unchanged in such cases, as the dynamic processes will add fuel to an already rich condition. One of the most important of these processes is the formation of a mist of fuel droplets inside the tank due to fuel 'sloshing' under normal aircraft vibration and motion. The understanding of these dynamic effects is qualitative in nature, due to the inherent broad variability of conditions that are describable as being 'dynamic'.

Figure D15: Dynamic versus static flammability envelope

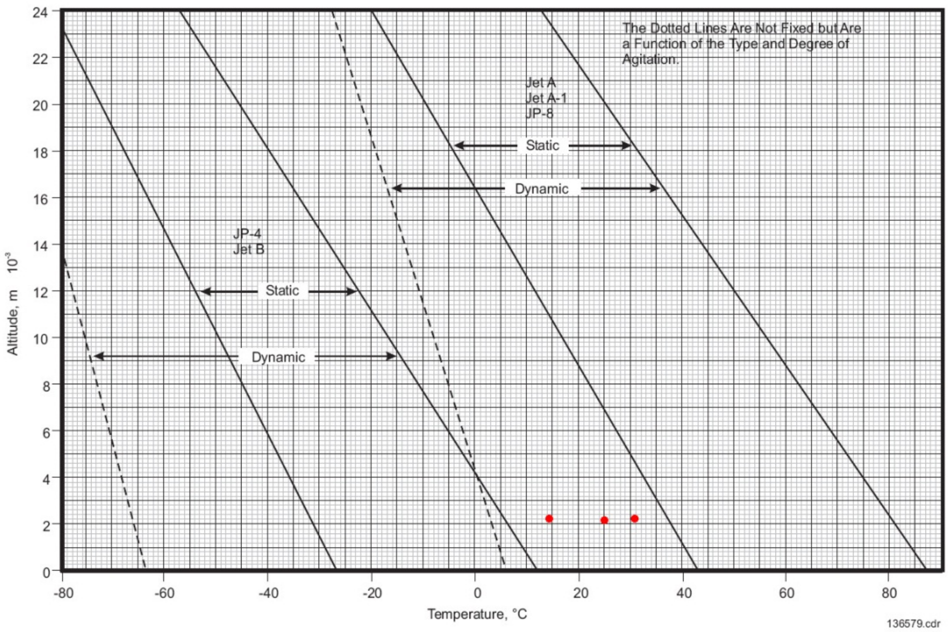

Image: Courtesy of Coordinating Research Council report No. 635, *Handbook of Aviation Fuel Properties*, 2004 Third Edition

Conditions within the aircraft's LIT

At the time of the engine failure, the aircraft was at an altitude of approximately 7,000 ft (2,100 m). The aircraft's flight data recorder provided information on the temperature within the fuel tanks at that time (Table D2).

Table D2: Conditions at the time of the occurrence

Location	Temperature (°C)
Feeder tank 2 (fuel)	26.5
Left outer (fuel)	29.7
Outside air temperature (OAT)	15

The temperature on the ground at the time of departure was 27.5 °C.

The three temperatures (feeder tank 2, left outer and OAT) are shown on Figure D15 with red dots. While the conditions represented by these data points

148 The rich limit is the upper flammability limit and is the highest concentration of a combustible gas that will produce a flash of fire in the presence of an ignition source.

were located below the static flammability lower limit, under dynamic conditions, the vapour mixture may fall within the broader flammability limits. If the agitated fuel droplets were subsequently vaporised by the passage of a fragment of hot debris through the fuel tank, the fuel vapour pressure in the ullage may have been further raised.

Potential ignition sources inside fuel tanks

A review of potential ignition sources within, and around the LIT was undertaken to establish whether these sources may have contributed to the fire event. Research has shown that potential ignition sources could include pumps, an electrical source (wiring malfunction, electrostatic discharge, fuel indicator system), lightning and fragment impact.

Hot surface ignition

A review of the current literature indicated that while in-tank fires can be produced by engine debris, their expected probability of occurrence is very low. It has been shown however that, in the presence of mechanical agitation (either by aircraft or fragment motion), the composition of the fuel/air mixture and the fuel temperatures may be sufficient to allow a fire to ignite, in the event of a fragment entering the tank ullage.

The favourable conditions for the ignition of a combustion process by a hot fragment depend on the fuel/air mixture, temperature of the fragment and the duration of exposure to the hot fragment. There is no unique threshold temperature for hot-surface ignition, as it will be influenced by numerous factors, such as surface geometry (concave or convex), whether in a closed or open environment, local air velocities and residence time.[149]

Moussa et al[150] provided an analysis of the convective heat transfer from a fragment to the ullage gases in order to determine the temperature of the fragment and its surrounding vapour film as a function of time. Their research showed that at sea level, fragment temperatures greater than 712 °C can raise the air/fuel film temperature sufficiently for ignition to occur (that is, above the auto-ignition temperature). As such, debris from the engine exterior or casing would be unlikely to cause ignition, as it generally does not reach such temperatures during operation. However, internal engine components (turbine blades, discs) frequently operate at temperatures exceeding this value, and thus present as a viable ignition source should they enter the tank ullage.

Information supplied by the engine manufacturer estimated that the IP turbine disc drive arm reached temperatures of between 1,125 and 1,290 °C. While the bulk of the disc may not have reached these temperatures, and the liberated sections would have begun to cool immediately following ejection, those estimated temperatures are well above the limits required for hot surface ignition. Additionally, the research

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[149] Fuel flammability task group, DOT/FAA/AR-98/26, *A review of the flammability hazard of Jet A fuel vapor in civil transport aircraft fuel tanks*, Federal Aviation Administration, June 1998.

[150] NA Moussa, MD Whale, DE Groszmann & XJ Zhang, DOT/FAA/AR-96/95, *The potential for fuel tank fire and hydrodynamic ram from uncontained engine debris,* Federal Aviation Administration, January 1997.

also suggested that the frictional effects associated with the passage of debris can generate high localised temperatures and increase the probability of ignition.

As previously discussed, for fuel vapour to ignite it must not only be at an appropriate temperature, but there must also be a specific mixture of fuel and air. Under the dynamic conditions associated with debris passage, the concentration of the fuel in the vapour near the fragment is higher due to the localised evaporation of the droplets by the hot fragment.

Whether the fuel tank becomes explosive following ignition is dependent upon the amount of fuel and air ready for immediate combustion, as well as the penetration hole size. An increased perforation area will result in a reduced pressure rise and can allow more air/oxygen to enter the tank.

The location at which the fragment enters the tank is also likely to affect whether ignition will occur when the temperature is within the flammability limits. Although not definitive, the research has shown that if the fragment enters the ullage while the vapour is within the flammable range, it is likely that ignition will occur. Conversely, if the fragment enters the ullage after passage through the liquid fuel, there is a potential for ignition. However, one source believed that the probability would be less than when the ullage was entered directly. Lastly, the research showed that should the fragment enter the liquid fuel only, ignition would be unlikely to occur.[151]

Electrical/Spark

A review of the literature indicated that information and contemporary understanding of spark ignition is incomplete. Some studies have shown that conservatively, 0.25 mJ of electrical energy is all that is needed to ignite a typical flammable mixture. The minimum ignition energy (MIE) is dependent on a number of variables and has been shown to be at a minimum when the fuel/air mixture is slightly rich, and higher at increased altitudes due to the decreased oxygen levels.

Examination of the wiring as discussed previously did not show any signs of arcing or overheating associated with electrical currents or short-circuiting.

Ignition prevention on the A380

Airbus advised that a number of design precautions had been taken to minimise the risk of ignition in the A380. The following is a select list of those precautions:

Wiring

- The materials used and the large number of fasteners on the attachment of the metallic structure for the wing and trim tanks ensure that the fuel tank structures are electrically bonded. Additionally, the aircraft’s composite ribs are bonded by metallic strips attached to the non-metallic structure.
- Components and equipment located in the aircraft’s fuel tanks have specified bonding procedures and bonding points to help prevent the build-

[151] NA Moussa, MD Whale, DE Groszmann & XJ Zhang, DOT/FAA/AR-96/95, *The potential for fuel tank fire and hydrodynamic ram from uncontained engine debris,* Federal Aviation Administration, January 1997.

up of static electricity in the structure, and to provide lightning strike protection.

- All metallic structures in the wing, trim tank and centre wing box have a layer of anti-corrosion epoxy primer to protect the main structure against an electrical short between the electrical harnesses and the structure.
- All of the fuel system wiring is insulated to protect against short circuits to other wiring and structure.
- The only wiring in any of the fuel tanks is for the fuel sensing and monitoring systems, which transmit very low energy.

Pumps

- Each pump within the fuel tanks has a pump element, which has an electric motor and is contained in a canister. The element is electrically bonded to the canister. Electrical connections to all pumps are made outside the fuel tank. The moving parts inside the pump are normally kept immersed in fuel so they cannot cause a spark during fault conditions.
- The pumps also have an electronic controller that supplies and controls the electrical voltage to the motor. The controller is kept cool by fuel and contains a thermal cut-out fuse for each phase of the electrical power supply.
- A ground fault interrupter contactor is installed between the power supply busbars and all fuel pumps. This is to remove the power supply to a pump if an intermittent short circuit is made to the structure.
- All the motor-operated valves are installed in the fuel tanks and are dual bonded. All the related actuators are installed outside the tank wall.
- The fuel pump pressure switches are installed on the front and rear spars and are separated from the fuel by a diaphragm. The electrical connectors are fully sealed to give protection from explosion.

Heat sources

- The wing leading edges contain hot air-ducting pipes from the engines to the bleed air and anti-icing systems. These pipes are insulated and remote from the tank boundary. A leak/overheat detection system is installed to sense if a hot air leak occurs near to the tank wall.
- A leak/overheat detection system is installed for the auxiliary power unit (APU) hot air ducting in the tail and fuselage areas.
- Heat-insulating blankets are installed on the front spar of the wing between Rib 1 and Rib 3 to give protection from the heat generated by the air generation unit located in the inner leading edge. A leak/overheat detection system is installed to sense if a hot air leak occurs in this area.

Fuel leaks

- Where leaks occur, fuel from the wing and trim tanks will go to the leading/trailing edge or the outside. If fuel overflows from a NACA[152] intake on the wings, it is directed downwards and away from the engines (heat source) by a fuel leak drip strip that is located inboard of the NACA intakes.
- The equipment installed in the wing leading and trailing edges has explosion protection and is insulated from any fuel leakage.
- The APU feed pipe and trim tank transfer pipe (in the rear fuselage) are both shrouded. If there is a fuel leak from the inner pipes, it is drained overboard through the drain mast.
- The trim tank external surface between Rib 4 left and Rib 4 right, which is inside the rear unpressurized fuselage, is painted with a protective layer of barrier paint. This barrier paint protects against fuel leaks (both vapour and liquid) and is resistant to deterioration from long-term contact with fuel and/or hydraulic fluid.

Fuel tank flammability minimisation – regulatory history

Efforts to reduce fuel tank flammability have been directly related to areas of the fire pyramid. Over time, regulatory bodies have looked at the fuel, heat (and ignition) sources and the fuel/oxygen ratio.

One of the first attempts to reduce fuel tank flammability, as early as the 1950s, was to promote the use of Jet A fuel over Jet B, due to the lower volatility of Jet A. The use of Jet B was subsequently prohibited by the Australian Department of Civil Aviation, and its use is now generally confined to locations where lower air temperatures require the use of higher flash point fuels.[153]

Focus then shifted to identifying potential ignition sources and removing them from fuel tanks. The MIE for hydrocarbon vapours was thought to be about 0.25mJ and thus the electrical energy applied to any component in the fuel tank has been limited to a value that is 10 times lower than this.

Special Federal Aviation Regulation 88 (SFAR 88) was released in 2001. The main goal of SFAR 88 was to preclude ignition sources from fuel tanks, and the safety assessments carried out during the implementation of the rule revealed some unexpected ignition sources. Further assessments and service difficulty reports following the implementation of SFAR 88 showed that ignition sources continued to be identified, with over 20 airworthiness directives subsequently released.

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[152] National Advisory Committee for Aeronautics. The NACA duct is located near the wingtip. As an air inlet, the duct ensures that the differential pressure between the interior of the fuel tanks and the outside atmospheric pressure is kept to a minimum. As a fuel vent system outlet, the duct is sized to permit fuel flow overboard in the event of an automatic refuel failure.

[153] Fuel flammability task group, DOT/FAA/AR-98/26, *A review of the flammability hazard of Jet A fuel vapor in civil transport aircraft fuel tanks*, Federal Aviation Administration, June 1998.

In 1998, a review by the Aviation Rulemaking Advisory Committee (ARAC) of fuel tank ignition events[154] showed that there had been 16 such instances, including eight wing tank and eight centre or fuselage tank events. Five of the eight wing tank events involved the use of Jet B/JP-4 fuel, while all the centre tank events involved the use of Jet A/A-1 fuel. Additionally, the ignition source was known in all the wing tank events, with no known wing tank explosions resulting from internal ignition sources. Historically, most commercial jets utilise integral wing tanks, with only some employing centre/fuselage tanks. As such, experience with the performance of wing fuel tanks is somewhat greater than that relating to centre or fuselage-based tanks.

In summary, conclusions drawn from the fuel tank fire data examined were:

- In general, unheated aluminium wing tanks fuelled with Jet A have exhibited an acceptable service history.
- A wing tank fuelled with JP-4 had approximately the same flammability exposure as an empty heated centre tank with Jet A; these two cases have not had an acceptable service history.

Fuel tank flammability minimisation – recent regulatory activity

Comparatively recent events involving aircraft fuel tank fires (after the introduction of SFAR 88) have lead the US Federal Aviation Administration (FAA) to move towards the implementation of another layer of protection. As such, the most recent work has begun to focus on the flammability of the fuel/air mixture within the tank ullage.

As shown previously, Jet A/A-1 vapours are considered too lean to be flammable at sea level. However, research has shown that at certain times and under certain conditions in the flight cycle, the fuel/air mixture may become flammable. The most recent efforts have been to reduce the flammability exposure level to an acceptable level, either through the inherent design of the tank, or by implementing flammability reduction means (FRM).

FAR 26.33 (effective as of 7 February 2009) stipulated that type certificate holders must perform a flammability exposure analysis of all tanks, except for those meeting the definition of a conventional, unheated aluminium wing tank. The fleet average flammability exposure of fuel tanks that were designed to be normally emptied, and where any portion was within the fuselage contour, could not exceed 3% of the flammability exposure evaluation time (FEET)[155] during the ground, takeoff or climb phases of flight on a warm day.[156] For all other fuel tanks, the fleet average flammability exposure could not exceed 7%.

Airbus advised a flammability analysis for the A380 fuel tanks was prepared and submitted to the FAA to show compliance with FAR 26.33. That analysis

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[154] Aviation Rulemaking Advisory Committee (ARAC), *Service History/Fuel Tank Safety Level Assessment*, Task Group 1, July 1998.

[155] Flammability exposure evaluation time is the time from the start of preparing the aeroplane for flight, through the flight and landing, until all payload is unloaded, and all passengers and crew have disembarked.

[156] Federal Aviation Administration, *Part 26 Continued Airworthiness and safety improvements for transport category airplanes, Subpart D – Fuel tank flammability*, Section 26.33, effective 7/2/09.

concluded that the wing and trim tanks were a low flammability risk (below 7% flammability exposure). As there is no centre wing tank on the A380 aircraft, there was no requirement to develop a flammability reduction system for any of the fuel tanks under the current airworthiness requirements.

ANALYSIS

Left inner tank fire

The witness report/photographs and subsequent testing of surface deposits indicated that a fire occurred within the left inner fuel tank (LIT) moments after a liberated fragment of the intermediate pressure (IP) turbine disc perforated the left wing. The recorded data showed no evidence of fuel loss from the LIT and it was believed that the disc fragment had passed through the ullage space only.

The steady-state conditions within the LIT at the time of the disc failure appeared to be outside the flammability range of Jet A-1 fuel. However, a literature review indicated that ignition was still possible, due to factors such as fuel sloshing, vibration and impact agitation that may have pushed conditions in the tank into the dynamic flammability envelope.

A number of potential sources of ignition were explored. However, the hot IP turbine disc fragment was considered the most likely, due to the proximate location of the soot around the tank perforations and the temperatures required to initiate combustion in Jet A-1 fuel. Friction between the liberated fragment and the tank skin could also have provided an additional, localised heat source. Despite the presence of severed feeder cables and electrical wiring looms that were located around the LIT, there was no evidence of arcing or electrical discharge that might have otherwise provided an ignition source.

It is likely that the fuel mist in the ullage as a result of sloshing and impact was ignited by a large disc fragment as it passed through the fuel tank, particularly as the fragment was significantly hotter than the auto-ignition and hot surface ignition temperatures calculated for Jet A-1 fuel.

Why the fire did not sustain

While the duration of the fire was not able to be ascertained with any level of confidence, the absence of thermal damage to the wing structure, external surfaces, or the tank's internal painted surfaces suggested that the associated temperature rise was relatively low – consistent with a short-lived combustion event. While the witness reported the presence of a fire for approximately 10 minutes duration, it was possible that the observation was actually a fire within the engine structure itself, as the rear of the engine was directly visible from the passenger's seat, through the holes in the wing.

There were a number of possible reasons why the LIT fire was not sustained, including:

- The fire consumed the oxygen inside the tank – reducing it to a level below that required for the combustion of Jet A-1.

- The temperature rise associated with the initial combustion was low and therefore insufficient to increase the temperature of the fuel in the tank to the lower static flammability limit of Jet A-1. Quenching of the flame by the fuel tank walls (as evidenced by the associated sooting) and the entry of cooling air through the tank perforations may have contributed in this regard.
- Conditions settled and became more static within the tank after the fragment’s passage, narrowing the vapour flammability envelope and limiting the potential for continued combustion.
- The damage sustained by the tank restricted any significant internal pressure rise that may have promoted accelerated combustion.

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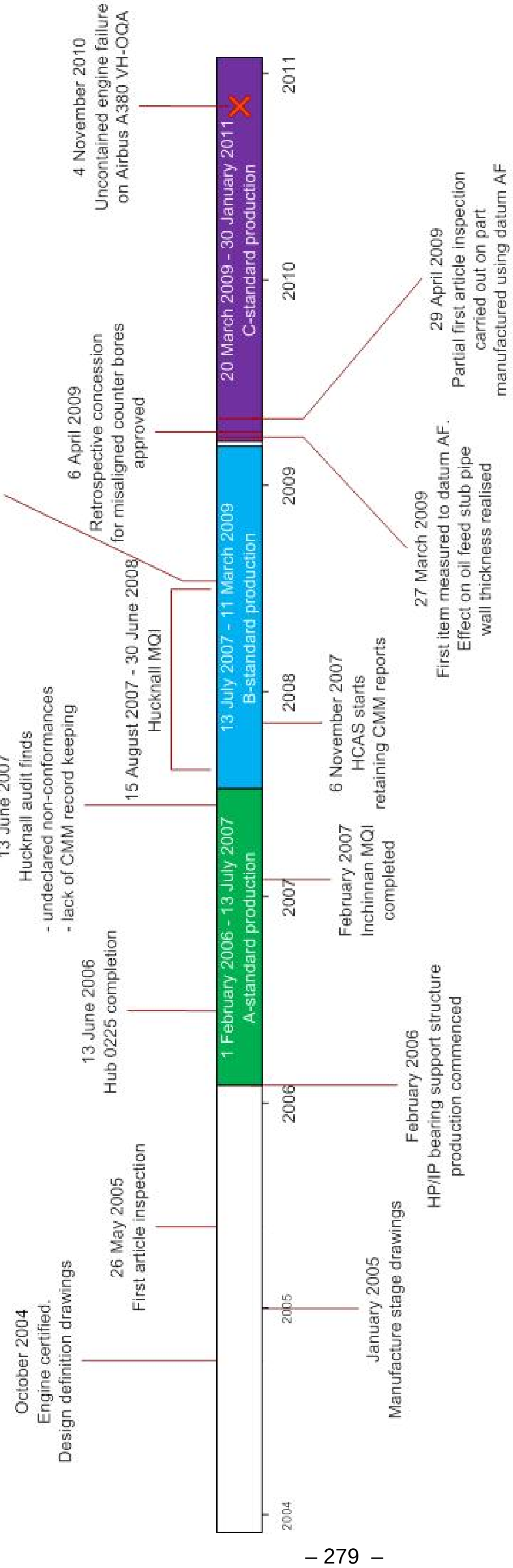
APPENDIX E: KEY EVENTS IN THE MANUFACTURE AND RELEASE OF NON-CONFORMING HP/IP BEARING SUPPORT ASSEMBLIES



APPENDIX F: SOURCES AND SUBMISSIONS

Sources of information

The sources of information during the investigation included the:

- Air Accident Investigation Bureau (AAIB) of Singapore
- Airbus SAS
- Airport Emergency Services of Changi Airport Group (Singapore)
- Australian Federal Police Chemical Criminalistics Laboratory
- Bureau d'Enquêtes et d'Analyses pour la sécurité de l'aviation civile (France) (BEA)
- Civil Aviation Safety Authority (CASA)
- Direction générale de l'Aviation civile (France) (DGCA)
- European Aviation Safety Agency (EASA)
- Flight and cabin crew
- Indonesian National Transportation Safety Committee (NTSC)
- International Civil Aviation Organisation (ICAO)
- Passengers
- Qantas Airways Ltd
- Rolls-Royce plc
- Singapore Air Traffic Control
- United Kingdom Air Accidents Investigation Branch (AAIB)
- United Kingdom Civil Aviation Authority (CAA)
- United States Federal Aviation Administration (FAA)
- United States National Transportation Safety Board (NTSB)

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Submissions

Under Part 4, Division 2 (Investigation Reports), Section 26 of the *Transport Safety Investigation Act 2003* (the Act), the Australian Transport Safety Bureau (ATSB) may provide a draft report, on a confidential basis, to any person whom the ATSB considers appropriate. Section 26 (1) (a) of the Act allows a person receiving a draft report to make submissions to the ATSB about the draft report.

A draft of this report was provided to Qantas Airways Ltd, flight and cabin crew, Airbus SAS, Rolls-Royce plc, relevant manufacturing personnel from Rolls-Royce plc, CASA, Indonesian NTSC, AAIB of Singapore, UK AAIB, UK CAA, BEA and EASA.

Submissions were received from all of those parties. The submissions were reviewed and where considered appropriate, the text of the report was amended accordingly.

Australian Transport Safety Bureau

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Investigation

ATSB Transport Safety Report
Aviation Occurrence Investigation

In-flight uncontained engine failure, Airbus A380-842, VH-OQA,
4 November 2010

AO-2010-089

Final – 27 June 2013