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DEPARTMENT OF TRANSPORTATION

Federal Aviation Administration

14 CFR Part 39

[Docket No. FAA-2019-0692; Product Identifier 2018-NE-19-AD; Amendment 39-19735; AD 2019-18-08]

RIN 2120-AA64

Airworthiness Directives; Engine Alliance Turbofan Engines

AGENCY: Federal Aviation Administration (FAA), DOT.

ACTION: Final rule; request for comments.

SUMMARY: The FAA is superseding Airworthiness Directive (AD) 2019-16-04 for all Engine Alliance (EA) GP7270 and GP7277 model turbofan engines. AD 2019-16-04 required a visual inspection of the 1st-stage low-pressure compressor (LPC) rotor assembly, referred to after this as the “engine fan hub assembly,” for damage, a one-time eddy current inspection (ECI) of the engine fan hub blade slot bottom and blade slot front edge for cracks; and removal of parts if damage or defects are found. AD 2019-16-04 also required replacement of the engine fan hub blade lock assembly for certain GP7270 and GP7277 model turbofan engines. This AD, for certain GP7270 and GP7277 model turbofan engines, reduces the compliance time for the initial ECI and requires repetitive ECIs of the engine fan hub blade slot bottom and blade slot front edge for cracks. This AD also retains the visual inspection requirements of the engine fan hub assembly for all GP7270 and GP7277 model turbofan engines. This AD was prompted by an uncontained failure of the engine fan hub. The FAA is issuing this AD to address the unsafe condition on these products.

DATES: This AD is effective October 9, 2019.

The Director of the Federal Register approved the incorporation by reference of a certain publication listed in this AD as of October 9, 2019.

The FAA must receive any comments on this AD by November 8, 2019.

ADDRESSES: You may send comments, using the procedures found in 14 CFR 11.43 and 11.45, by any of the following methods:

Federal eRulemaking Portal: Go to http://www.regulations.gov. Follow the instructions for submitting comments.

Fax: 202-493-2251.

Mail: U.S. Department of Transportation, Docket Operations, M-30, West Building Ground Floor, Room W12-140, 1200 New Jersey Avenue SE, Washington, DC 20590.

Hand Delivery: U.S. Department of Transportation, Docket Operations, M-30, West Building Ground Floor, Room W12-140, 1200 New Jersey Avenue SE, Washington, DC 20590, between 9 a.m. and 5 p.m., Monday through Friday, except Federal holidays.

For service information identified in this final rule, contact Engine Alliance, 411 Silver Lane, East Hartford, CT 06118; phone: 800-565-0140; email: help24@pw.utc.com; website: www.engineallianceportal.com. You may view this service information at the FAA, Engine and Propeller Standards Branch, 1200 District Avenue, Burlington, MA 01803. For information on the availability of this material at the FAA, call 781-238-7759. It is also available on the internet at http://www.regulations.gov by searching for and locating Docket No. FAA-2019-0692.

Examining the AD Docket

You may examine the AD docket on the internet at http://www.regulations.gov by searching for and locating Docket No. FAA-2019-0692; or in person at Docket Operations between 9 a.m. and 5 p.m., Monday through Friday, except Federal holidays. The AD docket contains this final rule, the regulatory evaluation, any comments received, and other information. The street address for Docket Operations is listed above. Comments will be available in the AD docket shortly after receipt.

FOR FURTHER INFORMATION CONTACT: Matthew Smith, Aerospace Engineer, ECO Branch, FAA, 1200 District Avenue, Burlington, MA 01803; phone: 781-238-7735; fax: 781-238-7199; email: matthew.c.smith@faa.gov.

SUPPLEMENTARY INFORMATION:

Discussion

The FAA issued AD 2019-16-04, Amendment 39-19707 (84 FR 41617, August 15, 2019), ("AD 2019-16-04"), for all EA GP7270 and GP7277 model turbofan engines. AD 2019-16-04 required a visual inspection of the engine fan hub assembly for damage, a one-time ECI of the engine fan hub blade slot bottom and blade slot front edge for cracks, and removal of parts if damage or defects are found that are outside serviceable limits. AD 2019-16-04 required an independent inspection of the engine fan hub assembly prior to reassembly of the engine fan hub blade lock assembly. AD 2019-16-04 also required replacement of the engine fan hub blade lock assembly for certain serial-numbered GP7270 and GP7277 model turbofan engines. AD 2019-16-04 resulted from the manufacturer's determination that an independent inspection of the fan hub assembly for damage was necessary prior to the reassembly of the engine fan hub blade lock assembly for all EA GP7270 and GP7277 model turbofan engines. The FAA issued AD 2019-16-04 to detect defects, damage, and cracks that could result in an uncontained failure of the engine fan hub assembly.

Actions Since AD 2019-16-04 Was Issued

Since the FAA issued AD 2019-16-04, the manufacturer identified a fatigue crack originating inboard of a blade slot after the manufacturer performed a metallurgical examination of the engine fan hub that was recovered, related to the September 30, 2017 event. After performing a risk assessment, the manufacturer determined the need to reduce the compliance time for the initial ECI and add a repetitive ECI. The FAA is issuing this AD to address the unsafe condition on these products.

Related Service Information Under 1 CFR Part 51

The FAA reviewed EA Alert Service Bulletin (ASB) EAGP7-A72-389, Revision No. 5, dated August 23, 2019. The ASB describes procedures for ECI of the EA GP7270 and GP7277 model turbofan engines fan hub assembly. This service information is reasonably available because the interested parties have access to it through their normal course of business or by the means identified in the ADDRESSES section.

Other Related Service Information

The FAA reviewed EA ASB EAGP7-A72-418, Revision No. 1, dated January 11, 2019. The ASB provides guidance on replacement or modification of the engine fan hub blade lock assembly.

The FAA also reviewed the following service information:

Subtask 72-31-42-210-001-A, of Task 72-31-42-000-802-A, from the A380 Aircraft Maintenance Manual (AMM). This subtask describes an on-wing visual inspection that is to be performed after removal of the engine fan hub blade lock assembly.

Figure 405 of Task 72-00-31-420-004 of the EA GP7000 Series Engine Manual (EM). This figure and task describe a visual inspection that is to be performed after removal of the engine fan hub blade lock assembly when the engine is in the shop.

Subtask 72-00-00-210-012-A, of Task 72-00-00-210-806-A, from the A380 Aircraft Maintenance Manual (AMM). This subtask describes an on-wing visual inspection that is to be performed after reassembly of the engine fan hub blade lock assembly.

Task 72-00-31-420-004, Paragraph 1.E.(13), of the GP7000 Series EM describes a visual inspection that is to be performed after reassembly of the engine fan hub blade lock assembly when the engine is in the shop.

Table 601 in Subtask 72-00-00-210-012-A, Task 72-00-00-210-806, from the A380 AMM or Task 72-00-31-220-010 of the EA GP7000 Series EM. Table 601 and Task 72-00-31-220-010 provide guidance on acceptable damage service limits.

FAA's Determination

The FAA is issuing this AD because all the relevant information was evaluated and the FAA determined the unsafe condition described previously is likely to exist or develop in other products of the same type design.

AD Requirements

This AD requires, for certain GP7270 and GP7277 model turbofan engines, an initial and repetitive ECI of the engine fan hub blade slot bottom and blade slot front edge for cracks. For all GP7270 and GP7277 model turbofan engines, this AD also requires an independent inspection of the engine fan hub assembly prior to the reassembly of the engine fan hub blade lock assembly and a visual inspection of the engine fan hub assembly for damage. For certain serial-numbered GP7270 and GP7277 model turbofan engines, this AD requires replacement of the engine fan hub blade lock assembly with a part eligible for installation.

FAA's Justification and Determination of the Effective Date

No domestic operators use this product. Therefore, the FAA finds good cause that notice and opportunity for prior public comment are unnecessary. In addition, for the reason stated above, the FAA finds that good cause exists for making this amendment effective in less than 30 days.

Comments Invited

This AD is a final rule that involves requirements affecting flight safety, and the FAA did not provide you with notice and an opportunity to provide your comments before it becomes effective. However, the FAA invites you to send any written data, views, or arguments about this final rule. Send your comments to an address listed under the ADDRESSES section. Include the docket number

FAA-2019-0692 and product identifier 2018-NE-19-AD at the beginning of your comments. The FAA specifically invites comments on the overall regulatory, economic, environmental, and energy aspects of this final rule. The FAA will consider all comments received by the closing date and may amend this final rule because of those comments.

The FAA will post all comments received, without change, to http://www.regulations.gov, including any personal information you provide. The FAA will also post a report summarizing each substantive verbal contact received about this final rule.

Regulatory Flexibility Act

The requirements of the Regulatory Flexibility Act (RFA) do not apply when an agency finds good cause pursuant to 5 U.S.C. 553 to adopt a rule without prior notice and comment. Because FAA has determined that it has good cause to adopt this rule without notice and comment, RFA analysis is not required.

Costs of Compliance

The FAA estimates that this AD affects zero engines installed on airplanes of U.S. registry. We have revised the estimate of work hours to complete the ECI based on updated service information.

The FAA estimates the following costs to comply with this AD:

Estimated Costs

Action	Labor cost	Parts cost	Cost per product	Cost on U.S. operators
ECI	20 work-hours × $85 per hour = $1,700	$0	$1,700	$0
Visual inspection	1 work-hour × $85 per hour = $85	0	85	0
Replace fan hub blade lock assembly	25 work-hours × $85 per hour = $2,125	28,000	30,125	0

The FAA estimates the following costs to do any necessary replacements that would be required based on the results of the inspection. The FAA has no way of determining the number of engines that might need these replacements:

On-Condition Costs

Action	Labor cost	Parts cost	Cost per product
Replace engine fan hub assembly	50 work-hours × $85 per hour = $4,250	$790,500	$794,750

Authority for This Rulemaking

Title 49 of the United States Code specifies the FAA's authority to issue rules on aviation safety. Subtitle I, Section 106, describes the authority of the FAA Administrator. Subtitle VII, Aviation Programs, describes in more detail the scope of the Agency's authority.

The FAA is issuing this rulemaking under the authority described in Subtitle VII, Part A, Subpart III, Section 44701, "General requirements." Under that section, Congress charges the FAA with promoting safe flight of civil aircraft in air commerce by prescribing regulations for practices, methods, and procedures the Administrator finds necessary for safety in air commerce. This regulation is within the scope of that authority because it addresses an unsafe condition that is likely to exist or develop on products identified in this rulemaking action.

This AD is issued in accordance with authority delegated by the Executive Director, Aircraft Certification Service, as authorized by FAA Order 8000.51C. In accordance with that order, issuance of ADs is normally a function of the Compliance and Airworthiness Division, but during this transition period, the Executive Director has delegated the authority to issue ADs applicable to engines, propellers, and associated appliances to the Manager, Engine and Propeller Standards Branch, Policy and Innovation Division.

Regulatory Findings

This AD will not have federalism implications under Executive Order 13132. This AD will not have a substantial direct effect on the States, on the relationship between the national government and the States, or on the distribution of power and responsibilities among the various levels of government.

For the reasons discussed above, I certify that this AD:

(1) Is not a "significant regulatory action" under Executive Order 12866, and

(2) Will not affect intrastate aviation in Alaska.

List of Subjects in 14 CFR Part 39

Air transportation, Aircraft, Aviation safety, Incorporation by reference, Safety.

Adoption of the Amendment

Accordingly, under the authority delegated to me by the Administrator, the FAA amends part 39 of the Federal Aviation Regulations (14 CFR part 39) as follows:

PART 39–AIRWORTHINESS DIRECTIVES

1. The authority citation for part 39 continues to read as follows:

Authority: 49 U.S.C. 106(g), 40113, 44701.

§ 39.13 [Amended]

2. The FAA amends § 39.13 by removing Airworthiness Directive (AD) AD 2019-16-04, Amendment 39-19707 (84 FR 41617, August 15, 2019), and adding the following new AD:

2019-18-08 Engine Alliance: Amendment 39-19735; Docket No. FAA-2019-0692; Product Identifier 2018-NE-19-AD.

(a) Effective Date

This AD is effective October 9, 2019.

(b) Affected ADs

This AD replaces AD 2019-16-04, Amendment 39-19707 (84 FR 41617, August 15, 2019) (“AD 2019-16-04”).

(c) Applicability

This AD applies to all Engine Alliance (EA) GP7270 and GP7277 model turbofan engines.

(d) Subject

Joint Aircraft System Component (JASC) Code 7230, Turbine Engine Compressor Section.

(e) Unsafe Condition

This AD was prompted by an uncontained failure of the engine fan hub. The FAA is issuing this AD to detect defects, damage, and cracks that could result in an uncontained failure of the engine fan hub assembly. The unsafe condition, if not addressed, could result in uncontained failure of the engine fan hub assembly, damage to the engine, and damage to the airplane.

(f) Compliance

Comply with this AD within the compliance times specified, unless already done.

(g) Required Actions

(1) For EA GP7270 and GP7277 model turbofan engines with engine fan hub assembly part numbers (P/Ns) 5760221 or 5760321, within 1,700 cycles since new, or within 150 flight cycles (FCs) after the effective date of this AD, or within 330 FCs since an eddy current inspection (ECI) was performed in accordance with the Accomplishment Instructions, For Fan Hubs at LPC Module Assembly Level, paragraphs 2.A and 2.B, of EA ASB EAGP7-A72-389, Revision No. 4, dated June 14, 2019, or earlier versions of that ASB; or within 330 FCs since overhaul, whichever occurs later:

(i) For engine fan hub assemblies at the low-pressure compressor (LPC) module assembly level, perform an ECI of the engine fan hub blade slot bottoms and front edges in accordance with the Accomplishment Instructions, For Fan Hubs at LPC Module Assembly Level, paragraphs 1.B. and 1.C., of EA ASB EAGP7-A72-389, Revision No. 5, dated August 23, 2019.

(ii) For engine fan hub assemblies at the piece part level, perform an ECI of the engine fan hub blade slot bottoms and front edges, in accordance with the Accomplishment Instructions, For Fan Hubs at Piece Part Level, paragraphs 1.A. and 1.B., of EA ASB EAGP7-A72-389, Revision No. 5, dated August 23, 2019.

(iii) For engine fan hub assemblies installed in an engine (on-wing or off-wing), perform an ECI of the engine fan hub blade slot bottoms and front edges, in accordance with the Accomplishment Instructions, For Fan Hubs Installed in an Engine, paragraphs 3.B. and 3.C., of EA ASB EAGP7-A72-389, Revision No. 5, dated August 23, 2019.

(iv) Thereafter, repeat the ECI of the engine fan hub blade slot bottoms and front edges at intervals not exceeding 330 FCs since the previous ECI required by paragraphs (g)(1)(i) through (iii) of this AD, as applicable.

(v) If any ECI of the engine fan hub assembly results in a rejectable indication per the Appendix, Added Data, of EA ASB EAGP7-A72-389, Revision No. 5, dated August 23, 2019, remove the engine fan hub assembly from service and, before further flight, replace with a part that is eligible for installation.

(2) For all GP7270 and GP7277 model turbofan engines, after the effective date of this AD:

(i) At the next disassembly of the engine fan hub blade lock assembly, visually inspect the following areas for damage:

(A) The fan hub blade lock retention hooks (also known as lock ring contact area); and

(B) The fan hub rim face.

(ii) At the next reassembly of the fan hub blade lock assembly, visually inspect the following areas of the engine fan hub for damage:

(A) The fan hub scallop areas;

(B) The fan hub bore area behind the balance flange;

(C) The fan hub fan blade lock retention hooks;

(D) The fan hub rim face; and

(E) The clinch nut holes.

(iii) After any reassembly per paragraph (g)(2)(ii), before further flight, perform an independent inspection of all areas of the engine fan hub referenced in paragraph (g)(2)(ii) of this AD for damage.

(iv) Thereafter, repeat the inspections required by paragraphs (g)(2)(i) through (iii) of this AD at each disassembly and reassembly of the engine fan hub blade lock assembly.

(v) As an optional terminating action to the inspection requirements and independent inspection requirements of paragraph (g)(2)(i) through (iii) of this AD, insert the requirements for the visual inspections and independent inspections required by these paragraphs as Required Inspection Items in the approved continuous airworthiness maintenance program for the airplane.

(vi) If damage is found outside serviceable limits during the inspections required by (g)(2)(i) through (iii) of this AD, before further flight, remove the engine fan hub assembly from service and replace it with a part eligible for installation.

(3) For GP7270 and GP7277 model turbofan engines with engine serial numbers P550101 through P550706, remove the engine fan hub blade lock assembly, P/N 5700451, by September 1, 2020, and replace with a part eligible for installation. Refer to EA ASB EAGP7-A72-418, Revision No. 1, dated January 11, 2019, for guidance on replacement of the engine fan hub blade lock assembly.

(h) Credit for Previous Actions

You may take credit for the inspections required by paragraph (g)(1)(i) through (iii) of this AD if you performed the inspections before the effective date of this AD using EA ASB EAGP7-A72-389, Revision No. 4, dated June 14, 2019, or an earlier version.

(i) Definitions

(1) For the purpose of this AD, a part eligible for installation for replacement of the engine fan hub blade lock assembly is:

(i) A part that is not P/N 5700451, or

(ii) An engine fan hub blade lock assembly that has been modified in accordance with EA ASB EAGP7-A72-418, Revision No. 1, dated January 11, 2019, or EA ASB EAGP7-A72-418, Revision No. 0, dated December 7, 2018.

(2) For the purpose of this AD, an independent inspection is a second visual inspection performed by an individual qualified to perform inspections who was not involved in the original inspection of the engine fan hub assembly following disassembly and reassembly of the engine fan hub blade lock assembly.

(j) Alternative Methods of Compliance (AMOCs)

(1) The Manager, ECO Branch, FAA, has the authority to approve AMOCs for this AD, if requested using the procedures found in 14 CFR 39.19. In accordance with 14 CFR 39.19, send your request to your principal inspector or local Flight Standards District Office, as appropriate. If sending information directly to the manager of the certification office, send it to the attention of the person identified in paragraph (k) of this AD. You may email your request to: ANE-AD-AMOC@faa.gov.

(2) Before using any approved AMOC, notify your appropriate principal inspector, or lacking a principal inspector, the manager of the local flight standards district office/certificate holding district office.

(3) AMOCs approved for AD 2019-16-04, AD 2018-11-16 (83 FR 27891, June 15, 2018), and AD 2019-03-04 (84 FR 4694, February 19, 2019) are approved as AMOCs for the corresponding provisions of this AD.

(k) Related Information

For more information about this AD, contact Matthew Smith, Aerospace Engineer, ECO Branch, FAA, 1200 District Avenue, Burlington, MA 01803; phone: 781-238-7735; fax: 781-238-7199; email: matthew.c.smith@faa.gov.

(l) Material Incorporated by Reference

(1) The Director of the Federal Register approved the incorporation by reference (IBR) of the service information listed in this paragraph under 5 U.S.C. 552(a) and 1 CFR part 51.

(2) You must use this service information as applicable to do the actions required by this AD, unless the AD specifies otherwise.

(i) Engine Alliance (EA) Alert Service Bulletin EAGP7-A72-389, Revision No. 5, dated August 23, 2019.

(ii) [Reserved]

(3) For EA service information identified in this AD, contact Engine Alliance, 411 Silver Lane, East Hartford, CT 06118; phone: 800-565-0140; email: help24@pw.utc.com; website: www.engineallianceportal.com.

(4) You may view this service information at the FAA, Engine & Propeller Standards Branch, 1200 District Avenue, Burlington, MA 01803. For information on the availability of this material at the FAA, call 781-238-7759.

(5) You may view this service information that is incorporated by reference at the National Archives and Records Administration (NARA). For information on the availability of this material at NARA, email: fedreg.legal@nara.gov, or go to: http://www.archives.gov/federal-register/cfr/ibr-locations.html.

Issued in Burlington, Massachusetts, on September 18, 2019.

Karen M. Grant,

Acting Manager, Engine & Propeller Standards Branch,

Aircraft Certification Service.

BEA
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INVESTIGATION REPORT

Accident to the AIRBUS A380-861
equipped with Engine Alliance GP7270 engines
registered F-HPJE
operated by Air France
on 30 September 2017
in cruise over **Greenland (Denmark)**

RÉPUBLIQUE
FRANÇAISE
Liberté
Égalité
Fraternité

BEA2017-0568.en/**September 2020**

SAFETY INVESTIGATIONS

The BEA is the French Civil Aviation Safety Investigation Authority. Its investigations are conducted with the sole objective of improving aviation safety and are not intended to apportion blame or liabilities.

BEA investigations are independent, separate and conducted without prejudice to any judicial or administrative action that may be taken to determine blame or liability.

SPECIAL FOREWORD TO ENGLISH EDITION

This is a courtesy translation by the BEA of the Final Report on the Safety Investigation. As accurate as the translation may be, the original text in French is the work of reference.

Contents

GLOSSARY

Abbreviation	English version
A/P	AutoPilot
AC	Advisory Circular (FAA)
AD	Airworthiness Directive (EASA)
ADCN	Avionics Data Communication Network
ADS-C	Automatic Dependent Surveillance – Contract (*Automatic data reports from the onboard navigation and position calculation equipment sent by the aeroplane to the ground system*)
AIB-DK	Accident Investigation Board Denmark
AMC	Acceptable Means of Compliance
AMS	Aircraft condition Monitoring System
ANSU	Aircraft Network Server Unit
ARP	Airport Reference Point
ASB	Alert Service Bulletin
ATC	Air Traffic Control
ATP	Acceptance Test Protocol
CAS	Computed Air Speed
CC	*Chef de Cabine* (Purser)
CCA	Cabin Crew Attestation
CCO Air France	Operational Control Centre
CCP	*Chef de Cabine Principale* (Chief Purser)
CMS	Centralized Maintenance System
CoBP	*Compresseur Basse Pression* (Low pressure compressor)
CPDLC	Controller-Pilot Data Link Communication (*Written messages between crew and controller, notably clearances and requests*)
CS	Certification Specifications (EASA)
CU	Cockpit Unit (*Unit of vibration felt in cockpit*)
CVR	Cockpit Voice Recorder
DGAC	*Direction Générale de l'Aviation Civile* (French civil aviation authority)
EA	Engine Alliance

Abbreviation	English version
EASA	European Aviation Safety Agency
EBSD	Electron BackScatter Diffraction
ECAM	Electronic Centralized Aircraft Monitoring
ECI	Eddy Current Inspection
EEC	Electronic Engine Control
EO	Engine Out
ETOPS	Extended Twin OPerationS
EVMU	Engine Vibration Monitoring Unit
FAA	Federal Aviation Administration
FADEC	Full Authority Digital Engine Control
FBO	Fan Blade Off
FC	Flight Cycle
FCOM	Flight Crew Operating Manual
FCTM	Flight Crew Techniques Manual
FD	Flight Director
FDR	Flight Data Recorder
FH	Flight Hour
FL	Flight Level
FMS	Flight Management System
FMU	Fuel Metering Unit
FO	First Officer
FOR-DEC	Facts, Options, Risks & benefits, Decide, Execution, Check
GE Aviation	General Electric Aviation
GEUS	Geological Survey of Denmark and Greenland
HGG	HydroGeophysics Group
IATA	International Air Transport Association
ICAO	International Civil Aviation Organization
IFR	Instrument Flight Rules
LCF	Low Cycle Fatigue
LP	Low Pressure
MCT	Max Continuous Thrust

Abbreviation	English version
MTR	Micro Texture Region
N1	Low pressure compressor and turbine rotation speed
N2	High pressure compressor and turbine rotation speed
NTSB	National Transportation Safety Board
NVM	Non Volatile Memory
ONERA	*Office National d'Études et de Recherches Aérospatiales* (French Aerospace Research and Design Office)
P&W	Pratt & Whitney
PF	Pilot Flying
PFD	Primary Flight Display
PM	Pilot Monitoring
PN	Part Number
REP	Aircraft system REPort
SAR	Smart Access Recorder
SB	Service Bulletin
SEM	Scanning Electron Microscope
TCAS	Traffic Collision Avoidance System
TSB Canada	Transportation Safety Board of Canada
UTC	Coordinated Universal Time
VFR	Visual Flight Rules

Synopsis

Time	At 13: 49[1]
Operator	Air France
Type of flight	Commercial air transport (passengers)
Persons onboard	Captain (initially PM then PF); First officer 2 (PF then PM); First officer 1 (relief pilot); 21 cabin crew; 497 passengers
Consequences and damage	RH outer (No 4) engine substantially damaged, adjacent structure slightly damaged

[1] Except where otherwise indicated, the times in this report are in Coordinated Universal Time (UTC). Three hours should be deducted to obtain the time in Greenland or at Goose Bay on the day of the event.

Uncontained failure of engine No 4 en route, diversion

On Saturday, 30 September 2017, the Airbus A380-861 operated by Air France, was carrying out scheduled flight AF066 from Paris (France) to Los Angeles (USA). It took off at 09:50. At 13:49, while the crew were changing en-route flight level, they heard an explosion and observed asymmetric thrust from the right side of the aeroplane, immediately followed by severe vibrations. The "*ENG 4 STALL*" and then the "*ENG 4 FAIL*" messages nearly simultaneously appeared on the ECAM. The crew diverted to Goose Bay airport (Canada) where they landed at 15:42 without any further incident.

A visual examination of the engine found that the fan, first rotating assembly at the front of the engine, along with the air inlet and fan case had separated in flight leading to slight damage to the surrounding structure of the aircraft.

The factors likely to have contributed to the accident include:

- engine designer's/manufacturer's lack of knowledge of the cold dwell fatigue phenomenon in the titanium alloy, Ti-6-4;
- absence of instructions from the certification bodies about taking into account macro-zones (i.e. colony of similarly oriented alpha grains) and the cold dwell fatigue phenomenon in the critical parts of an engine, when demonstrating conformity;
- absence of non-destructive means to detect the presence of unusual macro-zones in titanium alloy parts;
- an increase in the risk of having large macro-zones with increased intensity in large Ti-6-4 forgings due to bigger engines, and in particular, bigger fans.

After the accident, regular inspections of the fleet in service found that there were no cracks detected in the areas considered at risk on the fan hubs of the Engine Alliance engines equipping the A380. The certification bodies and engine manufacturers are currently considering how to better understand the cold dwell fatigue phenomenon and take it into account in the design of future engines.

ORGANIZATION OF THE INVESTIGATION

On 30 September 2017, around 19:00, the Air France Operations Control Centre informed the BEA that an Airbus A380 fitted with Engine Alliance GP7270 engines, registered F-HPJE, had diverted to Goose Bay airport (in Canada) after an uncontained failure of one of its engine while en route.

The Transportation Safety Board of Canada (TSB) initially opened a safety investigation and notified the BEA of the occurrence of a serious incident, on the assumption that the occurrence had taken place in their airspace.

On 1 October, four BEA investigators representing France, the state of registry, state of the operator, state of design and state of manufacture of the aeroplane travelled to Goose Bay, accompanied by advisers from Airbus and Air France. An investigator from the American investigation authority (NTSB), state of design and state of manufacture of the engines, accompanied by advisers from the engine manufacturers, General Electric and Pratt & Whitney (forming the Engine Alliance joint venture, engine designer and manufacturer) completed the team led by the Canadian investigators from the TSB. The investigation team were able to access the aeroplane the very next day.

On 2 October, a fifth BEA investigator travelled to Ottawa (TSB head office) in order to attend the read-out of the data recorded in the flight data recorder (FDR) which confirmed that the failure occurred over Greenland. From this point, the AIB DK, in charge of the safety investigations in Denmark, delegated the safety investigation to the BEA in accordance with the provisions of Regulation (EU) No 996/2010 of the European Parliament and of the Council of 20 October 2010 on the investigation and prevention of accidents and incidents in civil aviation. The BEA re-designated the occurrence as an accident. The BEA kept the members of the investigation team and the group structure initially defined and included the AIB DK (representing Greenland and Denmark) as the state of occurrence.

The safety investigation was organized into three working groups in the following fields: Aircraft, Aeroplane systems and Operations. The accredited representatives and advisers were split between these three groups.

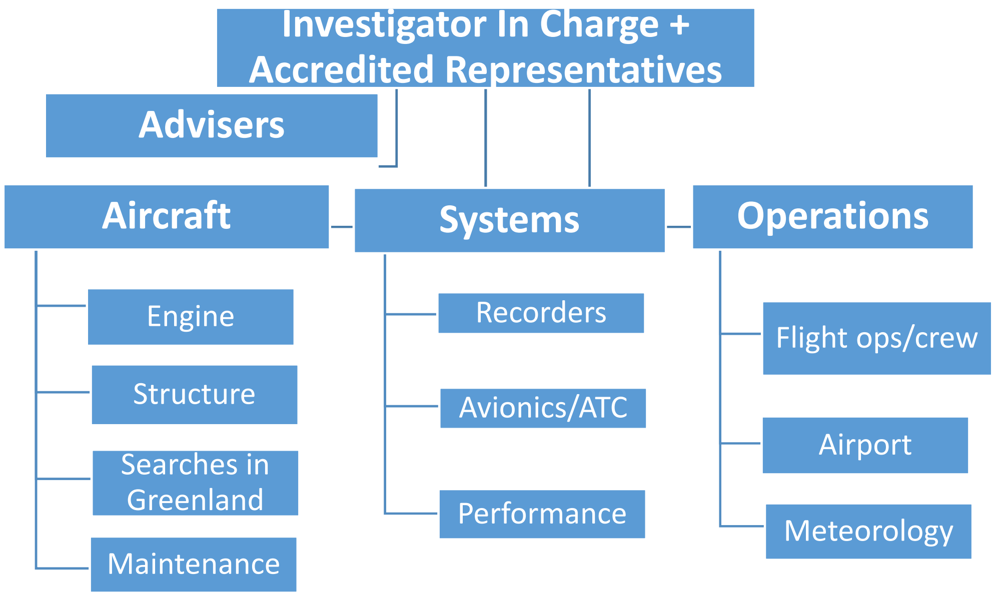


Figure 1: Organization of the investigation

On 3 October, the data contained in the flight data recorder (FDR) was used to determine the path and the precise position of the aircraft when the failure of the right outer engine (engine No 4) occurred, and to define a search zone to locate the parts which had separated from the aeroplane. This zone proved to be a deserted terrain covered with ice, situated approximately 100 km northwest of Narsarsuaq, in the southwestern part of Greenland.

On 4 October 2017, the Danish investigation authority (AIB DK) asked that a helicopter operated by Air Greenland fly over the identified zone (Phase I). Some parts were found and recovered during the three flights made in the following week until snow fall prevented further helicopter flights to the site. Snow finally covered all the parts which were still on the ground, preventing any new visual sightings.

It was determined quite early on in the investigation that the recovery of the missing parts and in particular, the fragments of the fan hub, was essential to establish the circumstances and factors explaining this accident. The use of other detection means was then envisaged. Due to the access difficulties and risks present during the winter (low temperatures, short days, changing weather, presence of crevasses, etc.), the spring of 2018 was the next closest period for contemplating search and recovery operations. After an assessment phase of search means, it was decided to set up two consecutive operations (Phase II):

- an aerial campaign, consisting in the use of synthetic aperture radars operated from an aeroplane, to try to detect and locate the missing parts on the ice sheet, under the layer of snow;
- a ground campaign, consisting in recovering the parts previously located during the aerial campaign, or in performing a systematic search in the zone with the help of ground penetrating radars if the aerial phase was unsuccessful.

Pending this search campaign, the engine manufacturer produced finite element simulations. All the components recovered in Greenland during Phase I were examined in order to understand the failure mechanism should the fragments of the fan hub not be found. A fault tree was produced and two scenarios, considered possible, were retained: that of a material defect (although there was no element confirming this) and that of tool damage during a maintenance operation (considered the most likely in view of the manufacturer’s in-service experience and the result of the inspections of the engines in service launched after the event).

The parts being searched for were not found in phase II. It was therefore decided to start work with a view to an ultimate search phase in the spring of 2019 (phase III), after developing a specific sensor and isolating a limited number of targets by analysing the data from phase II.

This campaign ultimately led to the discovery and extraction of a fragment of the fan hub in July 2019.

The examination of this fragment invalidated the above damage scenario considered as most likely by the engine manufacturer. Instead, it found a failure resulting from the progression of a crack originating in the part's subsurface. The crack origin was situated in a micro-texture region, also known as a macro-zone, in a slot bottom of the hub (under the blade root).

A technical report concerning search phases I and II was published[2], along with a second report describing phase III[3].

[2] https://www.bea.aero/uploads/tx_elyextendttnews/F-HPJE_TECHNICAL_REPORT.pdf

[3] https://www.bea.aero/uploads/tx_elyextendttnews/F-HPJE_Phase_III_PUBLICATION_June_2020.pdf

1 - FACTUAL INFORMATION

1.1 History of the flight

Note 1: The following information is based on the CVR and FDR, statements, radiocommunication recordings and the results of the examinations of engine damaged parts.

Note 2: In the rest of the report, FO/1 designates the first officer flying during the take-off from Paris-Charles-de-Gaulle and FO/2 designates the relief pilot.

On Saturday, 30 September 2017, the Airbus A380-861, powered by Engine Alliance GP7270 engine), registered F-HPJE, operated by Air France, was programmed to carry out scheduled flight AF066 from Paris Charles-de-Gaulle to Los Angeles. Twenty four crew members (3 flight crew and 21 cabin crew) and 497 passengers were present onboard.

The aeroplane took off at 09:50 with the three pilots (the captain and two first officers, FO/1 and FO/2) in the cockpit. The cruise altitude (FL 330) was reached around 25 minutes later. The crew agreed on the division of the rest time. FO/2 took the first duty period around 30 minutes after take-off. The aeroplane changed levels several times during the cruise and then stabilized at FL 370 at 11:14.

Radar coverage is not available for all of the transatlantic phase. At around 100 NM east of the Greenland coast, CPDLC was made with "*Gander Oceanic*"[4] in order to allow the crew to communicate by written messages with the control services. At 13:48, the crew asked to climb to FL 380. The controller accepted and asked them to report when the aeroplane had reached FL 380. The low pressure compressor and turbine rotation speed (N1) of the four engines increased from 98% to 107%[5].

[4] En route control centre in Canada.

[5] The maximum rotation speed accepted in flight (Red Line) is 111%.

At 13:49, the titanium fan hub of the right outer engine (No 4) separated into at least three parts. This failure was the result of the progression of a crack originating in the part's subsurface. The incipient crack was situated in a macro-zone, near a slot bottom of the hub (blade slot). The crack's nucleation and early progression were due to cold dwell fatigue. The central fragment of the hub stayed attached to the coupling shaft between the low pressure compressor and the low pressure turbine. The two other hub fragments were ejected, one upwards and the other downwards. The interaction between the liberated fan rotor fragments and the fixed parts of the engine caused the destruction of the engine casing and the separation of the air inlet which fell to the ground. Debris struck the wing and airframe without affecting the continuation of the flight.

After the failure, the aeroplane's heading increased by three degrees to the right in three seconds, and there were vibrations in the airframe for around four seconds. The crew perceived these variations and associated them with engine surging by analogy with the sensations reproduced in simulator sessions. An "*ENG 4 STALL*" ECAM message came up. The captain requested the "*ECAM actions*". He engaged A/P1 and indicated that he was taking the controls and would thus be PF. He reduced engine No 4 thrust by positioning the associated lever to IDLE. The engine performed an automatic shutdown and the FO/2 confirmed the sequence by depressing the Engine 4 Master and Engine 4 fire pushbuttons, a few seconds later.

The damaged engine could not be seen from the cockpit or in the image from the camera located on the fin of the A380. A member of the cabin crew brought to the cockpit, a photo of the engine taken by a passenger with his smartphone. FO/1 who had returned to the cockpit to help the flight crew on duty, went to the upper deck to assess the damage and take other photos. He observed damage on the leading edge slats and small vibrations in the flaps. The captain started the incident processing method taught by Air France: FOR-DEC (Facts, Options, Risks & benefits, Decide, Execution, Check).

From the time of the failure and for around 1 min 30 s, the CAS had decreased from 277 kt to 258 kt and level flight at FL 370 was maintained. The captain noticed this reduction in speed and decided to descend to the drift-down level calculated by the FMS (EO MAX FL 346) to maintain a constant speed in level flight. Observing that it was not possible to hold this level and this speed, he continued descending level by level. He selected FL 360, FL 350 then FL 330 and lastly FL 310. The level by level descent obliged the crew to stop their ECAM actions each time a descent was initiated. During level flight at FL 310, the N1 rotation speeds of the three remaining engines decreased to 103%. The captain stabilized the descent to FL 290 with a constant speed (CAS was 290 kt) by keeping the three engines in maximum continuous thrust (MCT). He decided to continue the descent and stabilize at FL 270 in order to spare the engines to destination. The speed stabilized at 279 kt. Around five minutes after the A380 had started its descent, the controller in the Gander Oceanic control centre with which the crew were in datalink contact (CPDLC), detected the deviation from the vertical profile of the path and sent a message: "*ATC NOW SHOWS YOU FL330. IS THERE A PROBLEM*".

At the same time, the control centre received an audio Mayday message from AF066, relayed by another aeroplane. One minute later, the PM replied to the CPDLC question with a MAYDAY. Direct audio communication between the aeroplane and ATC resumed a few minutes later on the 132.37 MHz frequency.

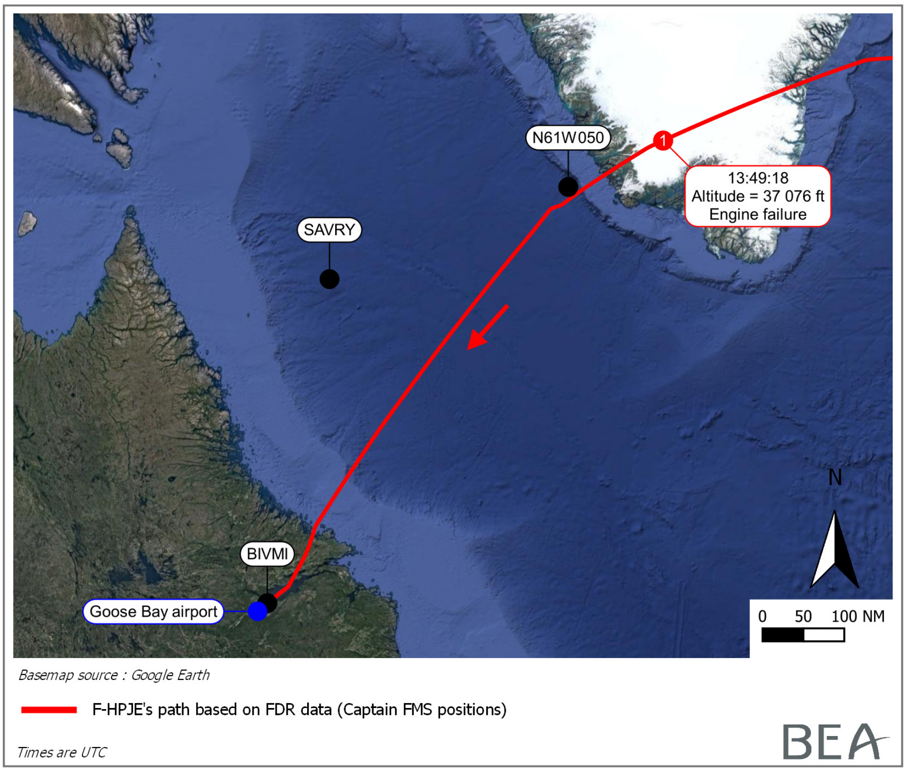


Figure 2: Flight path of aeroplane from engine 4 failure up to landing at Goose Bay

The crew decided, in agreement with the CCO, to divert to Goose Bay airport and asked the controller for a direct route. After studying the available approaches and taking into consideration the captain's experience and the airport's immediate environment, the crew confirmed the selection of Goose Bay airport as the alternate airfield even though it was at a greater distance than Kangerlussuaq airport in Greenland.

The crew started the descent to Goose Bay and were cleared to carry out the RNAV GNSS RWY 26 approach. They were then cleared to land on runway 26. They configured the aeroplane for landing. On approaching the altitude of 1,000 ft, the captain disconnected A/P1 and the flight director (FD) and continued the landing in manual flight. The aeroplane landed at 15:42. The taxiing phase to the stand took some time due to having to stop several times so that the airport services could collect the debris which had fallen onto the runway during the landing. At 16:22, all the engines were shut down. The CVR recording automatically stopped five minutes after the last engine was shut down in accordance with the end of recording logic.

The passengers were looked after by the crew, and airline ground personnel from the Montreal and New York bases. The passengers were not able to leave the airport because their number exceeded the handling capabilities of the airport immigration personnel and the accommodation available at Goose Bay. Certain passengers were able to go into the airport terminal. They were all served a meal on the aeroplane. The captain then saw all the passengers in groups of 50 in order to explain the situation to them. A re-routing solution to Los Angeles was proposed by the company. On 1 October 2017, the last passenger left the aeroplane at 08:10.

1.2 Injuries to persons

There was no corporal injury in this accident.

	Injuries		
	Fatal	**Serious**	**Minor/None**
Crew	-	-	24
Passengers	-	-	497
Others	-	-	-

1.3 Damage to aircraft

The aeroplane touched down without any particular difficulty at Goose Bay airport in Canada (Figure 3).

Figure 3: F-HPJE after landing at Goose Bay

1.3.1 Damage to right outer engine (No 4)

Engine No 4 was damaged (Figure 4): the major part of the fan hub, the majority of the fan blades, the inlet cone, the air inlet (over 2.5 m) and the right and left engine cowlings were missing. The fan case isogrid was missing with the remaining material petaled outward between 4 o'clock and 6 o'clock as well as between 8 o'clock and 12 o'clock[6]. The central conical part of the hub was still attached to the low-pressure turbine shaft. The hub fracture surface was matte with a granular appearance on nearly all of its circumference. The hub was blocked and could not be rotated. Various pieces of debris were found in the engine and on the runway: attachment bolts, pieces of honeycomb, pieces of fan blade and two pieces of the front cone. A fan blade was found embedded in the outlet guide vanes (OGVs). The other damage observed on the engine also seemed to be the result of the failure of the fan hub.

[6] The angular o'clock positions are given aft looking forward.

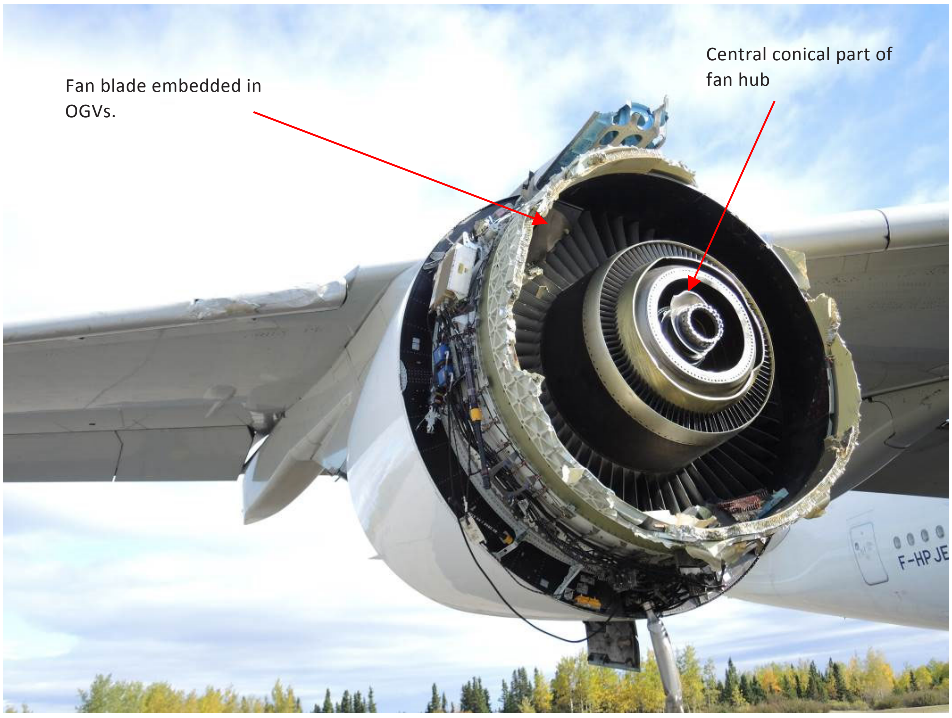


Figure 4: Damaged F-HPJE engine No 4 at Goose Bay

1.3.2 Wing damage

Several marks and deformations were observed on the wing and on the nacelle pylon of engine No 4, resulting from contact with the parts of the engine which separated in flight.

A few marks and deformations were also observed on the leading edge slats, wing flaps, ailerons and on the fairings of the flap rails both sides of the engine as well as on the trimmable horizontal stabilizer.

No debris penetrated the cabin.

1.4 Other Damage

Not applicable.

1.5 Personnel Information

The flight crew were composed of a captain and two first officers. The pilots had never flown together before. During the take-off, FO/1 was PF and the captain was PM. After the failure, the captain became PF and FO/2 became PM. FO/1 who was in the crew rest station joined them in the cockpit to help them.

1.5.1 Flight Crew

Licence, rating, training and checks

	Captain	First Officer 1	First Officer 2
Sex, age, nationality	Male, aged 60, French.	Male, aged 45, French.	Male, aged 42, French.
ATPL issued on	14 May 2008 by validation of PL of 1989	8 April 2005	27 July 2005
First Type Rating (TR) on A380	2011	2016	2017
A380 TR valid until	30 September 2018	31 July 2018	30 June 2018
Class 1 medical check-up valid until	24 April 2018	31 December 2017	30 April 2018

Experience

	Captain	First Officer 1	First Officer 2
Total experience (FH)	19,568 of which 15,260 as captain	8,549	8,811
On type A380 (FH)	3,249 of which 3,249 as captain	796	260
Previous three months (FH)	120	181	192
Previous 30 days (FH)	59	71	50
Previous 72 hours (FH)	4	3.5	0.5

Aviation career

	Captain	First Officer 1	First Officer 2
Employed as	Captain on the B737 from 1995 to 2001, on the A340 from 2001 to 2006, on A330 from 2005 to 2006 and on the B747 from 2006 to 2011 and then on A380	FO on the A320 from 2002 to 2007 and FO on the B747 from 2007 to 2016	FO on the A320 from 2002 to 2007 and FO on the B747 from 2007 to 2012 and the B777 from 2012 to 2017
SUP NIV PRO[7]	4 August 2017	-	-
TR renewal and proficiency check (C1)[8]	15 August 2017	22 July 2017	18 April 2017
E1 training[9]	16 August 2017	-	-
4S module training[10]	21 September 2017	-	-
Ground recurrent training	-	31 March 2017	31 March 2017
Line check	-	6 August 2017	9 June 2017

[7] Simulator session compensating for the absence of recent experience (used to carry out take-offs and landings).

[8] Renewal of Instrument Rating and annual proficiency check simulator session.

[9] Session followed subsequent to C1 in order to rework certain skills seen during C1 check.

[10] Annual ground training day devoted to flight safety.

1.5.2 Cabin crew

	Chief purser	Purser 1	Purser 2
Sex, age	Female, aged 59	Male, aged 53	Female, aged 47
Date of CCA	8 September 1983	2 January 1990	21 May 1991
Purser/chief purser since	2010	2005	2002
Date of conversion training	29 September 2016	21 September 2016	23 November 2009
Date of recurrent training on the A380	5 July 2017	Nil	21 February 2017
Date of medical check-up	2 May 2016	30 May 2017	12 October 2016
Total experience: in flight hours.	20,398	14,726	8,054
On type A380: in flight hours.	227	7	1,322

At the time of the occurrence, the chief purser was resting and purser 1 was carrying out the chief purser's duties. He added that holding this position with such little experience on the aeroplane was difficult.

1.5.3 Flight Crew Techniques Manual (FCTM)

The purpose of the FCTM is to define a way of comprehending normal or abnormal situations when they do not occur according to the principle of the procedures read or they do not fall within the scope of known procedures for which a sense of urgency or an aeroplane particularity is specifically defined. The FCTM provides the crew with the general operational philosophy specific to Airbus (for example, design and utilization principles, golden rules for pilots), additional information to the FCOM such as why and how it is to be done, best practices, operating techniques on manoeuvres, and handling as well as information about situational awareness. The Airbus and Air France A380-861 FCTMs are identical.

1.5.4 Flight Crew Operating Manual (FCOM)

The FCOM is the support documentation for flight crews. It provides them with all the necessary information about the operational, technical, procedural and performance characteristics of the A380 to ensure the safe and effective operation of the aeroplane during the normal, abnormal and emergency situations which could occur on the ground or in flight. This manual is not designed to supply basic aircraft piloting techniques or information considered as basic principles for trained flight crews who know this type of aeroplane and its general piloting characteristics.

The FCOM is designed to be used:

- directly as an operating manual for the flight crew;
- as a complete reference guide during initial and refresher training of flight crews.

1.5.5 Decision making method

FOR-DEC (Facts, Options, Risks & benefits, Decide, Execution, Check) is the decision making method chosen by Air France to be systematically applied in the event of a failure or any other irregularity. It starts at the assessment step to process a failure, or on an irregularity being announced.

1.5.6 Descent strategy

There are two directives in the FCOM in the event of an engine failure in cruise. The first directive, the "*standard strategy*" applies if there is no obstacle to avoid on the path. If this is not the case, the "*obstacle strategy*" must be applied.

Two V speeds are stipulated: the greendot speed and the long range cruise speed. The greendot speed permits the best climb performance as it corresponds to the best lift/drag ratio. This speed also guarantees the shallowest possible descent gradient if obstacles have to be avoided during the descent. The long range cruise speed is defined as the Mach for which the specific flight range is 99% of the maximum flight range. The FMS calculates the EO MAX (Engine Out Maximum flight level) to maintain level flight at the long range cruise speed with the engines in MCT and the failed engine windmilling. The FMS PERF page indicates the EO MAX FL in the right upper corner. The EO MAX FL is the driftdown ceiling which does not take into account a possible deterioration of the aerodynamic characteristics.

The FCOM standard strategy instructions recommend initially not to decelerate to below the greendot speed, to set the engines to MCT, disengage the A/THR and then determine the EO MAX FL via the FMS performance pages. The descent must be carried out at Mach 0.85 or 300 kt until EO MAX FL is reached.

There can be several reasons why the actual driftdown level is different to that specified by the FMS: degraded drag or lift due to severe damage, degraded engine thrust or icing.

1.5.7 Choice of descent speed

When an engine failure occurs at an altitude above EO MAX and the aeroplane is climbing, the CLB mode changes to the OP CLB or V/S mode; the target speed decreases at a rate of 1 kt/s, from the actual speed to the greendot speed with one engine out (EO-GDOT).

The climb from FL 370 to FL 380 in CLB mode had been started with a managed target speed of 277 kt. After the failure, the CAS decreased, the OP CLB mode was engaged and the managed target speed decreased to 263 kt. The theoretical greendot speed was calculated in the conditions of the occurrence based on the information in the FCOM performance section. The greendot speed at FL 370 was 262 kt. This calculation does not take into account the hypothesis of additional drag.

Two minutes after the failure, the speed was no longer managed, the selected speed was 270 kt and it remained at this value during the level by level descent until approaching FL 290.

AIRFRANCE

380-861
FLIGHT CREW
OPERATING MANUAL

PROCEDURES

ABNORMAL AND EMERGENCY PROCEDURES

ECAM ABNORMAL AND EMERGENCY PROCEDURES - ALL ECAM PROCEDURES

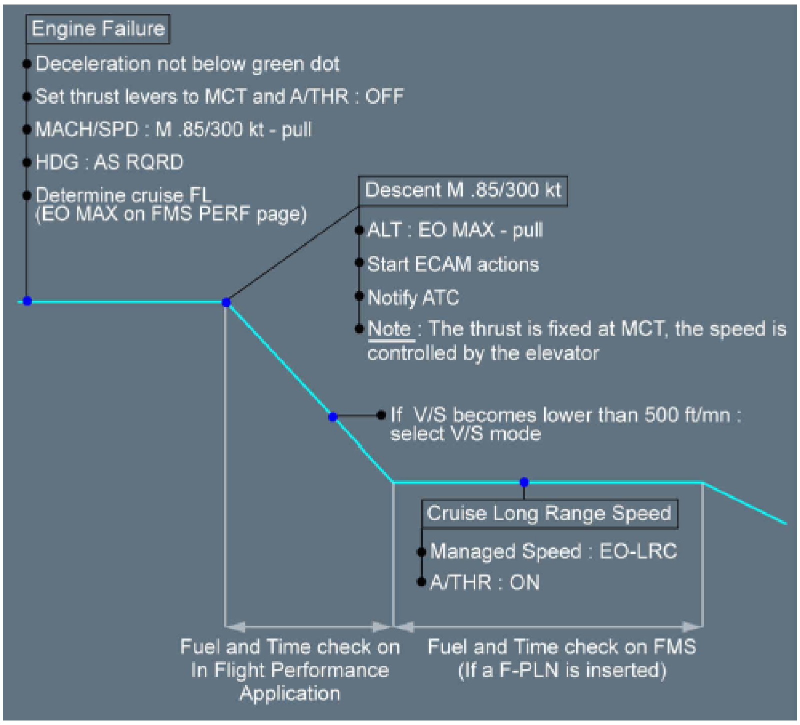
ENG 1(2)(3)(4) FAIL (Cont'd)

Ident.: PRO-ABN-ECAM-10-70-10-00017001.0002001 / 14 SEP 17

ENGINE FAILURE DURING CRUISE

In cruise, the flight crew may apply either:

- The standard strategy:



- Or, the obstacle strategy:

Continued on the following page

AFR 380-861 For A/C: ALL 380 FCOM ← H → PRO-ABN-ECAM-10-70 P 13/112 14 SEP 17

The manufacturer's calculations during the investigation showed, taking into consideration the estimated additional drag caused by the damage due to the failure, that the minimum descent gradient from FL 350 would have been reached for an ideal theoretical CAS speed of 270 kt-273 kt.

1.5.8 Performance

The failure of engine 4 being uncontained, the crew were surprised by the amount of additional drag which degraded the aeroplane's performances. In particular, during the interview, the crew mentioned the difference between the theoretical driftdown indication with one engine out (FL 346) and the final driftdown level (FL 270).

The descent was carried out level by level from FL 370, with the aim of holding the altitude at the selected speed of 270 kt.

Airbus determined that the long range cruise speed in the conditions of the flight at the time of the engine failure was 288.6 kt. The EO MAX FL associated with this speed was 33,961 ft (FL 340). This calculation does not include the additional drag due to the uncontained failure of an engine.

Whether at FL 350 or FL 330, levels close to the EO MAX FL (FL 346 expected), the selected speed of 270 kt could not be held with the three remaining engines in MCT. At FL 290, the ratings were 103% to maintain a speed of 290 kt, which led the captain to choose FL 270 at a speed of 279 kt in order not to continue demanding excessive power.

The greendot speed calculated with the estimated additional drag at FL 270 was 231 kt. From 14:15, the aeroplane was in level flight at FL 270 with a speed of 279 kt, and this for 45 minutes.

1.5.9 Choice of alternate airfield

The crew envisaged the closest three alternate airfields: Kangerlussuaq (SFJ), Iqaluit (YFB) and Goose Bay (YYR). The en-route alternate airfields (by extension ETOPS airfields, not applicable to the A380) in certain zones are not generally flown to by the company. In this occurrence, Goose Bay (versus Kangerlussuaq) was chosen, after consultation between the CCO and the flight crew, taking into consideration the need to land as quickly as possible given potential damage to the aeroplane which could not be detected in flight. Kangerlussuaq was closer, however the pilots were not familiar with the sole 09 approach. The crew feared that the reduction in performance could be detrimental to the handling of the aircraft and wanted to avoid flying close to the mountains located in the airport's immediate environment (high ground). Other considerations, principally the length of the runway should the flaps block at 0 supported this choice. The fire safety level (RFF) was not taken into consideration. Iqaluit was not selected because of the adverse weather conditions.

1.5.10 Crew statements

At the time of the engine No 4 failure, FO/1 had been in the crew rest station for around 15 minutes; the captain (PM) and FO/2 (PF) along with a cabin crew who had brought a food tray for FO/2 were in the cockpit. The crew mentioned that two passengers, invited to the cockpit, were also present when the failure occurred. They left the cockpit just after the failure. The crew added that their presence did not interfere with the processing of the failure.

Around ten cabin crew including the chief purser (CCP) were in the crew rest area. The economy class purser (CC1) had replaced the CCP and was eating with the business class purser (CC2) in the galley between the first class cabin and the economy class cabin. At the time of the failure, the member of the cabin crew in the cockpit lost her balance and partially spilt the contents of her food-tray behind the flight crew seats.

FO/1 who was in the crew rest station, said that he heard two thuds within two seconds of each other. He then heard another continuous noise, which he put down to a reduction in the engine rating for two to three seconds, and he felt the asymmetry associated with an engine failure. He thought at this moment, that they would not continue to the planned destination. He also perceived the start of a descent. He thought that there had been an engine surge followed by a more serious engine failure. He specified that the noises and sensations perceived were similar to those artificially reproduced in a simulator. He quickly got ready to return to the cockpit. Not wanting to disturb the pilots flying, he said that he waited for a suitable moment to ask for access to the cockpit. He did not envisage ringing or using the emergency code so as not to interrupt the pilots in their tasks. He thought that his entry could destabilize his colleagues. He considered that this "*suitable*" moment was too late to allow him to effectively support the crew with the immediate processing of the engine failure. FO/1 specified that he entered the cockpit at the same time as the CC2 who had come to the cockpit with some information. He considered that he contributed to the FOR-DEC and the diversion, and then later to the arrival.

FO/2 felt a yaw swerve to the right and then a roll to the left; this reminded him of wake turbulence (he had experienced this situation on the B747). He said that when he heard the captain (PM) say "*ECAM ACTIONS*", he probably focused on his instruments. The taking back of the controls by the captain, even though he was not advised of this, did not surprise him. He specified that the vibrations lasted between 10 s and 15 s. He saw the ENG 4 FAIL checklist on the ECAM.

The captain said that at the time of the occurrence, he felt vibrations, he put his hand on the thrust levers and at the same moment saw an "*ENG 4 STALL*" message appear on the ECAM; he specified that this message quickly disappeared to be replaced with an "*ENG 4 FAIL*" message. He had the same perception of the aircraft banking to the left as FO/2 and a swerve to the left. He asked the PF (FO/2) to perform the ECAM actions; as he did not react, he took the controls by changing A/P2 to A/P1 without formally advising him of this. He specified that he immediately thought it was engine surging with substantial damage as he had experienced a similar situation on a B737.

The captain observed that FO/1, providing support after the occurrence, helped the crew to stand back with his suggestions. This way of functioning, common in crews of three, is not formalized in the company's procedures.

The preservation of the CVR was anticipated while in cruise by the captain who asked FO/1 to get ready to take the necessary steps once on the ground. After landing, FO/1 went to the upper avionics bay but was unable to find the CVR circuit breaker due to an error in the onboard aircraft documents. He contacted maintenance by satellite telephone who then gave him the correct part numbers. He specified that it was difficult to access the avionics bay and that pilots are not in the habit of going into it. He added that the CVR circuit breaker could not be differentiated from the other circuit breakers.

All the crew said that training had allowed them to effectively reproduce the procedures. It also made them more receptive in their exchanges.

1.6 Aircraft information

The Airbus A380 is a two-deck, long-haul jumbo jet with four jet engines, certified in 2006. The A380-861 is equipped with four GP7270 engines manufactured by Engine Alliance, an American joint venture between GE Aviation and Pratt & Whitney. The GP7270 was certified in 2005. The Airbus A380 MSN 052, registered F-HPJE and operated by Air France, was built in France in 2011. The aeroplane was delivered new to Air France in May 2011.

1.6.1 Airframe

Manufacturer	**AIRBUS**		
Type	A380 - 861		
Serial Number	52		
Registration	F-HPJE		
Entry into service	2011		
Airworthiness certificate	133411	dated 20 May 2011	
Airworthiness review certificate	2017/133411	from 7 April 2017	to 20 April 2018
Operation as on 30 September 2017	27,184 h		
Owner	DS RENDITE (GER)		
Operator	Air France		

1.6.2 Engines

1.6.2.1 General information

The GP7270 engine (Figure 5) is a variant of the GP7200 engine. P&W[11] is responsible for the design and manufacture of the engine fan.

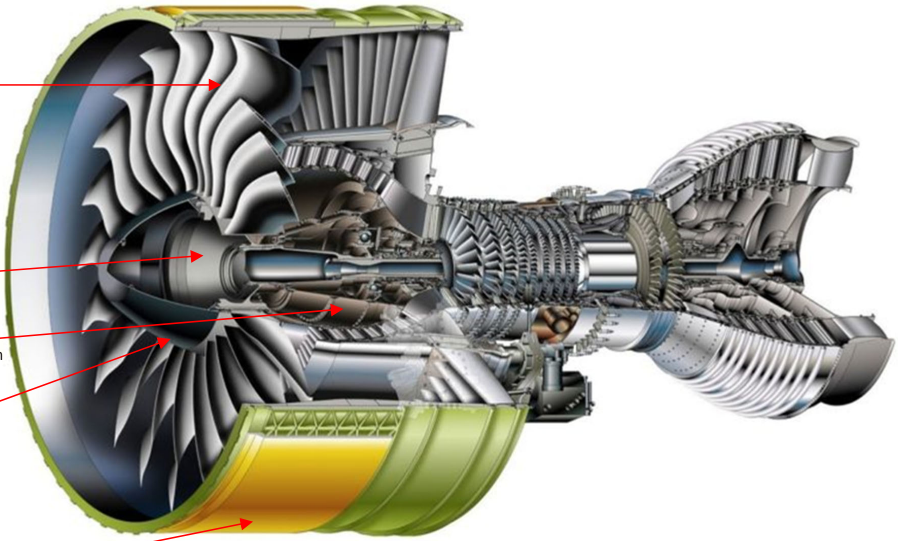


Figure 5: Cross-section of GP7270 engine

	Engine No 1	Engine No 2	Engine No 3	Engine No 4
Manufacturer	Engine Alliance	Engine Alliance	Engine Alliance	Engine Alliance
Type	GP7270	GP7270	GP7270	GP7270
Serial Number	550274	550137	550152	550178
Date of manufacture	28 October 2011	3 August 2008	6 January 2009	30 September 2009
Date of installation	8 December 2015	16 July 2016	14 March 2017	17 April 2013
Total operating time (cycle) at the date of installation	14,062 (1,487)	21,029 (2,516)	18,219 (2,098)	11,731 (1,516)
Total operating time (cycle) since last inspection	21,678 (2,326)	5,084 (567)	2,635 (304)	30,769 (3,534)
Total operating time (cycle) on 30 September 2017	21,678 (2,326)	26,113 (3,083)	20,854 (2,402)	30,769 (3,534)

[11] In the rest of the document, the abbreviation P&W will be used to designate the engine manufacturer who designed and manufactured the parts concerned.

1.6.2.2 Removal and disassembly of engine No 4

After the visual examinations carried out at Goose Bay, engine No 4 was removed while at Goose Bay and dispatched to an approved workshop (GE Wales, Cardiff, Wales) on a specially chartered aeroplane, in order to be examined in the presence of the BEA.

Low pressure (LP) section

The fan hub and the blade found embedded in the OGVs were removed and along with the blade fragments found at Goose Bay (sent to GE Wales with the engine), were sent to the BEA laboratory for further examinations. A preliminary inspection of the fan hub found characteristic features of overload failure. The blade roots exhibited marks on their contact surface with the fan hub. These marks were examined in detail by the BEA laboratory. The low pressure compressor assembly exhibited contact marks between the rotor and stator stages. Unusual circular marks were observed on the low pressure turbine shaft. The outer race flange of bearing No 2 was found fractured into several parts.

High pressure (HP) section

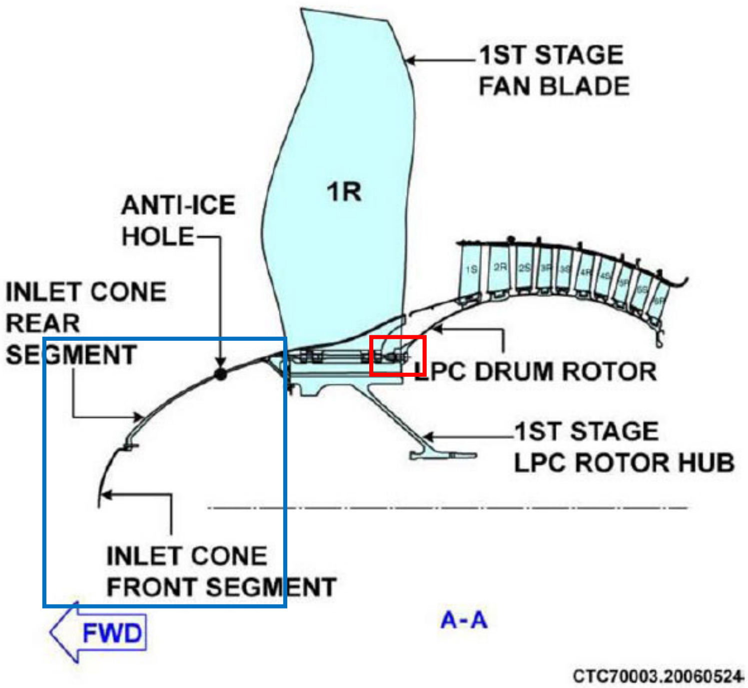
The high pressure compressor assembly, high pressure turbine assembly and combustion chamber were removed. As no major damage was found, no additional examinations were carried out on these elements.

The observations made during the disassembly of the engine were consistent with damage resulting from the failure of the fan hub and its separation from the engine.

1.6.2.3 Detailed examination of engine No 4 elements initially found

Fan blades

The first stage of the low pressure compressor or fan is composed of a fan hub and 24 fan blades, rotating in the clockwise direction, aft looking forward. The cross-section of the LP compressor, including the fan, is shown in Figure 6.



Source: EA

Figure 6: Cross-section of LP compressor of GP7270. The red rectangle identifies the bolts attaching the fan hub to the LP compressor drum. The blue rectangle identifies the inlet cone.

The hollow fan blades are made from the titanium alloy Ti-6Al-4V (or Ti-6-4). This alloy includes 5.5 to 6.75 % in weight of Aluminium (Al) and 3.5 to 4.5 % in weight of Vanadium (V). The blade root contact surfaces with the hub are peened and a copper-aluminium plasma spray coating is applied. A dry film lubricant (DFL) of molybdenum disulphide is then applied to these surfaces. The presence of the plasma spray coating and the DFL reduces wear in service between the blade root and the hub slot.

Twenty-six fan blade fragments were found and examined. The position of nine blades (i.e. eleven of the twenty-six fragments) could be determined by means of the serial number still present and legible (Figure 7). The examined blades mainly showed interaction marks (sliding or impact) with other parts. Certain blades were bent in the rotation direction of the engine, others in the opposite direction. The contact surface of the blades which still had their root, had axial sliding marks, from the rear forwards, except for one (blade No 18 found embedded in the OGVs) which had significantly different damage to its metal coating. This could suggest that the fan hub fractured at slot No 18.

No evidence of progressive failure was observed during these examinations. The fracture surfaces, when they were not damaged, were characteristic of sudden failure due to overload. In addition, no evidence of a bird strike could be observed.

Figure 7: Illustration of fan blades found, when their position is known. Blades 5, 6, 7, 13, 14, 15, 16 and 19 were found on the Greenland ice sheet. Blade 18 was found embedded in the OGVs. The fragments found in the engine at Goose Bay no longer had any identification numbers and could not be positioned on the engine. Position 1 was chosen arbitrarily at the bottom of the engine for illustration purposes.

Engine inlet cone

The engine air inlet cone is composed of a front section and a rear section (Figure 6) made of composite materials. A fragment of the rear section was found on the Greenland ice sheet during phase I of the searches. Two other fragments were found on the runway at Goose Bay. One of these fragments was part of the inner skin of the rear section and the other fragment was all of the front section still attached to a fragment of the rear section. No evidence of a bird strike was observed during the examinations carried out on these fragments. All the failures observed were characteristic of a sudden failure from overload.

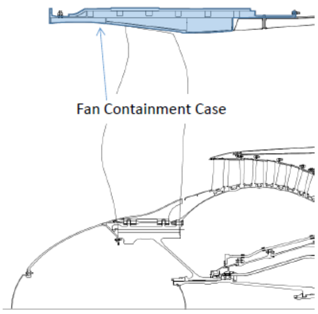
Drum and attaching bolts of LP compressor

The bolts attaching the fan hub to the LP compressor drum (Figure 6) along with the markings on the front face of the drum were examined to help understand the failure scenario. The analysis of the bolt shearing directions and the markings on the drum brought to light two clearly distinct bolt groups. This observation was interpreted as being the result of the fan hub fracturing into two fragments (without counting the conical part still integral with the low-pressure shaft).

Kevlar® fan containment case

The fan containment case of the GP7200 engine (Figure 8) is composed of two Kevlar® belts made up of several layers (an inner belt and an outer belt) plus an environmental wrap. All of the Kevlar® case was found on the Greenland ice sheet during phase I of the searches.

P&W examined the Kevlar® belts in March 2018. The result of this examination was compared with the observations on the engine.



Source: EA

Figure 8: Diagram of fan case and its components

Large perforations were visible between the 10 o'clock and 11 o'clock positions and between the 4 o'clock and 6 o'clock positions (Figure 9). In addition, all the Kevlar® layers had failed at 2 o'clock. A detailed examination of the fibre failures found that these had failed for the most part, under a tensile overload.

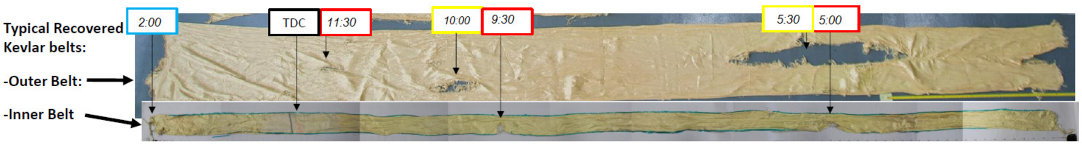


Figure 9: Typical example of recovered Kevlar® belt

The damage observed on the inlet cowl aft bulkhead and on the fan case confirmed the hypothesis of the fan hub breaking into two fragments (without including the conical fragment still integral with the LP shaft), one being ejected towards the 11 o'clock position and the other towards the 5 o'clock position. A second impact occurred on the containment case around the 2 to 3 o'clock position, generating less severe damage than the two previous ones.

Fan hub

The GP7200 fan hub is forged from a titanium alloy Ti-6-4 billet[12]. This alloy is obtained by a triple vacuum arc re-melt (VAR) process.

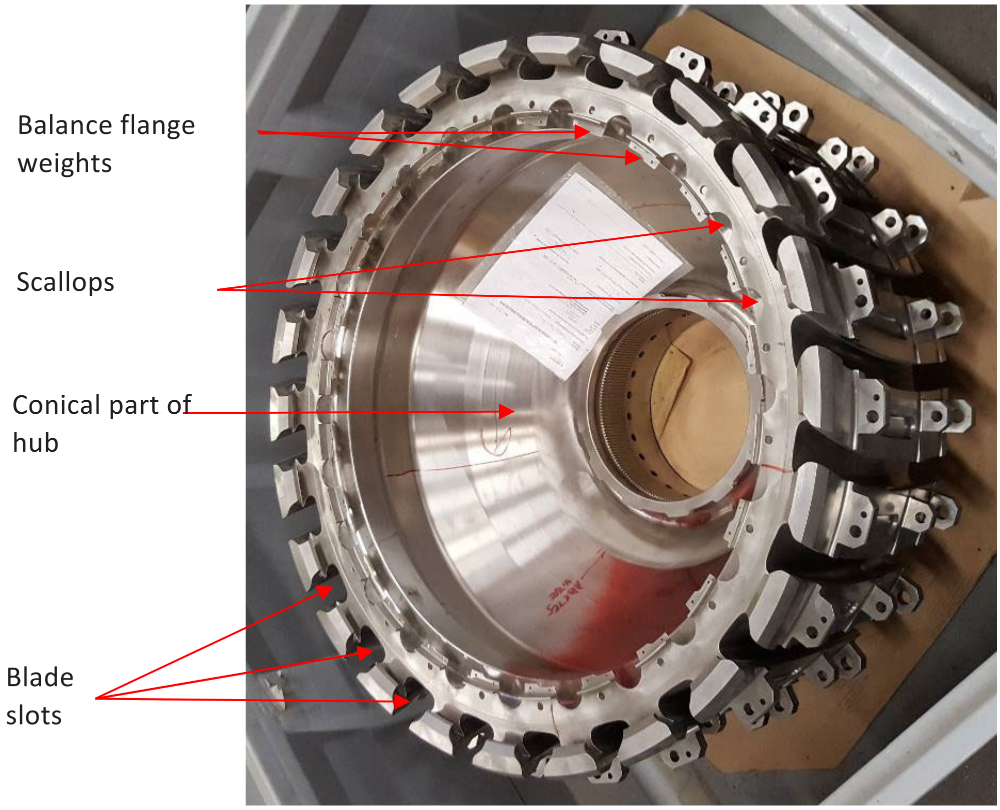


Figure 10: fan hub exemplar. The centre of the conical part is connected to the LP shaft. It is topped by a thicker cylindrical part with 24 machined blade slots.

The central conical part of the hub ([Figure 11]) was still attached to the low-pressure turbine shaft. The fracture surfaces were visually examined and by means of a Scanning Electron Microscope (SEM). They were characteristic of a sudden failure due to overload. The examination of the material's microstructure and its chemical composition did not find any anomaly. The examinations carried out on this fragment were not able to identify the cause of the hub failure.

Figure 11: Hub fragment after its removal from damaged engine

[12]Metallurgy industry semi-product produced from a cast alloy generally in the form of a circular bar.

1.6.3 Maintenance

1.6.3.1 Airframe

The last inspections carried out on F-HPJE before the accident were:

- Last 6 year inspection: from 1 December 2016 to 7 February 2017;
- Last four year inspection: from 26 October 2014 to 14 November 2014;
- Last two year inspection: from 1 December 2016 to 7 February 2017.

The scheduled maintenance operations of F-HPJE were carried out in accordance with Air France's Aircraft Maintenance Program (AMP), developed on the basis of Airbus' Maintenance Planning Document (MPD) and approved by the DGAC, who are also responsible for oversight.

The examination of the maintenance documentation and program did not reveal any anomaly.

1.6.3.2 Engine No 4

The GP7270 engine is a modular engine comprised of 13 modules. It is possible to interchange the modules of different engines according to maintenance requirements.

Engine serial number 550178, was delivered new and complete for installation on the Air France A380 registered F-HPJB (serial number 040) in position No 1. It was removed from this aeroplane on 19 December 2012.

The engine is composed of a propulsor module, designated by the letter P, that includes the fan hub, low pressure assemblies, high pressure assemblies and the combustion chamber, and a fan module, designated by the letter F, that includes the fan blades, inlet cone, fan blade lock ring, and fan containment case assembly. Engines 550178 and 550152 were split. Module P550178 was mated with F550152 on 20 March 2013 and installed on F-HPJE in position No 4 on 17 April 2013. This engine has the serial number 550178.

When the failure occurred, the operating information was the following:

- Time Since New (TSN) = **30,769** hours;
- Cycle Since New (CSN) = **3,534** cycles;
- Time since last fan hub inspection (fan blade relub) = **5,602** hours;
- Cycle since last fan hub inspection (fan blade relub) = **622** cycles.

The fractured hub had accumulated 3,534 cycles.

The scenario of damage resulting from a maintenance operation (removal of fan blades) was examined following the inspection campaign launched after the accident (refer to section 1.16.6). The operation to remove and then reinstall the blade lock ring was considered difficult due to the rigidity of this ring. After the engine event, its design was changed in order to facilitate its installation and its removal and thus reduce the risk of damaging the front face of the fan hub.

1.6.4 Engine computers

A FADEC (Full Authority Digital Engine Control) system controls each engine on the A380. The FADEC system includes an EEC (Electronic Engine Control) and an EVMU (Engine Vibration Monitoring Unit).

The EEC is a dual-channel computer which manages the fuel flow (via the Fuel Metering Unit (FMU), carries out operational tests and monitors failures linked to valves, solenoids, data buses, electrical power supplies, sensors and probes.

The EEC data is principally used by the EIPM (Engine Interface Power Management), the Control and Display System (CDS), the Flight Warning System (FWS) and the flight control primary computer (PRIM). The EEC sends the failures to the Centralized Maintenance System (CMS) and records the faults and associated parameters for maintenance purposes. Each engine has two uni-axial accelerometers which measure vibrations: one situated on bearing No 1 attached to the edge of the bearing support of the low pressure compressor and the second attached to the rear of the high pressure turbine. The EVMU combines these vibrations with the speed signals. It also records dated faults, vibration reports and spectrum analyses for the previous ten flights for maintenance purposes.

Air France only analyses the EVMU data when balancing an engine. In the event of abnormally high vibrations on an engine, the first step consists in lubricating the fan blade roots with molybdenum disulphide. If the lubrication has not significantly reduced vibrations during the ground run, the second step is to carry out a balancing test with the EVMU. The result of the balancing test with the EVMU determines the balance weights to be added and their positions.

Air France confirms that lubrication is carried out at the end of a task scheduled every 4,000 flight hours or every 500 cycles (MRI 723100- E7001-51AFR01), or if there has been a level 1 alert (corresponding to a vibration level measurement of the low pressure compressor and turbine exceeding 3.5 CU[13] in flight). The measurement of vibration in the high pressure compressor and turbine in flight does not initiate a lubrication operation. The vibration measurements by the four EVMUs installed on F-HPJE never led to a balance test being carried out. The lubrication operations carried out kept the vibration level within the nominal range.

[13] CU designates Cockpit Unit which is a quantification of the vibration perceived in the cockpit.

Data recorded in EEC

The analysis of the maintenance data recorded in the EECs associated with engines 1, 2 and 3 did not reveal any fault in connection with the event.

The analysis of the data recorded in the EEC associated with engine 4 found that multiple faults were recorded during the accident flight. These faults were the result of the engine failure and were linked to the independent measurement systems. As the EEC is at the centre of the FADEC and continuously receives numerous engine control and monitoring parameters, the absence of a fault prior to the failure is an indicator of the suddenness of the occurrence. The operational tests of the EEC showed that no operating anomaly had been detected before the failure.

Data recorded in EVMU

After removal from the aeroplane, the EVMU associated with engine No 4 was tested on the equipment manufacturer's bench, BAE Systems, and the recorded data was analysed.

The two vibration reports triggered nominally during the transitions to the take-off phase (09:50) and cruise phase (10:17:07) did not have abnormal vibration parameters. The values were well below the maximum threshold defined by the manufacturer (5 CU).

A third vibration report was triggered at 13:49:30 by the 5 CU threshold being exceeded on the low pressure compressor and turbine. In this report coinciding with the engine failure, the amplitude of the vibrations detected reached the extreme amplitudes of 30 mils pk[14] for the low pressure compressor and turbine vibrations and 18 ips pk[15] for the high pressure compressor and turbine vibrations. As a comparison, 8 mils pk correspond to the threshold of 5 CU for the low pressure compressor and 2 ips pk correspond to the threshold of 5 CU for turbine vibrations.

The vibration peak at 18 ips pk was too sudden to trigger the 5 CU exceedance.

The flight history contains the maximum vibration values according to the low and high pressure compressor and turbine speeds in the stable rating phases. No vibration of a large amplitude was measured during the phase in which the stability criteria was met. The maintenance data recorded by the computer showed that the engine No 4 vibrations were below the threshold of 5 CU before the separation of the fan at 13:49:18, this being the threshold which generates a warning in the cockpit.

No operating anomaly which could have affected the monitoring of the vibration parameters was revealed by the examinations.

The examinations of the computers confirmed that before the engine No 4 failure, no early sign of abnormal operation of engine No 4 was detected. The monitoring of the vibration level did not reveal any anomaly. The engine control was not the cause of the failure. The engine failure was sudden.

1.7 Meteorological Information

1.7.1 General situation

The east half of the south part of Greenland was covered with five to eight okta Stratocumulus, Altocumulus and Altostratus between FL 070 and FL 180. Light to moderate icing was possible between FL 110 and FL 150. The west half of the south part of Greenland was covered with five to seven okta Stratocumulus, locally three to four okta, between the ground and FL 070. A cold front was approaching the east coast of Greenland, heading in a west-south-westerly direction at around ten knots. The surface projection of an occluded front was behind the cold front, heading in a west-south-westerly direction at around five knots. Iso 0°C was measured at 2,500 ft on Narsarsuaq airport (Greenland). From leaving Paris and up to the south of Iceland, the aeroplane had flown near a zone of clear air turbulence created by an (east to west) jet stream at FL 340, situated west of Ireland. The aeroplane encountered moderate zones of turbulence. For the rest of the flight and up to the spot where the failure occurred, F-HPJE had been subject to a 230° crosswind at 50 kt, without significant windshear and with temperatures of around -52°C. Satellite images showed few to scattered clouds, composed of cirrus, above FL 250.

[14] Mils pk (milli-inch peak to peak) is a vibration amplitude measurement unit, corresponding to 0.001 inch peak to peak displacement.

[15] Ips pk (inch per second speed peak to peak) is a vibration velocity measurement unit, corresponding to 1 inch per second peak to peak displacement.

The weather conditions were not linked to the failure and did not affect the diversion.

1.7.2 Aerodrome weather reports and forecasts at Goose Bay

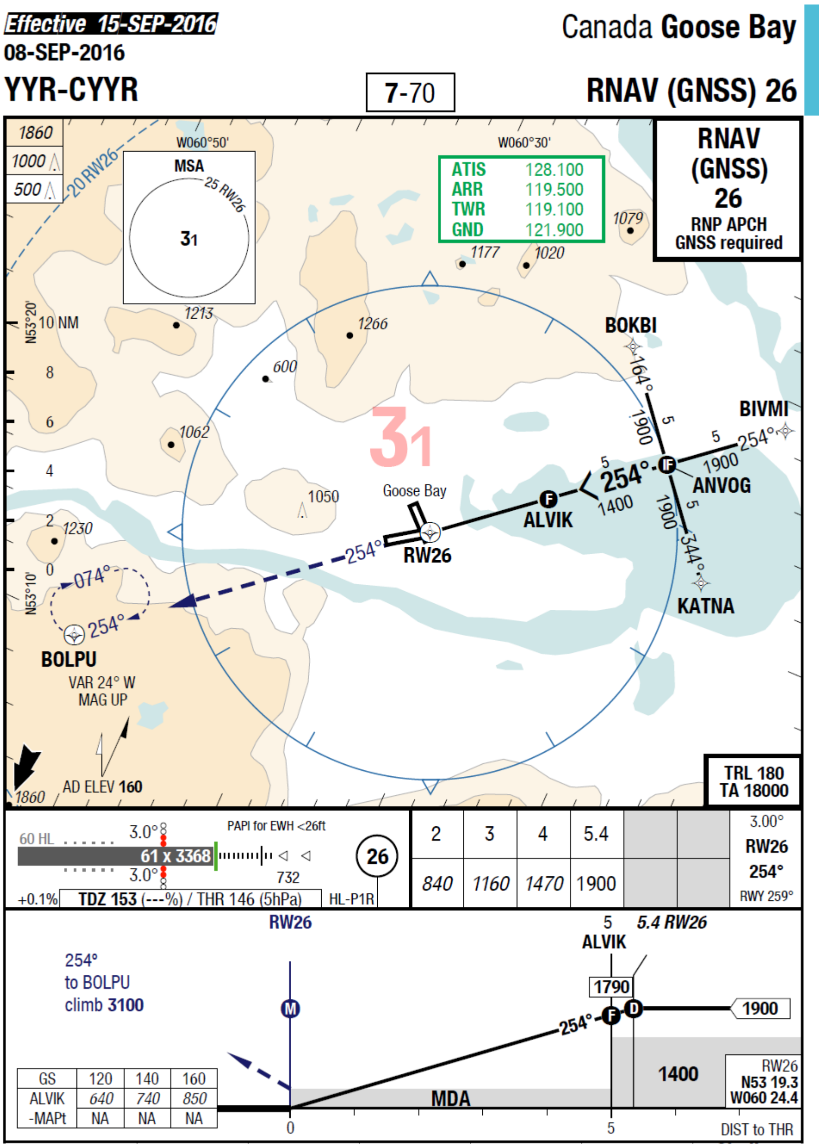
The wind was from 250° at 15 kt, visibility was greater than 10 km, the temperature was 8°C, the dew point 0°C, few clouds were present at 3,000 ft. The sky was covered at 6,000 ft.

1.8 Aids to navigation

Only the navigations aids for the alternate aerodrome (Goose Bay) are described. Given the particularities of Kangerlussuaq (Greenland) discussed in section 1.10.2, the crew chose not to land there.

The Goose Bay airport runways have the following approaches:

ILS Z 08, PAR 08 et 26, LOC Back CRS Z 26, NDB 26, RNAV (GNSS) 08, 16, 26 and 34.



The RNAV LNAV (GNSS) 26 approach was carried out by the crew of F-HPJE.
It has the particularity of having a 5° offset between the approach path and the runway axis.

1.9 Communications

The Datalink is an air-ground digital communication method allowing messages to be exchanged between the crew and air traffic control services or the operator via the HF and V-UHF bands or satellite links. The advantage of this system is the possibility it provides of dispensing with voice communications to transmit information to the crew or to ensure communication between the aeroplane and the ATC when the latter cannot communicate via HF or VHF.

The Datalink messages include the following messages:

- ATIS: Automatic Terminal Information Service (information about the airports such as runways in use and the weather);
- CPDLC : Controller-Pilot Data Link Communication (written messages between crew and controller, notably clearances and requests);
- ADS-C: Automatic Dependant Surveillance - Contract (automatic data reports from the onboard navigation and position calculation equipment sent by the aeroplane to the ground system).

At the time of the engine No 4 failure, the aeroplane was outside radar coverage. The change in the vertical and lateral profiles of the path, after the separation of the engine No 4 fan, was therefore not recorded and it was not possible to identify the potential conflicting paths during this period. No conflicting traffic was reported by TCAS.

The event is time stamped 13:49:18 when the aeroplane was in datalink contact with Gander Oceanic. Several "*ALTITUDE RANGE CHANGE*" ADS-C messages were sent between 13:51:57 and 13:54:59 which confirms that the aeroplane was descending and a "*LATERAL DEVIATION CHANGE*" ADS-C message was sent at 13:55:15.

The Gander Oceanic control centre received a Mayday audio message from AF066 at 13:56, relayed by another aeroplane.

From 13:56:53, the air traffic control services and crew communicated via CPDLC messages. The control sent the message: "*ATC NOW SHOWS YOU FL330. IS THERE A PROBLEM*".

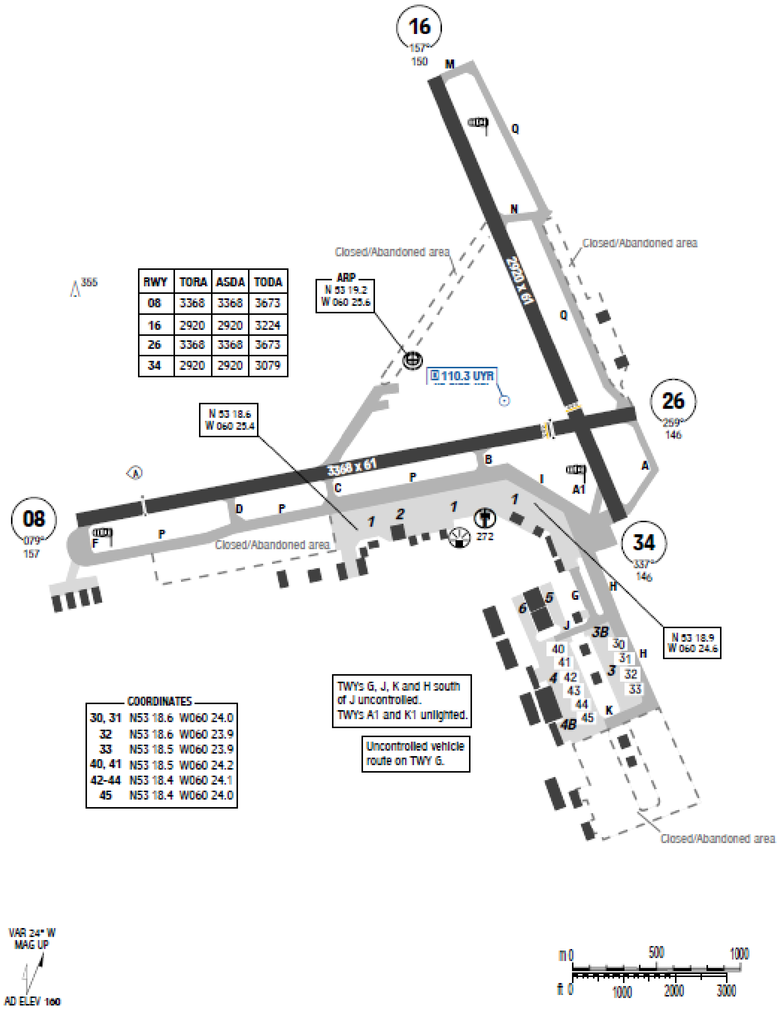
The crew subsequently replied with a Mayday message. The ATC informed the crew of the state of the surrounding traffic and accompanied them in their desire to divert.

The change in the lateral and vertical profiles of the path following the failure had not conflicted with the paths of other aeroplanes. The AF066's state of emergency was relayed and taken into account by the air traffic control services.

1.10 Aerodrome information

1.10.1 Goose Bay airport

Mixed-use Goose Bay airport (IATA: YYR, ICAO: CYYR), is situated in the Happy Valley-Goose Bay municipality (Terre-Neuve-et-Labrador, Canada). It is used as a military airbase by the Royal Canadian Air Force (RCAF) and by civil aircraft. The aerodrome is considered as the joining point by Nav Canada. It is often used as an alternate airfield by aeroplanes carrying out transatlantic flights. The Canada Border Services Agency indicates that the aerodrome can handle international flights performed by general aviation aircraft transporting no more than 15 passengers. The passenger disembarking and handling capabilities are limited. The handling capabilities for aircraft (towing, jetway, GPU) are also not adapted to an A380.



This aerodrome is equipped with two intersecting concrete runways with an asphalt finish: Runway 08/26: 3,368 x 61 m and runway 16/34: 2,920 x 61 m.

Air traffic control is provided 24/7 for VFR and IFR flights.

The Aircraft Rescue and Fire Fighting service is level 7.

1.10.2 Kangerlussuaq airport

Kangerlussuaq airport (IATA: SFJ, OACI: BGSF) is situated in the municipality of Qeqqata in the centre-west of Greenland (Denmark). Along with Narsarsuaq airport, it is one of only two civil aerodromes in Greenland able to handle large airliners and can be designated as an ETOPS alternate airfield.

The aerodrome is equipped with an asphalt runway 09/27, of 2,810 x 60 m. It does not have a turn-around area at the end of the runway. There are no towing means available for very heavy aeroplanes.

Air traffic control is provided Monday to Saturday, between 11:00 and 20:00, and outside of these hours on receiving a 9 h notice except in the case of an emergency (by contacting the Sondrestorm flight information centre).

The Aircraft Rescue and Fire Fighting service is level 5, or 8 on receiving a 4 h notice.

The operator's documentation indicates that only the LOC approach on runway 9 is authorized (possibility of landing with a tailwind component).

It also specifies that the aerodrome is situated in a polar region where storms and icing conditions are frequent. In addition, the magnetic declination is large. There are no centreline markings and the approach slope (3.45°) is steeper than that indicated by the PAPI (3.2°). For these reasons, landing at this aerodrome is only recommended when another adequate aerodrome is not accessible.

Although situated at around two flight hours from the geographical position where the failure occurred, the crew chose Goose Bay airport which was better adapted than Kangerlussuaq for handling a damaged A380 and all the persons onboard.

1.11 Flight Recorders

1.11.1 Regulatory recorders

In accordance with the regulations in force, the aeroplane was equipped with two flight recorders (FDR and CVR).

Flight Data Recorder (FDR)

- manufacturer: L3 Communication
- model: FA2100
- part number: 2100-4045-00
- serial number: 637136

It is a Solid State Flight Data Recorder (SSFDR) with a recording capacity of at least 25 hours. The document to convert binary data into physical values provided by the manufacturer, provides information about approximately 3,000 parameters.

Cockpit Voice Recorder (CVR)

- manufacturer: L3 Communication
- model: FA2100
- part number: 2100-1025-02
- serial number: 571575

It is a Solid State Cockpit Voice Recorder (SSCVR) with a recording capacity of at least two hours. This recorder can also record Datalink messages.

The CVR and FDR were synchronized using the disconnection of the autopilot, the landing gear compression parameter on wheel touchdown and the Master Cautions during the landing. This synchronization was then fine-tuned using the parameter recording the pressing of the radio transmission button.

1.11.2 Read-out of regulatory recorders

The flight data recorder contained around 84 hours of data corresponding to seven flights, including the accident flight.

The aeroplane took off from Paris Charles-de-Gaulle at 09:50:47. The FDR recording contained data up to 15 h after the engine shutdown on the Goose Bay tarmac in accordance with the FDR end of recording logic, as the aeroplane remained powered. The CVR audio recording started at 14:23:05, i.e. 36 minutes after the engine No 4 failure.

The following tracks were recorded:

- track 1: captain's radio communications and microphone signal;
- track 2: first officer's radio communications and microphone signal;
- track 3: third pilot's radio communications and microphone signal (rear seat), the FSK signal (time signal) and the Public Address;
- CAM track: cockpit area microphone signal.

The recording time of the captain's, first officer's and Public Address' tracks was 2 h 03 min 53 and the recording time of the CAM track was 02 h 04 min 00.

After the occurrence, the following elements were noted on the CVR:

- At 14:27:11, the crew checked the ECAM status and saw that all engine No 4 information was no longer available.
- At 14:28:11, FO/1 reported his visual observations of engine No 4 and the wing.
- At 14:48:15, the crew changed frequency to keep contact with Gander (135.4 MHz).
- At 15:03:42, the crew contacted Goose Bay Approach (119.5 MHz).
- At 15:42:16, the aeroplane landed.
- At 16:22:05, the aeroplane was on the tarmac and the engines shut down.

The CVR recording automatically stopped five minutes after the last engine was shut down in accordance with the CVR automatic end of recording logic. Around 45 minutes therefore elapsed between wheel touchdown and the CVR stopping. The time which elapsed between the occurrence and when the CVR stopped was 2 h 37 min.

1.11.3 Preservation of CVR

Paragraphs CAT.GEN.MPA.105 and CAT.GEN.MPA.195 of the European regulation, *Air Operations* No 965/2012 (of October 2019) and the associated AMCs designate the captain as being responsible for the preservation of the CVR and the operator as being responsible for drawing up the procedures to do this.

After arrival on the apron, FO/1 went to the upper avionics bay but was unable to find the breaker due to a positioning error in the onboard aircraft documents. He then contacted maintenance by satellite telephone who gave him the correct part numbers. The CVR recording had already stopped following the shutdown of the last engine, before the FO/1 went to the avionics bay. A review of the onboard documentation found that the information available to the crew did not allow them to effectively preserve the CVR before the engine shutdown as the procedure available did not reflect the aeroplane's configuration. The indicated position of the breaker was not correct. As a consequence, the preservation of the CVR chiefly relied on the engine being shut down.

At EASA, the work of the RuleMaking Task RMT.0249, Recorders installation and maintenance thereof - certification aspects, led to an update of the European regulation 2018/1139 in 2019. In particular, paragraph CS-25.1457 specifies that:

- (d)Each cockpit voice recorder must be installed so that –
  (5) There is a means for the flight crew to stop the cockpit voice recorder function upon completion of the flight in a way such that re-enabling the cockpit voice recorder function is only possible by dedicated manual action;
- The CVR may be preserved by the crew using the breaker in accordance with AMC 25.1457.

1.11.4 Other recordings

The architecture of the information system networks onboard the A380 has three distinct domains: avionics, flight operations and communication & cabin.

The architecture of the avionic domain includes two identical and redundant ANSU-OPS (OPerationS Aircraft Network Server Units) which support the OIS applications, documentation, database and calculation operations. The ANSU-OPS record:

- A copy of the FDR data (VQAR).
- A copy of the DAR data (VDAR).
- The SAR (Smart Access Recorder) files which record the parameters from the various computers for analysis, notably for maintenance purposes.
- The REP (aircraft system REPort) files which supply a report concerning the operation of the aircraft systems according to pre-determined activation conditions. The REP files only contain a few parameters over a few seconds unlike the SAR files which contain a continuous parameter recording.
- The CMS (aircraft Condition Monitoring System) messages contain the main messages displayed in the cockpit during the flight (reported effects) and the main maintenance messages.

The first two items recorded in the REP files were engine surging (REP #6) and an attempt to start up the engine again (REP #10) at 13:49:18. The analysis of the SAR files supplemented the analysis of the FDR data by providing additional parameters to those recorded in the FDR. The SAR data was synchronized with the FDR data.

1.11.5 Synthesis of recordings

The large amount of information available from the read-out of the regulatory (CVR and FDR) and non-regulatory recordings made it possible to confirm the sudden nature of the engine No 4 failure without explaining its cause.

The FDR data along with the EEC and EVMU computers confirmed that there was no precursory element to the fan separation. It was not possible to precisely describe this separation due to the small sample of available parameters.

1.12 Wreckage and Impact Information

Not applicable.

1.13 Medical and Pathological Information

Not applicable.

1.14 Fire

Not applicable.

1.15 Survival Aspects

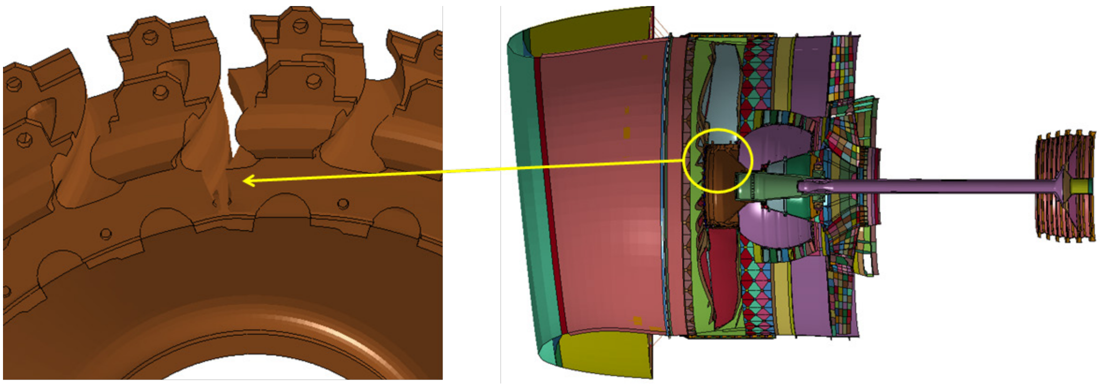
The debris striking the structure did not protrude into the cabin. No perforation in the pressurized zone was observed. No emergency evacuation was needed nor carried out.

No passenger was injured.

1.16 Tests and Research

1.16.1 Simulation of rotor failure

In the absence of a hub fragment showing evidence of the origin of the failure, other possible failure scenarios had to be explored. To do this, P&W created a finite element model of the GP7270 fan module in order to carry out dynamic calculations using the LS-Dyna calculation code. This model was first used to assess the different failure scenarios proposed and to provide insight into the observations on the engine. In a second phase, this model was used to determine the probable mass, size and ejection velocities of the hub fragments which would guide the searches for the parts in Greenland. The finite element model included the low pressure rotor system (fan case, intermediate case, low pressure compressor and static casing structure), the low pressure turbine, and the low pressure rotor bearing supports. The physical finite element models that make up the LS-Dyna model utilized calibrated material models and modelling techniques and procedures that had been validated for the P&W engine development programs. These validations were carried out by correlating model results with test results, specifically during Fan Blade Off and Bird Ingestion tests carried out as part of the engine's certification. The material models used in this analysis were elastic-plastic. The model was not a full engine model, but consisted of a sub-set of engine hardware with the goal of evaluating the behaviour of the fan hub and its local surroundings in the fan module. Because of this limitation, the model results were considered valid for only the initial 20-30 milliseconds after the start of the failure sequence. Nevertheless, this time period was able to simulate the events from the initial part fracture until roughly the time at which the hub fragments (for the cases where the hub ruptured) penetrated the fan containment case. As LS-Dyna is not a fracture mechanics analysis, it was necessary to artificially start the event by forcing the initial fracture of the part. Several scenarios were analysed, the one which was the most consistent with observed damage on engine No 4 was the fan hub "*bore to rim*" fracture (Figure 12).

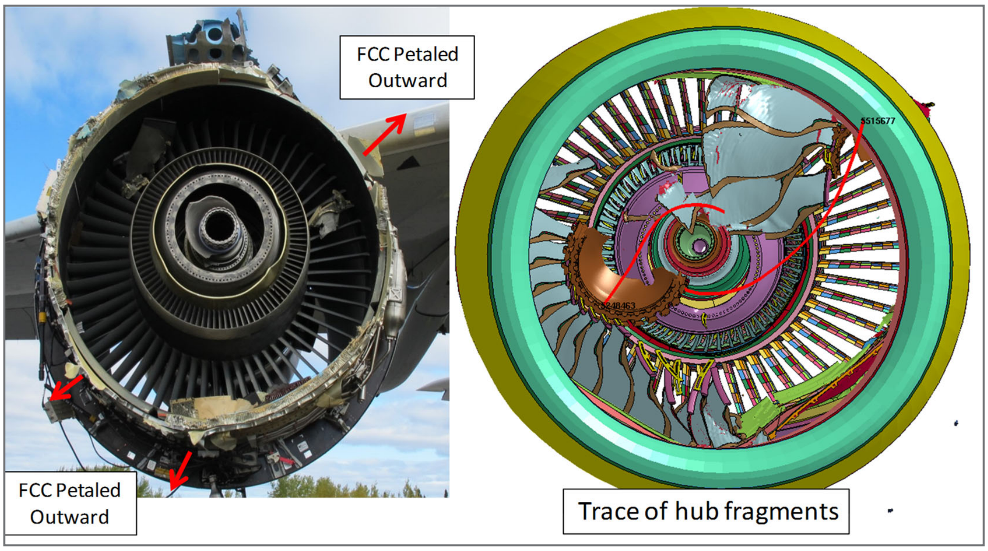


Source: EA

Figure 12: Hub "bore to rim" fracture scenario

The fan hub "*bore to rim*" scenario encompasses any scenario that would result in a hub fracture, including scenarios such as cracks in a blade slot bottom, a buried material defect, and cracks resulting from front face scallop damage (the underlying assumption being that the reason for a crack would not influence the results of the ultimate hub fracture). The other scenarios analysed such as a failure in the middle of a blade or in the blade root, or the failure of a blade retaining lug (disk post) did not lead to the fracture of the rest of the hub (the fan hub remained integral with the engine in the simulations). Therefore, these three scenarios were considered unlikely for explaining the failure of the engine during the occurrence.

The collateral damage and part fragmentation associated with the fan hub "*bore to rim*" fracture in the simulation were consistent with the hardware findings observed on engine No 4. In particular, the fan containment case damage, Kevlar® belt damage, low pressure compressor drum front face damage patterns, and the condition of the fan blades recovered both from the ground in Greenland and in the engine (Figure 13) supported this scenario.



Source: EA

Figure 13: Simulation of fan hub "bore to rim" failure

1.16.2 3D laser scan

On the disassembly of the engine in Cardiff and after the recovery of certain debris in Greenland, it was decided to scan certain elements using 3D laser technology.

The purpose of this was to dispose of images to document the damage, to virtually assemble the recovered parts in order to determine those which were missing and to help with the analysis of the occurrence.

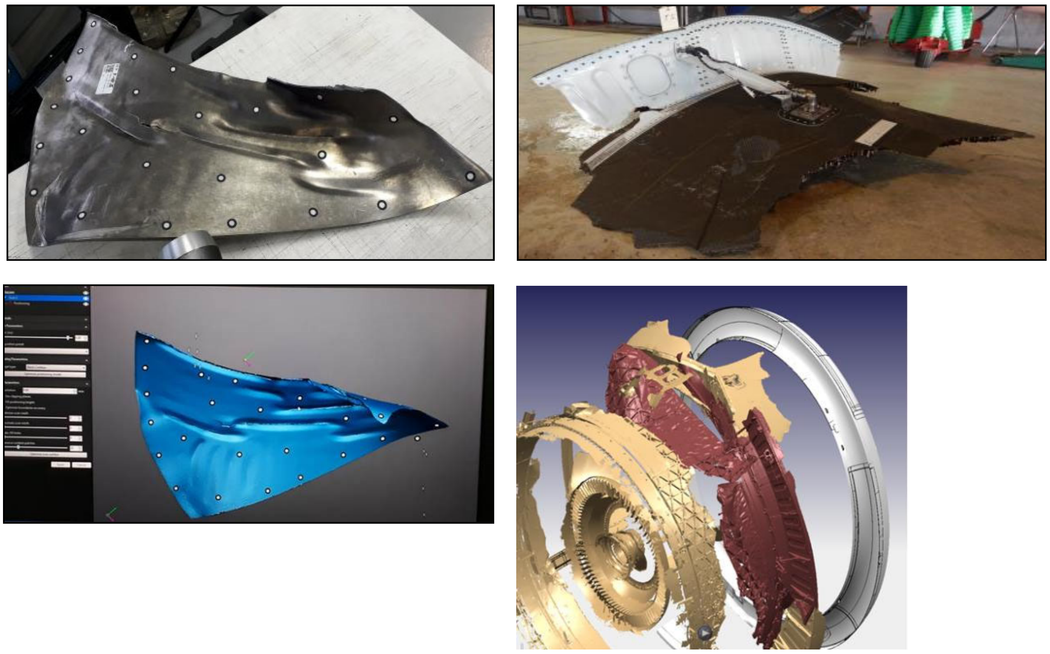


Figure 14: fan blade during scan (top left) and result of scan (bottom left). Part of inlet cowl aft bulkhead (top right) and result of scan (in red, bottom right) inserted in a more complete 3D model of the damaged engine.

One of the main results was the obtaining of the extent of the deformation of the scanned parts. This information was compared with the results of the software simulations in order to reinforce certain failure scenarios in the absence of the fan hub.

1.16.3 Fault tree

Before the recovery of the fan hub fragment, Engine Alliance produced a causal fault tree based on the observations made on engine No 4, and on its historical experience with other engines in service. The objective of this causal fault tree was to identify all the possible scenarios for the fan hub failure on engine No 4 and their root cause in order to discard the causes that could be discarded and to carry out the necessary safety actions for the remaining scenarios.

This fault tree had five main categories:

- Analysis
- Damage
- Defect
- Multiple blade separation
- Engine operation

According to the manufacturer, before recovering the fragment in Greenland, the most likely scenario was damage in maintenance, with the failure arising from tool damage during fan blade lock ring removal. Feedback from the post-accident in-service inspections was in keeping with this scenario.

The examinations carried out on the "*sister*[16]" hubs did not reveal any material discrepancy. Engine Alliance considered that the root cause of a material defect was thus very unlikely.

[16] Hubs manufactured from the same titanium billet (T2, B1, B2 and B3). The F-HPJE hub came from the T1 (see paragraph 1.18.2).

If the part of the hub in which the failure originated had not been found, the BEA could not have checked conclusively this hypothesis.

In this fault tree, the cold dwell fatigue phenomenon, identified after the examination of the hub part found in Greenland as being the cause of the failure (see paragraph 1.16.5), had been classed as unlikely based on the manufacturer's feedback and on all the data in its possession, notably:

- The cumulative forging strains during the "*Near Net Shape*" process (NNS, see paragraph 1.18.1), were consistent with good forging practices under the P&W Engineering Source Approval process and considered sufficient for limiting the risk of having large macro-zones with increased intensity.
- The electron backscatter diffraction (EBSD) analyses on GP7000 forging cut-up material and on a sister hub from the event heat of material did not indicate a high level of macro-zones.
- The absence of failures in other fan hubs installed on other engines used in commercial air transport, forged in Ti-6-4 using billets of similar diameters.

1.16.4 Searches in Greenland

NB: All of the search phases are the subject of two dedicated technical reports.[17]

[17] TECHNICAL REPORT, Accident to the Airbus A380 registered F-HPJE and operated by Air France on 30 September 2017 en route over Greenland, October 2017 - June 2018, Search phases I and II. TECHNICAL REPORT, Accident to the Airbus A380 registered F-HPJE and operated by Air France on 30 September 2017 en route over Greenland, July 2018 - July 2019, Search phase III.

Phase I

Search phase I consisted in initially determining, straight after the occurrence, a "*rough zone*" where debris was likely to be found and to recover, as far as possible, all the parts visually identified.

During this phase, three helicopter flights were carried out between 4 and 11 October 2017 in a zone defined by the BEA, based on data from the FDR. Around 30 pieces of debris were recovered: fan blade fragments, fan containment case, front cone fragments, the complete lip of the air inlet (in three parts) and parts of the nacelle. No fan hub fragment was found at this time. No additional debris was found during the third flight. Snowfalls and the wind had covered the parts still present in the zone with snow, preventing further visual detections. The decision was taken to end search phase I on 12 October 2017.

It then became necessary to explore other methods to locate and recover the missing parts, the priority being the fan hub fragments.

Phase II

Search phase II consisted in assessing detection means to locate the hub fragments on the Greenland ice sheet as well as preparing and carrying out the search operations which took place in April and May 2018. The detection means had to be compatible with the specific environmental conditions in the zone where the debris had fallen and with all the associated operational constraints. It is not possible to go to this region during the winter as the safety level is considered insufficient (very low temperatures, short days, changeable weather, presence of crevasses, etc.). The spring of 2018 was the closest period which could be considered for search and recovery operations. After an assessment phase of search means, it was decided to set up two consecutive operations:

- an aerial campaign, consisting in the use of synthetic aperture radars operated from an aeroplane, to try to detect and locate the missing parts under the layer of snow (SETHI Radar operated by ONERA);
- a ground campaign, consisting in recovering the parts previously located during the aerial campaign, or in performing a systematic search with the help of ground penetrating radars (GPR) if the aerial phase was unsuccessful (GEUS).

Despite the efforts made in the operations described above, the fan hub fragments were not detected at the end of June 2018.

The SETHI technology is experimental and its deployment over the ice sheet to detect parts buried under the snow was new. Due to both the higher than expected background scatter noise and the less than expected radar penetration, no target with a sufficient confidence level was detected in the relatively short time before the ground campaign carried out by GEUS.

ONERA finally indicated six moderate-confidence targets to GEUS for its ground campaign. The data processing was refined in order to improve the quality of the radar detections. The GEUS ground campaign was first based on the targets detected in ONERA's aerial campaign; once the six targets that had been provided had been explored without any debris being found, it became a systematic search campaign. The GPR towed on the ice behind a snowmobile proved to be a sub-optimal sensor for a wide-area search. In all, 430 km of GPR measurements were analysed without being able to certify that if the part had been located under these swaths, it would have been identified. Despite the search zones being given priorities following the more accurate ballistic calculations carried out by Airbus and the NTSB, no debris was found before this second search campaign came to an end.

The confidence of the ONERA experts that it was possible to improve the processing of the SETHI Radar data to identify targets with a higher confidence led the BEA to envisage continuing the work. The ONERA team continued processing the radar images acquired during the aerial campaign after the team's return to France. In November 2018, the process was still ongoing. New specific algorithms detected the test hub[18] in the X band which was a promising result. In conjunction with this work, the investigation team thought that it was necessary to test new ground sensors with a wider swath and a more reliable return before initiating a new search phase in 2019.

[18] The test hub was a portion of a hub to a scale of 9:10 supplied by Engine Alliance. It had been deliberately positioned on the site prior to the searches for calibration and detection test purposes for the aerial campaign.

Phase III

The finite element simulations carried out by P&W at the beginning of 2018, based on the hypothesis of a fan hub "*bore to rim*" fracture, provided information about the probable size, weight and ejection speeds of the hub fragments. New ballistic calculations were carried out based on this refined data in order to reduce the size of the search zone. At the end of 2018, the electromagnetic detection system developed by the HydroGeophysics Group (HGG) of Aarhus university (Denmark) was modified so as to be able to detect a titanium part at a distance of five to six metres under the snow. In parallel, the post-processing of the SETHI Radar data acquired during phase II had been completed. ONERA sent the coordinates of a high-probability target and of two less obvious targets to the investigation team. The decision to carry out a new expedition was taken at the end of February 2019 for a departure in May 2019. The expedition kick-off was delayed due to weather conditions which were incompatible with the mission. Despite this hold-up and the resulting reduction in the mission's duration, an unambiguous detection was obtained at the very end of the campaign at the most promising spot indicated by ONERA. A signal about two orders of magnitude higher than that obtained for the buried test hub was recorded close to the spot where the GPR had already made a detection, indicative of buried metal. The detection was situated one metre north of a four-metre wide crevasse which had a six-metre thick bridge. An excavation campaign was organized in June 2019. A fan hub fragment was carefully extracted and transported to Narsarsuaq airport. The fragment returned a signal to the system used by HGG which was a lot stronger than that of the buried test hub, in part due to its larger size and shallower burial depth. The blades still attached and the presence of other materials (aluminium and steel) may also have increased the signal strength. The part was handed over to the BEA in July 2019 and examined at the engine manufacturer's, P&W, in order to determine the cause of the failure.

Source: Austin Lines

Figure 15: Extraction of fan hub fragment

1.16.5 Examination of hub fragment found during phase III

The hub fragment found in Greenland was sent to P&W in July 2019 in order to carry out an examination, supervised by the BEA. The fragment found is shown in Figure 16. Fan blade fragments were still attached to the hub. Two fracture surfaces were visible, in slots No 10 and No 18. The failure of its conical part was confirmed as matching the conical fragment found still attached to the engine.

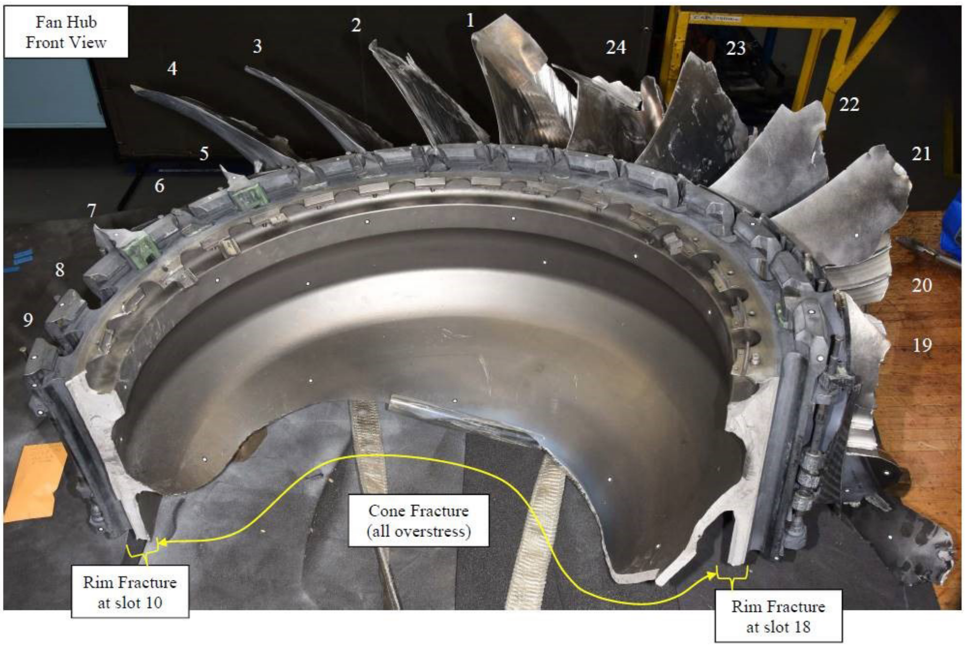


Figure 16: fan hub fragment found in Greenland during phase III. The slot numbers are given in white. The fracture surface extends from the bottom of slot No 10 to slot No 18, passing through the conical part of the hub (yellow line).

The examination determined that the hub failure was caused by a LCF[19] cracking process which originated in the part's subsurface. The origin of the crack was located practically in the centre of slot No 10 (Figure 17), around 14 cm (5.6 inches) behind the front face of the hub and 1.4 mm (0.055 inches) below the surface of the slot bottom. No material quality (composition, microstructure) or manufacturing related anomaly was found.

[19] Low Cycle Fatigue.

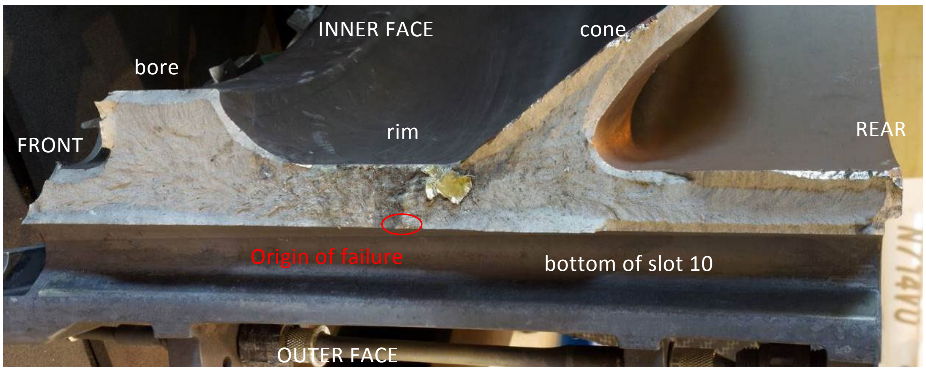


Figure 17: Fracture surface of slot No 10

A region of fatigue striations, characteristic of LCF progression was observed between the origin of the failure and the hub's inner face. The striations were evenly spaced at distances of between 25.4 x 10^{-5} mm/cycle and 25.1 x 10^{-4} mm/cycle (~1 x 10^{-5} inch/cycle and ~9.9 x 10^{-5} inch/cycle respectively). The grains situated between the origin of the failure and the surface of the slot bottom chiefly had near-cleavage[(20)] facets (faceted growth) with the occasional presence of striations. P&W counted the number of striations in order to determine the number of progression cycles required for the crack to become a fracture. The hypothesis that one striation equals one aeroplane cycle is generally accepted in LCF for rotor parts essentially subject to centrifugal loads. To do this count, different striation images were taken with the Scanning Electron Microscope (SEM), at different distances from the origin, from 0.076 mm to 1.7 mm (0.003 to 0.068 inches). On each image, the ratio of number of striations per millimetre was calculated by dividing the number of striations observed by the distance separating the first striation observed from the last. However, some of these images were taken in mixed zones where there were also faceted growth. A method was used to determine the ratio between the two fracture modes (cleavage and striations) and to adjust (reduce) the number of progression cycles according to this fracture mode ratio. The total number of striations was assessed at 1,652. During this operation, it was observed that there was a transition zone from which the definition of the striations got better. This zone was attributed to the transition between crack progression in a vacuum and progression in the air, i.e. it indicated the moment when the crack reached the surface of the slot and became surface connected. This zone was situated at around 0.41 mm (0.016 inches) from the origin and corresponded to 773 cycles. This means that a little less than half of the stable progression of the crack, in number of cycles, occurred in a vacuum, subsurface.

(20) The cleavage is the predisposition of a material to fracture along planes in determined directions when it is subject to a mechanical load (impact or continuous pressure). The existence and orientation of cleavage planes depend on the symmetry and crystal structure (weakest bond planes in the structure).

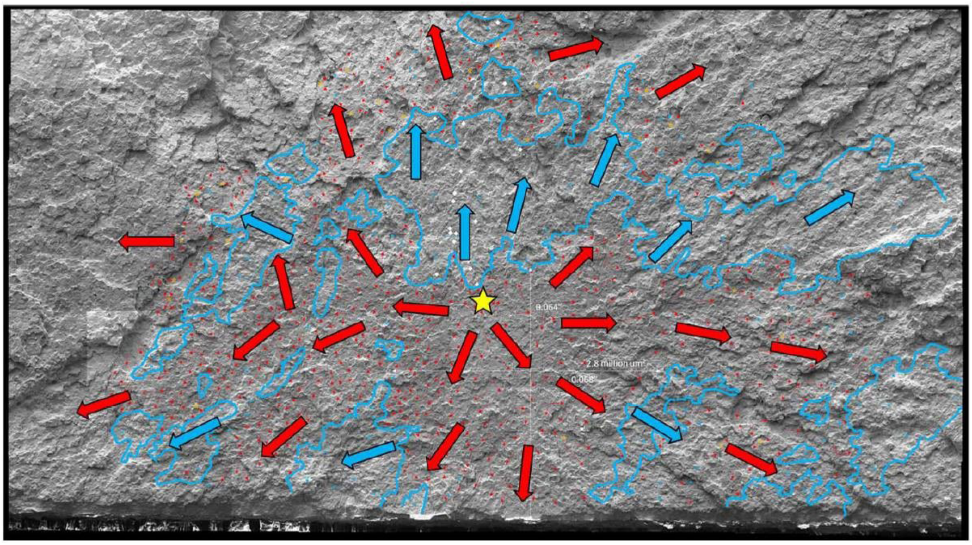


Figure 18: SEM detail view of origin of fracture. The red arrows show the crack growth direction in the faceted regions and the blue arrows the growth direction in the striated regions. Areas where the striations are predominant are outlined in blue. The orientations obtained confirm the existence of a primary origin at 1.4 mm below the surface of the slot bottom (yellow star) in a predominantly faceted region.

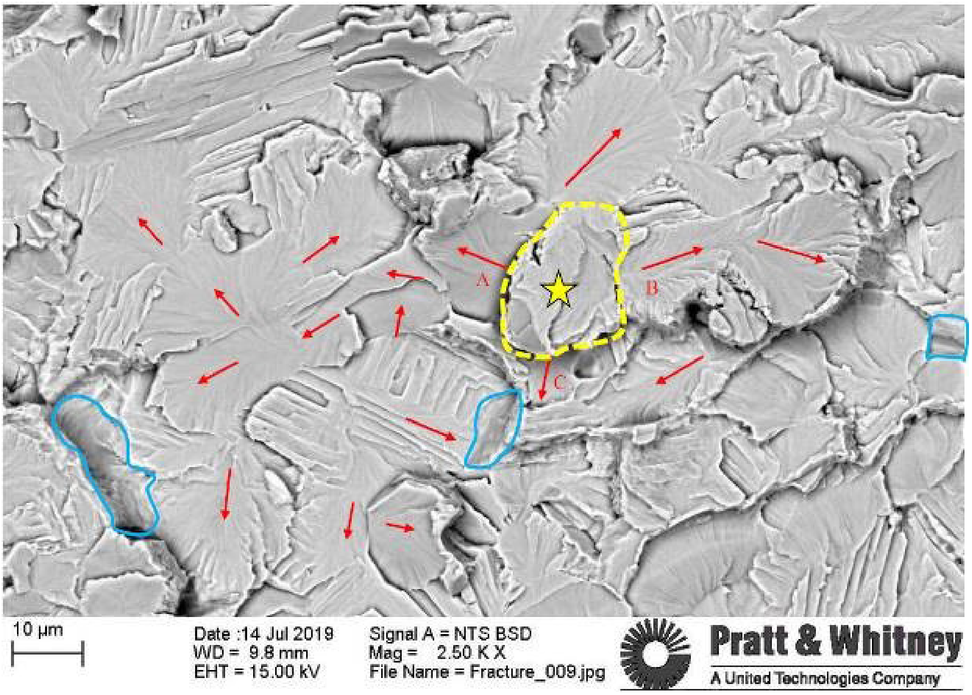


Figure 19: SEM view of origin of fracture (faceted growth region). The red arrows indicate the growth directions. The grains A, B and C exhibit a crack progression direction away from a small area covered by a flap (yellow dotted line). The grain under the flap was revealed using a Focus Ion Beam (FIB).

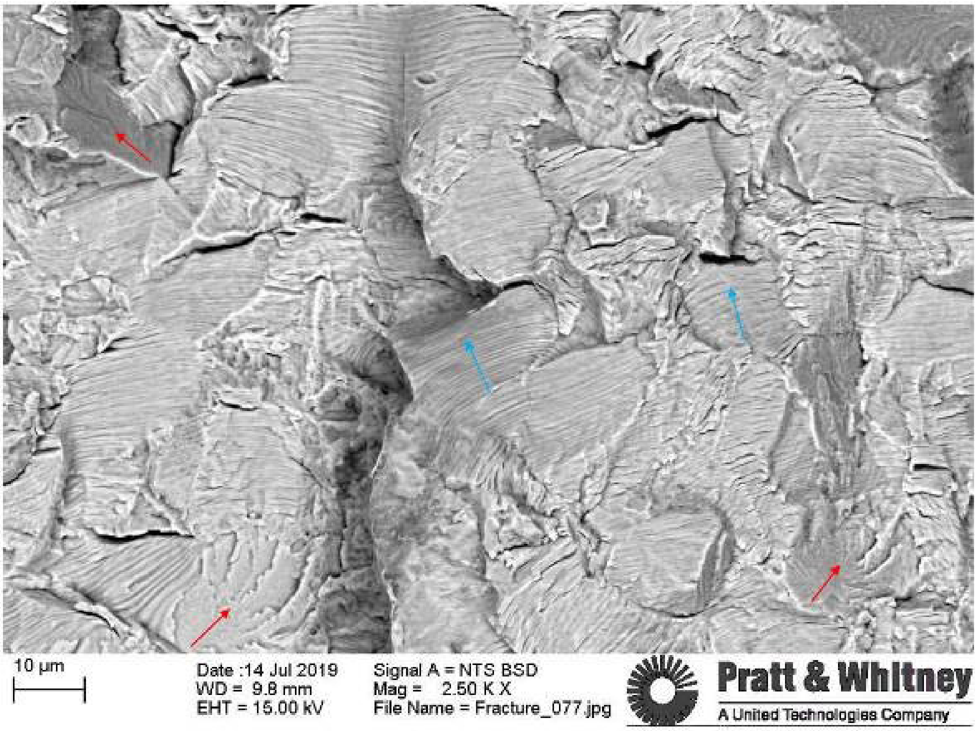


Figure 20: SEM view of a predominantly striated region. The crack progression direction in the striated regions is shown by the blue arrows. Near-cleavage facets were also present (red arrows).

Based on the fractographic examination that revealed the presence of a large faceted growth region, the crack started in a micro-textured region (MTR) 1.75 mm (0.069 inches) wide by 1.63 mm (0.064 inches) deep (Figure 21).

A metallographic cross-section was prepared by polishing lightly into the fracture surface. This cross-section underwent an Electron Back Scatter Diffraction (EBSD) analysis at the end of which a grain orientation map was obtained.

A strong correlation was found between the map showing a region with predominantly basal oriented α grain, perpendicular to the hoop stress (yellow dotted line in left image Figure 21) and the location of the predominantly faceted region (yellow dotted line, right image Figure 21).

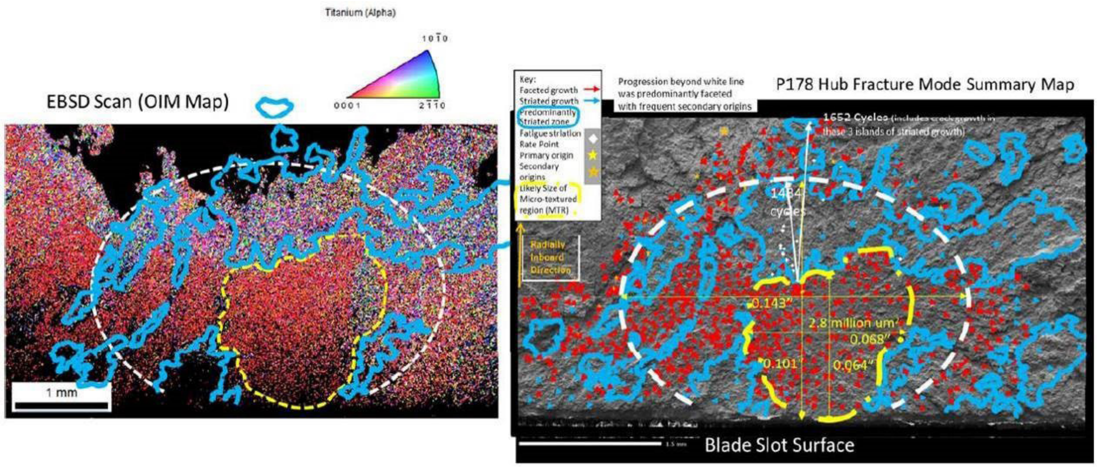


Figure 21: Origin planar micro of fracture showing the crystallographic orientation map (on left) obtained with EBSD after light polishing into the fracture surface. The red regions correspond to grains which have their basal planes oriented perpendicular to the hoop stress direction through the hub rim. This orientation is favourable to crack initiation under fatigue. The regions with predominantly near-cleavage facets, in red on right, and the macro-zone prediction (yellow dotted line) correspond to those regions where there is a high degree of basal oriented grains. The striation areas (blue edges) exhibit a more random crystallographic orientation.

Pure titanium has a compact hexagonal crystalline structure at ambient temperature (α phase). The crystallographic indexes make it possible to indicate certain characteristics of this structure. Thus, the base plane, also call the basal plane, is indicated by its crystallographic index (0001). It is shown in red in Figure 22. The basal direction is the perpendicular direction to the basal plane (direction *c[0001]* in Figure 22). Due to its hexagonal structure, the α phase is intrinsically anisotropic at the crystalline level which has significant consequences on the elastic and plastic properties of titanium and its alloys. The elasticity modulus of the α phase depends on the angle between the loading direction and the axis <c> of the crystalline lattice.

Above 882°C, titanium has a body centred cubic structure (β phase) up to its fusion temperature, 1,670°C. Thus, when it is heated to above 882°C, it passes from the α phase to the β phase. The alloy content and the thermomechanical processing during manufacturing determine the morphology and the fraction of the α and β phases of the microstructure. The transition temperature from the β phase to the α phase is called β-transus (Tβ) and depends on the composition of the alloy. At ambient temperature, Ti-6-4 has a α/β two-phase structure with a small volume percentage of residual β.

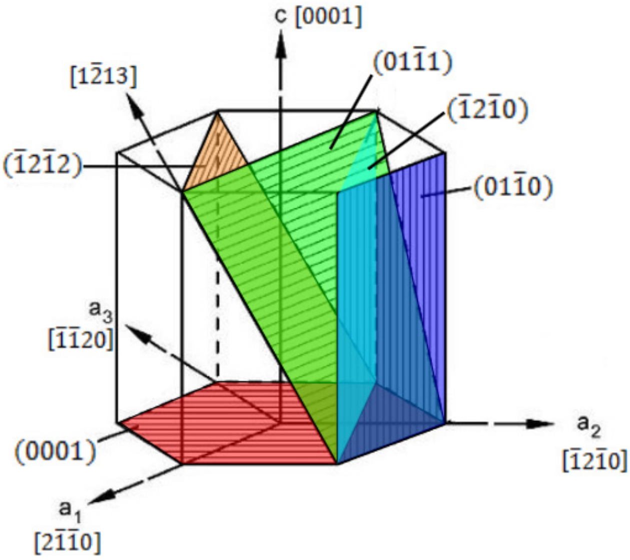


Figure 22: Crystallographic indexes of α phase, compact hexagonal structure

P&W's classification of the MTR associated with the origin of the failure confirmed that it was, according to its experience, larger and more intense[21] than the mean MTR statistics (maximum size of 1.1 x 10^6 μm^2 and a maximum intensity of 6.58).

The metallurgical examinations of this fragment found that the fracture was due to a cold dwell fatigue phenomenon. It originated in a macro-zone where the orientation of the grains is unfavourable with respect to the (hoop) maximum stress direction, in the middle of slot No 10. The crack progressed around 19.7 mm (0.775 inches) before becoming unstable.

No material quality (chemical composition, microstructure) or manufacturing related anomaly was found which could be associated with the area in which the fracture originated.

No evidence of damage arising from a maintenance activity was found on the front face of the hub in the vicinity of slots No 10 and No 18.

The various metallurgical and mechanical checks carried out during the investigation found that the material was consistent with properly processed Ti-6Al-4V alloy vs. existing P&W requirements for rotor-grade material.

(21) The intensity of an MTR as defined by P&W includes the following three values: the size of the macro-zone, the density of the α grains with the same alignment in the macro-zone and the deviation from the alignment with the basal direction in the macro-zone.

1.16.6 In-service inspections

In 2010, EA published Alert Service Bulletin (ASB) EAGP7-A72-139 requiring a one-time inspection of the fan hub in the scallop zones of the front face, to check for nicks, dents or scratches. The reason for this was that damage up to 0.229 mm (0.009 inches) deep had been found on a fan hub when withdrawn from the fleet for a maintenance action. The damage was located in a high-stress zone and was out of tolerance. An analysis showed that damage of this size could lead to the development of a crack if the hub stayed in service. The hubs found with this type of damage were to be withdrawn from service pending a repair method being introduced.

Following the accident to F-HPJE, Engine Alliance, the FAA and EASA published several documents to reinforce these inspections and to provide the operators of the GP7000 engines with information and instructions. These documents are shown in Figure 23.

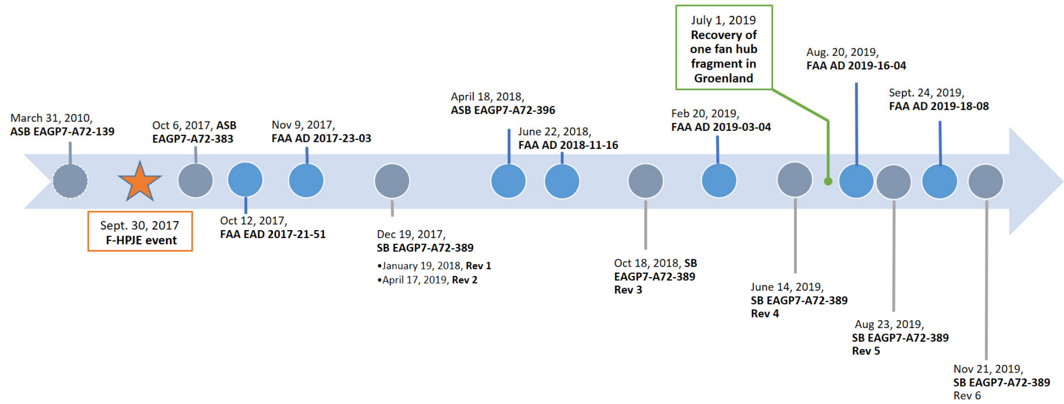


Figure 23: Publication sequence of documents concerning fan hub inspection (time scale not respected)

On 6 October 2017, EA published Alert Service Bulletin EAGP7-A72-383 which required a one-time visual inspection of the fan hub to search for damage such as nicks, dents or scratches on the visible parts of the hub after removal of the engine inlet cone. The hubs with damage outside the limits given in the ASB were to be removed from service and the damage assessed by EA. This ASB superseded ASB EAGP7-A72-139 issued in 2010.

On 12 October 2017, EA published revision 1 of this ASB in order to incorporate modifications resulting from the feedback from the first inspections and to provide new limits for the damage observed in certain zones. The same day, the FAA published emergency airworthiness directive EAD 2017-21-51, with immediate effect and which referred to the Engine Alliance ASB A72-383. This AD set out the requirements of the ASB and required a one-time visual inspection of all GP7200 engine fan hubs. The compliance time was based on the total accumulated flight cycles. All fan hubs with damage outside of serviceable limits were to be removed from service. This Emergency AD was adopted by EASA under the provisions of ED Decision 02/2003.

On 9 November 2017, the FAA published AD 2017-23-03 which superseded AD 2017-21-51. This new AD only introduced editorial changes with respect to AD 2017-21-51. However, it was published with an effective date of 24 November 2017 which extended the compliance time by an additional six weeks. Although EASA agreed with the technical content of the new AD, it did not agree with the introduction of a new effective date and the consequent extension of the compliance time. For this reason, EASA did not adopt this new AD. For engines installed on aeroplanes registered in EASA Member States, the Emergency AD 2017-21-51 remained valid.

On 19 December 2017, EA published SB EAGP7-A72-389 which required operators to carry out a one-time Eddy current inspection (ECI) of the slot bottoms of the fan hub to search for cracks and to carry out a visual inspection of the hub to search for damage. Hubs with damage or ECI indications outside the specified limits were to be removed from service or repaired.

On 19 January 2018, revision 1 was published in order to add a one-time Eddy current inspection of the front edge of the blade slots and to specify that the inspection could be carried out either with the fan hubs installed on an engine on-wing or off-wing, on fan hubs at the piece part level or on fan hubs at the LP compressor module assembly level. Revision 2 was published on 17 April 2018 to add some hub serial numbers in the "*conformity*" section, modify some damage limits and to add a one-time visual inspection of the bore area behind the balance flange weights.

On 18 April 2018, EA published Alert SB EAGP7-A72-396 which required a non-destructive inspection of the fan hub to search for damage, requiring an additional minimal disassembly of the LP compressor module. Compliance with this Service Bulletin was repetitive, namely on each workshop inspection of the engine or LP compressor module, which met the three criteria in the conformity section.

On 22 June 2018, the FAA published AD 2018-11-16 with an effective date of 2 July 2018. This AD required additional visual inspections of the GP7200 engines to those required by AD 2017-23-03. It also asked for an Eddy current inspection of the blade slot bottoms and front edges to search for cracks, which did not exist in AD 2017-23-03, and the removal from service of parts with damage outside the specified limits. This AD was adopted by EASA under the provisions of ED Decision 02/2003.

On 18 October 2018, revision 3 of SB EAGP7-A72-389 was published to add fan hub serial numbers to the applicability section and some information in the compliance section.

On 20 February 2019, the FAA published AD 2019-03-04, effective from 6 March 2019, which superseded AD 2018-11-16 for all the EA GP7270 and GP7277 engines with a certain fan hub assembly. AD 2019-03-04 retained the inspection requirements of AD 2018-11-16 but extended the inspections to a wider population of fan modules and revised the compliance time for these inspections. The reason for the publication of this AD was the FAA's determination that inspections needed to be expanded to all EA GP7270 and GP7277 turbofan engines. This AD was adopted by EASA under the provisions of ED Decision 02/2003.

On 14 June 2019, revision 4 of SB EAGP7-A72-389 was published to correct certain serial numbers in the previous version. The engine No 4 fan hub was found at the end of June 2019 and the metallurgical examinations were started on this hub at the beginning of July 2019.

On 20 August 2019, the FAA issued AD 2019-16-04 which superseded AD 2019-03-04 for all EA GP7270 and GP7277 model turbofan engines with a certain engine fan hub assembly installed. For certain GP7270 and GP7277 model turbofan engines, this AD continued to require a one-time ECI of the engine fan hub blade slot bottom and blade slot front edge for cracks and a visual inspection of the engine fan hub assembly for damage. For all GP7270 and GP7277 model turbofan engines, this AD also required an independent inspection of the engine fan hub assembly prior to reassembly of the engine fan hub blade lock assembly. For certain serial-numbered GP7270 and GP7277 model turbofan engines, this AD required replacement of the engine fan hub blade lock assembly. This AD was adopted by EASA under the provisions of ED Decision 02/2003.

On 23 August 2019, revision 5 of SB EAGP7-A72-389 was published in order to change the inspection calendar requirements from a one-time inspection to a repetitive inspection. It also modified the initial inspection limit, the list of hubs to be inspected (specific list replaced by all PN 5760221 and PN 5760321 part numbers) and withdrew the visual inspection requirements which were, by this time, incorporated in the Airbus Aircraft Maintenance Manual and in the Engine Manual. The compliance time requirements called for the Eddy current inspection to be carried out by 1,700 cycles since new, by 150 cycles (as of 1 September 2019), by 330 cycles since the last ECI or by 330 cycles since the last maintenance, whichever is the later applying. The Eddy current inspection was then to be carried out every 330 cycles.

On 24 September 2019, the FAA issued AD 2019-18-08 which superseded AD 2019-16-04 for all EA GP7270 and GP7277 model turbofan engines. This AD, for certain GP7270 and GP7277 model turbofan engines, reduced the compliance time for the initial ECI and required repetitive ECIs of the engine fan hub blade slot bottom and blade slot front edge for cracks. This AD also retained the visual inspection requirements of the engine fan hub assembly for all GP7270 and GP7277 model turbofan engines. This AD was adopted by EASA under the provisions of ED Decision 02/2003.

On 21 November 2019, revision 6 of SB EAGP7-A72-389 was published to add an ultrasonic inspection of the rim. The ultrasonic inspection requirements were the same as those for the Eddy current inspections, and were the same as those set out in revision 5.

This inspection campaign carried out on the hubs of the A380 fleet equipped with EA engines, in compliance with these SB, revealed the presence of mechanical damage on the front face of several hubs, notably in the scallop zones which are areas with a high concentration of stress. This damage may have been caused by the use of tools or inappropriate practices during maintenance operations. The maximum size of the reported damage was to a depth of 0.36 mm (0.014 inches). Corrective actions were introduced by EA in order to reduce the risk of causing hub front face damage (see paragraph 4.3). Eddy current indications were obtained during the inspections. These were generally due to wear on the slot bottom. The possible reduction in fatigue strength linked to the presence of this wear was the subject of a specific assessment by testing specimens in LCF.

1.17 Organizational and Management information

Not applicable.

1.18 Additional Information

1.18.1 Fan hub sizing principles

1.18.1.1 Certification requirements

The GP7200 engine was certified in 2005, in accordance with the requirements of the FAA Code of Federal Regulations (CFR)[22], in force at the time of the certification. These standards will be subsequently referred to as "*Part 33*". With respect to the Part 33 requirements relating to rotors, excerpts from the relevant regulations in the scope of the F-HPJE occurrence and in compliance with the engine certification date are listed below:

Section 33.7 Engine ratings and operating limitations.	"(a) Engine ratings and operating limitations are established by the Administrator and included in the engine certificate data sheet specified in § 21.41 of this chapter, including ratings and limitations based on the operating conditions and information specified in this section, as applicable, and any other information found necessary for safe operation of the engine.[…] (c) For turbine engines, ratings and operating limitations are established relating to the following: […] (12) The number of start-stop stress cycles approved for each rotor disc and spacer. [...]"
Section 33.14 Start-stop cyclic stress (low-cycle fatigue). Amdt. 33-10, Eff. 3/26/84	"By a procedure approved by the FAA, operating limitations must be established which specify the maximum allowable number of start-stop stress cycles for each rotor structural part (such as discs, spacers, hubs, and shafts of the compressors and turbines), the failure of which could produce a hazard to the aircraft. A start-stop stress cycle consists of a flight cycle profile or an equivalent representation of engine usage. It includes starting the engine, accelerating to maximum rated power or thrust, decelerating, and stopping."
Advisory circulars **Section 33.14-1** Damage Tolerance for High Energy Turbine Engine Rotors Date 01-08-2001	This advisory circular describes an acceptable means for showing compliance with the requirements of section 33.14 of the Federal Aviation Regulations (Title 14, Code of Federal Regulations). Section 33.14 contains requirements applicable to the design and life management of high energy rotating parts of aircraft gas turbine engines.
33.15 Materials. Amdt. 33–8, 42 FR 15047, Mar. 17, 1977, as amended by Amdt. 33–10, 49 FR 6850, Feb. 23, 1984	§33.15 Materials. The suitability and durability of materials used in the engine must— (a) Be established on the basis of experience or tests; and (b) Conform to approved specifications (such as industry or military specifications) that ensure their having the strength and other properties assumed in the design data.
Advisory circulars: **33.15-1 -** Manufacturing Process of Premium Quality Titanium Alloy Rotating Engine Components	This advisory circular provides guidance for compliance with the provisions under Title 14 under the Code of Federal Regulations, Part 33 (14 CFR 33) pertaining to the materials suitability and durability requirements, 33.15, as applicable to the manufacture of titanium alloy high energy rotating parts of aircraft engines.
Section 33.19 Durability. Amdt. 33-10, Eff. 3/26/84	"(a) Engine design and construction must minimize the development of an unsafe condition of the engine between overhaul periods. The design of the compressor and turbine rotor cases must provide for the containment of damage from rotor blade failure. Energy levels and trajectories of fragments resulting from rotor blade failure that lie outside the compressor and turbine rotor cases must be defined."

[22] Title 14 (Aeronautics and Space), Chapter I (FEDERAL AVIATION ADMINISTRATION, DEPARTMENT OF TRANSPORTATION), Sub-chapter C (AIRCRAFT), Part 33 (AIRWORTHINESS STANDARDS: AIRCRAFT ENGINES).

Section 33.27 Turbine, compressor, fan, and turbosupercharger rotors. Amdt. 33-10, Eff. 3/26/84	"(a) Turbine, compressor, fan, and turbosupercharger rotors must have sufficient strength to withstand the test conditions specified in paragraph (c) of this section. (b) The design and functioning of engine control devices, systems, and instruments must give reasonable assurance that those engine operating limitations that affect turbine, compressor, and turbosupercharger rotor structural integrity will not be exceeded in service. (c) The most critically stressed rotor component (except blades) of each turbine, compressor, and fan, including integral drum rotors and centrifugal compressors in an engine or turbosupercharger, as determined by analysis or other acceptable means, must be tested for a period of 5 minutes-- (1) At its maximum operating temperature, except as provided in paragraph (c)(2)(iv) of this section; and (2) At the highest speed of the following, as applicable: (i) 120 percent of its maximum permissible r.p.m. if tested on a rig and equipped with blades or blade weights. (ii) 115 percent of its maximum permissible r.p.m. if tested on an engine. (iii) 115 percent of its maximum permissible r.p.m. if tested on turbosupercharger driven by a hot gas supply from a special burner rig. (iv) 120 percent of the r.p.m. at which, while cold spinning, it is subject to operating stresses that are equivalent to those induced at the maximum operating temperature and maximum permissible r.p.m. (v) 105 percent of the highest speed that would result from failure of the most critical component or system in a representative installation of the engine. (vi) The highest speed that would result from the failure of any component or system in a representative installation of the engine, in combination with any failure of a component or system that would not normally be detected during a routine preflight check or during normal flight operation.] Following the test, each rotor must be within approved dimensional limits for an overspeed condition and may not be cracked."
Section 33.62 Stress analysis. [Amdt. 33-6, 39 FR 35466, Oct. 1, 1974]	"A stress analysis must be performed on each turbine engine showing the design safety margin of each turbine engine rotor, spacer, and rotor shaft."
Section 33.75 Safety analysis. Amdt. 33-10, Eff. 3/26/84	"It must be shown by analysis that any probable malfunction or any probable single or multiple failure, or any probable improper operation of the engine will not cause the engine to-- (a) Catch fire; (b) Burst (release hazardous fragments through the engine case); (c) Generate loads greater than those ultimate loads specified in Sec. 33.23(a); or (d) Lose the capability of being shut down."

1.18.1.2 Low Cycle Fatigue (LCF)

During the certification of the GP7200 engine, stresses and their evolution during a flight (both their level and concentration factors) were calculated by P&W using 2D and 3D finite element models run through a flight cycle mission analysis. The low cycle fatigue lives resulting from these mission analyses were calculated using P&W's LCF lifing system, valid for Ti-6-4, developed by P&W and approved by the FAA. This system is based on P&W's accumulated specimen and component test data. The resulting predictions by this tool are statistical data in terms of B0.1[23] (i.e. 1 in 1,000 parts will have developed a 0.79 mm (1/32 inch) crack at predicted life).

P&W determined the critical zones of the hub in terms of LCF. These were the scallops, bolt holes and slot bottoms, as shown in Figure 24: critical zones in LCF (step climb). The maximum stress conditions in the hub occur during step climb, when the fan rpm reaches its maximum speed. Over 600 specimens and 17 components were used to create P&W's approved lifing system. According to the manufacturer, all specimens, when tested close to the maximum stress level of the critical zones, had lives above 58,000 cycles. Data was also available for stress levels above these maximum calculated values.

[23] The life to initiation B(X) is the time estimated necessary for the crack initiation probability to reach X%. E.g. the value B0.1 is the life at which 1/1,000 (0.1%) of the population will have developed a crack. Likewise, the value B50 is the average life, in other words 50% of the population will have developed a crack at this time.

The safe life[24] of the hub (i.e. LCF life to initiation of a 0.79 mm or 1/32 inch crack) was established to be 15,000 cycles based on B0.1 LCF data.

[24] The structure must have no detectable cracks for all of its service life.

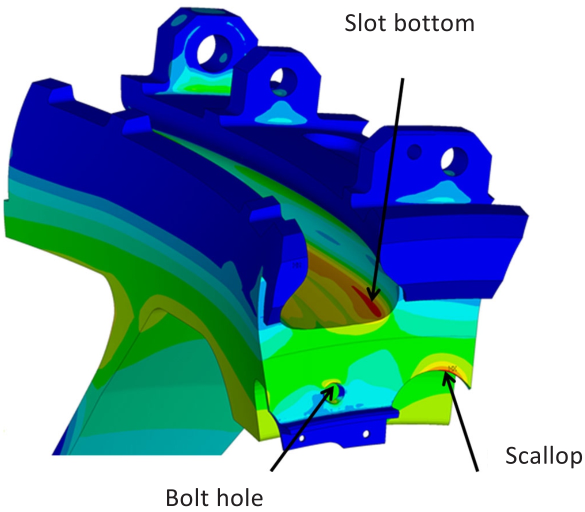


Figure 24: critical zones in LCF (step climb)

1.18.1.3 Burst margins

The GP7000 fan hub was certified analytically for burst using P&W's FAA approved methodology. This methodology was calibrated for the titanium alloy, Ti-6-4 on the basis of 30 overspeed tests. According to the manufacturer, these tests covered the critical zones of the engine hub, the range of temperatures encountered and the test speed ratio (ratio between the speed at which a previous test had been successfully carried out for five minutes and the burst speed).

The manufacturer stated that the analysis of the burst strength of the GP7200 engine hub in overspeed conditions predicted a burst margin of 136% above the maximum rotation speed authorized in service (Red Line), taking into account an additional safety margin. The FAA only requires (Part 33.27) that the part withstands 120% of the Red Line or 105% of the highest rotation speed resulting from a failure, as explained in paragraph Part 33.27(c) (see 1.18.1.1). In the case of the GP7200 engine, the highest rotor speed that can be reached is 100.2% of the Red Line, in the case of the LP shaft shearing. P&W calculated that the yield stress margin of safety was 1.35 and the ultimate stress margin safety was 1.60.

In terms of the data collected during the investigation, no evidence of an exceedance of the Red Line was recorded. It was therefore considered very unlikely that the hub had burst due to overspeed conditions.

1.18.1.4 Safe life parts and damage tolerance

At the time of the certification of the GP7270, AC 33.14-1[25] provided guidelines for integrating a damage-tolerance type approach in the certification process of safe-life parts such as high energy rotors. The aim was to improve the life management sizing of these parts. The AC only considered hard-α[26] type manufacturing anomalies. According to P&W, the hub of the GP7200 complied with the recommendations of AC33.14-1.

In the scope of the investigation and before the hub fragment was found in Greenland, a crack progression analysis was carried out by the manufacturer in order to determine the minimum initial size of a defect required for a failure to occur at 3,500 cycles if the defect was located at 0.76 mm (0.030 inches) below the surface. The static and vibrating stresses along with the crosswind conditions were considered for this analysis. The minimum size of the defect in the slot bottom was 0.91 mm (0.036 inches). According to the graphs in the FAA AC33-14.1, the probability of having a defect of 0.91 mm in the hub is 1 in 10,400.

The same assessment was carried out in the scallop zones but this time taking into account surface damage. The greatest damage reported in inspections was 0.36 mm (0.014 inches) deep. The number of crack progression cycles required to arrive at a hub failure, starting from such damage, was estimated at 4,800 cycles. The minimum defect size for failure in 3,500 cycles was also calculated, supposing the presence of an active crack in the first cycle. The result was 0.38 mm (0.015 inches) deep. These results were obtained using average statistical values (B50).

1.18.2 Check of fan hub production

In order to determine if any manufacturing defect could have led to the hub failure, the manufacturing history and its conformity with the manufacturer's requirements were investigated.

There are four primary steps to manufacture a GP7270 titanium fan hub:

(1) **Fusion**: a combination of alloy source materials are melted together to make a titanium alloy ingot. This ingot is triple melted using the vacuum arc remelting (VAR) process. The diameter of the ingot obtained is 86 cm (34 inches).
(2) **Conversion into a billet**: the ingot is then converted into a billet with a diameter of 36 cm (14 inches) using a thermo-mechanical reduction or conversion process. Due to the resulting elongation of the billet during this process, it is initially cut into two sections (Top and Bottom) to facilitate continued reduction to the required 36 cm. The billet outer surfaces are machined to facilitate ultrasonic inspection.
(3) **Forging**: the billet is cut into smaller sections (mults) which are then forged. The forging process includes a series of upsetting steps and a final closed die forging step. The occurrence hub was forged according to a manufacturing process known as Near Net Shape (NNS). This was the second change to the forging process since its initial configuration. The current process known as FPF (Final Production Forging) is the third forging configuration change. The fan hub forging is then machined to a recti-linear (pre-machined) shape for ultrasonic inspection.

[25] Advisory Circular AC33.14-1 - Damage Tolerance for High Energy Turbine Engine Rotors to the appearance of cracks in the defect and/or in its direct vicinity during shaping operations.

[26] Also known as HID (High Interstitial Defects), hard-α inclusions are small volumes of metal enriched with alpha interstitial elements such as oxygen and nitrogen. These defects are characterized by high hardness or brittleness with respect to the surrounding titanium matrix which contributes to the appearance of cracks in the defect and/or in its direct vicinity during shaping operations.

(4) **Machining/finishing**: the last step involves machining the obtained product into the part's final shape and performing the associated finishing operations and inspections.

A unique serial number is assigned to each part. Hubs manufactured from the same billet as the occurrence hub are called sister hubs. They were identified, removed from service, isolated and inspected. The fan hub of engine No 4 of the occurrence flight was manufactured from the top mult of the billet, called T1. Metallurgical examinations were performed on the T2 hub, whose mult was the closest to the T1 (occurrence) hub. This hub was considered as the closest in terms of material properties to the hub of the occurrence. Other examinations and tests were carried out on the bottom hubs (B1, B2 and B3) from the same heat.

No anomaly in the engine No 4 hub manufacturing process was found in the course of the investigation.

At the time of publication of this safety investigation report, there is currently no industry standard to quantify or check for the presence of macro-zones (see paragraph 1.18.3). No major change in the forging process was made following the discovery of the cause of the hub failure. During inspections, six indications above #1 FBH[27] were detected in the bore or rim of hub T2. These indications were further evaluated by means of a focused immersion ultrasonic inspection. They were then requalified as "*acceptable*". Mechanical damage up to a depth of 0.15 mm (0.006 inches) was observed on the hub B3 front face. Very slight wear up to a depth of 0.076 mm was observed in the blade slot bottoms of hub B3. Lastly, superficial scuffing was observed in the hub T2 slot bottoms.

[27] FBH stands for flat bottom hole. #1 FBH has a diameter of 1/64th inch, or 0.015 inch or 0.40 mm.

No material nor mechanical property anomaly was detected on the sister hubs.

1.18.3 Presence of macro-zones (micro-texture regions) in titanium

The microstructure of the titanium alloy, Ti-6-4, is characterized by the shape, size, proportion and texture of its α and β phases. It depends to a great extent on the manufacturing method and heat treatments applied. The texture can be defined by the distribution of the grain crystallographic orientations of a phase. As the latter is never totally homogeneous, more or less textured zones are naturally present in a material. A region with a strong texture (i.e. an aggregate of grains with a preferred crystallographic orientation) which can be millimetric in size, is called a micro-texture region (MTR) or macro-zone. The EBSD[28] technique is used to characterize the grain crystallographic orientation and thus reveal these macro-zones (Figure 25). These macro-zones can then be classified in order to assess their size and their severity. The presence of macro-zones is inherent to the manufacturing process of forged titanium parts. They originate during the ingot to billet conversion process. They are then reduced in size and density during the part forging process by means of various successive thermomechanical treatments. The direct consequence of microstructural heterogeneities is to produce a high local anisotropy of the mechanical properties.

[28] Electron Back Scatter Diffraction.

Such variations in the properties affect the fatigue and cold dwell fatigue life dispersion (see paragraph 1.18.4). Furthermore, MTR are likely zones for fatigue or cold dwell fatigue crack initiation.

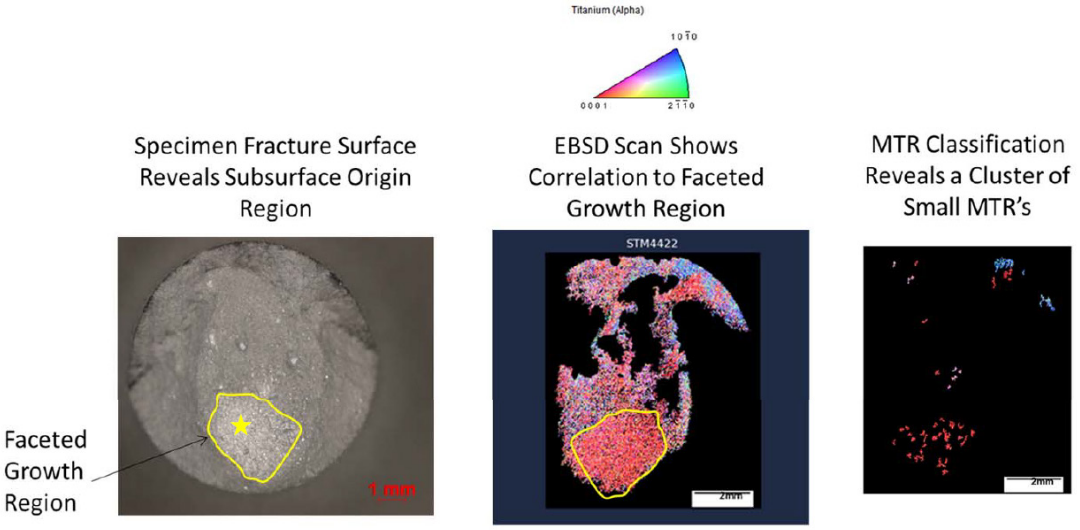


Figure 25: fracture surface of Ti-6-4 test specimen after cold dwell fatigue test (2 minute dwell) on left, with a subsurface fatigue progression region (outlined in yellow). The start of the crack is shown by a yellow star. The crystallographic orientation map obtained with EBSD (centre) was obtained on the opposite fracture surface. It is thus a mirror image of the left figure. The classification of the macro-zones is shown in the right image. The fatigue zone has macro-zones of an average size of 1.8×10^4 µm^2, of a maximum size of 7.8×10^4 µm^2 and of a maximum intensity of 0.19.

1.18.4 Cold dwell fatigue phenomenon

Metals in temperatures greater than around one third of their fusion temperature can deform by creep when they are subject to a constant load. However, titanium alloys are known to creep at ambient temperature[29].

This "*cold creep*" deformation mode, also called cold dwell or dwell was the cause of premature failures of fan disks at the beginning of the 1970s. In 1972, one of the Rolls Royce RB211 engines equipping a Lockheed 1011 Tristar failed due to its fan disk fracturing. In 1973, several premature failures also occurred to fan disks made of titanium alloy IMI 685.

The analysis of these occurrences found that the failure of the components due to cyclic mechanisms was significantly accelerated by loads being maintained during the flight. This gave rise to the term "*dwell*". Fatigue type loading with maintained stress is called dwell fatigue. Subsequently, extensive research was started to characterize the influence of the composition, microstructure and texture of the alloy on the initiation and progression of cracks in cold dwell fatigue.

Scientific literature seems to confirm that several factors affect the sensitivity of titanium alloys to the cold dwell fatigue effect[30]. Generally speaking, α or near-α alloys with a small β-phase volume fraction, such as Ti-6242, IMI 685, IMI 829 and IMI 834, studied, in particular, for high-temperature applications, are generally more sensitive to the cold dwell fatigue effect. α/β or β alloys such as Ti-6-4, Ti-6246 or Ti17 were, on the other hand, considered as having little or no sensitivity to cold dwell fatigue[31].

(29) M. R. Bache, A review of dwell sensitive fatigue in titanium alloys: the role of microstructure, texture and operating conditions, Int. J. Fatigue, vol. 25, pp. 1079-1087, 2003.

(30) Mechanics and micro-mechanisms of LCF and dwell fatigue in Ti-6Al-4V, P. Tympel, PhD thesis, 2016.

(31) Effet dwell: relation microstructure-microtexture-propriétés mécaniques de l'alliage de titane Ti-6242, Immanuel Freiherr von Thungen, PhD thesis, 2016.

Likewise, the stress level applied influences the cold dwell fatigue effect. When the stress applied in cold dwell fatigue exceeds the yield strength, the number of cycles to failure is drastically reduced with respect to pure fatigue loading. When the stress is situated at around 90% of the yield strength, the life of Ti-6242, for example, is reduced by a factor of 2 while at 105% of the yield strength, this factor increases to 30. Results are contradictory regarding the duration of the dwell. Some report a reduction in cold dwell fatigue life with the increase in dwell time. Others, to the contrary, observe a saturation of the cold dwell fatigue effect after 40 seconds. At the time of the certification of the GP7200 engine, the Ti-6-4 alloy was considered insensitive to cold dwell fatigue by the scientific community[32], the industry and the certification authorities. This was in part due to the significant number of service hours logged by Ti-6-4 components without any incidents, unlike the Ti-6242, IMI 685 and IMI 834, and in part due to its chemical composition placing it with α/β alloys. In addition, although there had been in-depth research into this phenomenon with respect to certain alloys, the mechanisms at the origin of the initiation of a cold dwell fatigue crack were still not completely understood.

[32] Titanium, Luetjering G. Williams, 2e edition, 2007.

The results from a component test carried out by an engine manufacturer in 2007 gave rise to internal studies and to the development of its own criteria in order to take into account the effects of cold dwell fatigue on the titanium alloy Ti-6-4. In 2010, this manufacturer modified the loading limits of its Ti-6-4 alloy parts. EASA became aware of this during the investigation. No other formal communication concerning the identification of this phenomenon was shared during this period, with the certification authorities or in international work groups. It is not certain that if this had been the case, there would have been a generalized modification of the processes.

According to P&W, the service life sizing of the GP7200 fan hub took into account the location of specific operational stresses and the yield strength of the material according to the operational temperature, in specific locations on the part. Distribution of macro-zone sizes or dwell time were not considered at that time.

Cold dwell fatigue tests are typically carried out at 120 or 125 ksi (827 or 862 MPa) at ambient temperature. However, the maximum stresses on the part are well below this.

In addition, the fan hub is operated between 21°C and 70°C (70°F to 160 °F), whereas the dwell effect is supposed to decrease as the temperature increases and disappear on the temperature exceeding around 200°C (392°F).

After the accident, the manufacturer wanted to re-assess the cold dwell fatigue effect on Ti-6-4 and carried out tests on samples taken from hubs from the same forging configuration as that of the engine No. 4 hub. P&W carried out LCF tests with and without dwell. An EBSD analysis was carried out on the fractured specimens in order to characterize and classify their macro-zones. The results were compared with P&W's existing LCF life calculation system. They showed life debits accentuated by the dwell. The life was reduced by a factor of 6.1 between tests with a dwell of two minutes and pure fatigue tests without dwell.

At the site where the hub fracture started, the maximum in-service stress is below 80% of the yield strength[33]. P&W did not observe any cold dwell failures during dwell fatigue tests carried out at stresses below 85% of the yield strength. The predicted life in this zone with the value B0.1, without taking into account the dwell effect, is 38,500 cycles. The number of cycles to initiation (number of cycles to failure less the number of crack progression cycles measured) was estimated at around 1,880 cycles for the F-HPJE hub.

This means that a debit of a factor of 20 is necessary to explain the initiation of a crack in the blade slot bottom in 1,880 cycles, compared to a debit factor of 6.1 observed during the tests on specimens.

The test samples generally used are such that only a small volume of material is tested in comparison with the actual part. In this respect, they show average typical debits due to macro-zones of an average size and severity that can be found in these specimens. The debits obtained are not necessarily representative of those which may be associated with macro-zones observable in actual parts, by a simple scale effect.

To take into account the cold dwell fatigue effect for large volumes, EA is in the process of developing a visco-plastic model to calculate LCF life which considers a macro-zone size distribution in a specimen. Simulations carried out with this model show that a dwell time of ten minutes increases the life debit by 30% compared to a dwell time of two minutes. The final purpose of this development is to calculate the hub life taking into account the typical missions of the GP7200 engine and the macro-zone distribution which depends on the hub manufacturing process.

1.18.5 In-service occurrences involving a cold dwell fatigue phenomenon

1.18.5.1 Fan blades[34]

On 13 February 2018, the Boeing 777 registered N773UA performing United Airlines flight 1175 suffered a failure on a right P&W PW4077 engine, leading to the loss of the air inlet and cowlings of this engine during the descent to Honolulu airport (Hawaii). The NTSB's preliminary report specified that the engine failure was due to the failure of a fan blade made of the Ti-6-4 alloy. The examinations carried out by the NTSB revealed that the blade failure was produced by a cracking process due to cold dwell fatigue, starting from an initiation site slightly subsurface. The SEM examinations revealed the presence of macro-zones close to the origin of the failure.

On 10 March 2019, the A380 registered F-HPJC, performing Air France flight AF703, suffered a failure of one of its GP7270 engines during the climb from Abidjan airport (Ivory Coast). The crew landed at Abidjan. The engine failure was caused by the failure of a Ti-6-4 fan blade, at approximately mid-height, due to cold dwell fatigue. The examinations found that the initiation of the crack was in a macro-zone.

On 30 December 2019, the FAA published an airworthiness directive[35] concerning all the GP7270 and GP7277 engines. This AD requires an ultrasonic inspection of the fan blades and the replacement of any fan blades that fail the inspection.

(33) P&W specify that the minimum acceptable value for the part's yield strength is 125 ksi. It is checked during tensile tests on specimens representative of the part in various stages of its manufacturing process.

(34) The fan blades referenced in this section were produced with a cross-rolled plate manufacturing process for the raw material, which is different to the rotor forging process. It is known to be more prone to producing large MTR zones in the finished product.

(35) AD 2019-25-13: https://www.federalregister.gov/documents/2019/12/30/2019-27889/airworthiness-directives-engine-alliance-turbofan-engines.

1.18.5.2 Disks or hubs

The first accidents attributed to the cold dwell fatigue phenomenon occurred on Lockheed 1011 Tristar aircraft[36] (see paragraph 1.18.4). These were equipped with Rolls Royce RB211 engines put into service at the beginning of 1972. Barely a year later, there were several premature failures of fan disks made of titanium alloy IMI 685.

Since the Tristar, at least four other incidents to civil aeroplanes have been attributed to the cold dwell fatigue effect on compressor hubs made of Ti-6242 on GE CF6 engines: in 1985 at Dakar, in 1991 at Seoul, in 1993 at Los Angeles and in 1995 at Bangkok[37]. During the 1985 incident, a CF6-50 engine, certified for 15,000 cycles, suffered the failure of the 9th compressor disk after only 4,075 flight cycles[38]. In the other cases, the same type of premature failure was identified. The recommendations issued by the NTSB and the FAA (Airworthiness Directive 91-20-1 dated 25 October 1991) led to the inspection of a large proportion of the parts still in service. The ultrasonic and Eddy current inspections carried out by GE detected before failure 16 HP compressor bodies (stages 3 to 9) containing cracks attributed to the cold dwell fatigue. No in-service failure of a Ti-6-4 disk or hub that could be attributed to this effect had been reported before that of F-HPJE.

1.19 Useful or effective investigation techniques

The means and methods used to search for the fan hub are described in the reports dedicated to the Greenland searches.

https://www.bea.aero/uploads/tx_elyextendttnews/F-HPJE_TECHNICAL_REPORT.pdf

and

https://www.bea.aero/uploads/tx_elyextendttnews/F-HPJE_Phase_III_PUBLICATION_June_2020.pdf

[36] Immanuel Freiherr von Thungen. Effet dwell: relation microstructure-microtexture-propriétés mécaniques de l'alliage de titane Ti-6242. PhD Thesis.

[37] J. F. Garvey, Safety Recommendation, A-98-27 through -33, Washington, D.C. 20594, 1998.

[38] J. E. Hall, AIRCRAFT ACCIDENT REPORT: Uncontained Engine Failure DELTA AIR LINES Flight 1288 MCDONNELL DOUGLAS MD-88, Washington, D.C. 20594, 1996.

2 - ANALYSIS

2.1 Introduction

En route over the south of Greenland, while climbing from FL 370 to FL 380, F-HPJE suffered an uncontained failure of an Engine Alliance GP7270 engine installed in the right outer position (engine No 4). Two large fragments of the Ti-6-4 fan hub were radially ejected, one upwards and one downwards. The interaction between the liberated rotor fragments and both the engine casing and the air inlet caused the in-flight separation of these engine parts. The low pressure shaft and other integral assemblies locked. All of the damage generated significant additional drag. The structure of the airframe nearby was only slightly damaged and this did not affect the continuation of the flight.

The crew detected, analysed and processed the failure based on the operational standard procedures. The management of the vertical profile of the path to the driftdown level was complicated due to the suddenness of the event and subsequent damage affecting the performance, at a level not anticipated by the crew.

The crew chose to divert to Goose Bay aerodrome (Canada) where they landed without any further problem.

In order to guarantee the airworthiness of the other engines in operation, in the absence of confirmation of the failure mode, the manufacturer and the certification authorities required inspections of the fan hubs to be carried out just after the accident. These observations found a number of fan hubs with surface damage, giving rise to a probable scenario linked to an inappropriate maintenance operation.

The perseverance in carrying out the search operations resulted in the finding and examination of a piece of fan hub debris twenty-one months after the accident. The results of the examinations invalidated the maintenance damage scenario considered the most likely up to this point and showed a failure mode which was originally ruled out as it was considered as highly unlikely.

This failure mode had already been seen on other titanium alloys, however, no titanium Ti-6-4 hub had failed in service under cold dwell fatigue before this on commercial airplanes.

The analysis below principally concerns the failure of engine No 4. It is structured so as to explain the failure mode of the fan hub. The factors which caused this failure along with the measures implemented during the design to prevent failure in a certified operating envelope are also set out in detail.

This accident revealed a failure phenomenon which had not been observed on the titanium alloy, Ti-6-4 (cold dwell fatigue) and was difficult to anticipate. The factors conducive to its appearance are being studied by manufacturers and need to be analysed in detail.

2.2 Engine No 4 failure

This uncontained failure was the result of the failure of the fan hub subject to a cold dwell fatigue mechanism (see paragraph 1.18.4), originating in a macro-zone present in the subsurface of a hub blade slot (blade root housing).

The failure occurred during a normal operational phase (cruise climb) without the appearance of any precursory event. The radial (upward and downward) expelling of the two fan hub high-energy fragments at the time of the separation was random. A lateral ejection (left and right) of the same debris might have had catastrophic consequences, for instance if the airframe had been pierced or if the wing spar, control surfaces or flight control systems had been damaged.

Due to fatigue, a crack was initiated and progressed in the subsurface of a hub blade slot, opposite the blade root. This crack progressed for around 1,650 cycles, the part failing at around 3,500 cycles since new, i.e. four times earlier than the minimum life shown by the designer for this titanium part (15,000 cycles).

The metallurgical examination of the failure confirmed that no maintenance induced damage was at the origin of the start and progression of this crack. Moreover, no scheduled maintenance action, at the time of the accident, would have detected it while it was still below the surface of the slot bottom.

2.3 Damages during maintenance operations

The design and certification principle of critical parts is to remove them from service at the end of an approved life before hazardous effects can occur, such as the initiation of cracks, which may propagate to cause failure.

The feedback from in-service inspections carried out on fan hubs, after the accident, revealed damage on the hub front face of certain engines. In all likelihood it can be attributed to practices or the use of tools which do not comply with the maintenance procedures during the removal and installation of the blade lock ring. The rigidity of this ring makes the removal and installation operation difficult, even when using the appropriate tools.

The simulations carried out by the manufacturer during the investigation showed that the size and location of this damage, in zones of high concentration of stress, were compatible with the scenario of a fan hub failure in a way similar to that observed in the case of F-HPJE. The size of the surface damage required for a crack to initiate and propagate to failure, in a number of cycles comparable to that of F-HPJE engine No 4, was of the same order of magnitude as certain damage observed during these post-accident inspections. This scenario was initially considered the most likely before being ruled out after finding the fan hub fragment in Greenland. It remains, however, an undesirable scenario against which measures have been taken.

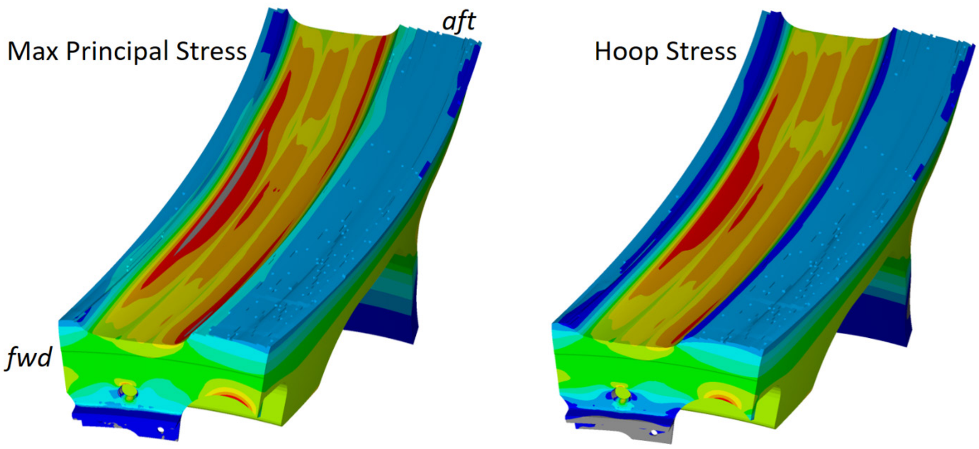
2.4 Fan hub sizing and taking into account cold dwell fatigue

2.4.1 Fan hub sizing

The fracture started in the subsurface slot bottom, close to the centre of the slot. The crack progressed perpendicularly to the hoop stresses caused by the centrifugal loads applied to the hub. This zone is not the zone with the highest stress. The critical zones in LCF defined by the manufacturer are situated on the hub front face (Figure 24) where maximum stresses are locally a lot greater. However, the crack's initiation area corresponds to a local high stress region (Figure 26). The presence of a subsurface macro-zone in the slot bottom probably contributed to the initiation of a crack in this zone which is subject to lower stresses than those of the critical zones.

The search to improve performance has led manufacturers to develop engines with a higher bypass ratio, which has led to the design of very large diameter fans. This pushes them to try to limit the weight of these critical parts as much as possible.

The present day digital simulation and design tools allow parts to be designed as accurately as possible, by optimizing the margins. Thus, to obtain the design objectives, it may prove necessary to move outside the usual operating envelope known through in-service experience (stress, time, temperature). This condition can lead to the development of physical phenomena less fully anticipated.



Source: EA

Figure 26: Maximal principal stress (left) and hoop stress (right) in a slot bottom

2.4.2 Damage tolerance

At the time of the certification of the GP7270 engine, AC 33.14-1[39] provided guidelines for integrating a "*damage tolerance*" type approach in the certification process of safe-life parts such as high energy rotors. The aim was to improve the sizing and life management of these parts. The circular only considered the hard-α type manufacturing anomalies. It did not deal with surface damage, caused mechanically for example.

In 2009, i.e. after the certification of the GP7270, the FAA published a circular, AC33.70-1[40] which required a deterministic calculation of surface damage tolerance to show a crack progression life of 3,000 cycles assuming an initial crack 0.38 mm (0.015 inches) deep. Even though it was not required, EA showed that the fan hub sizing met this requirement.

[39] Advisory Circular AC33.14-1 - Damage Tolerance for High Energy Turbine Engine Rotors.

[40] Advisory Circular AC33.70-1 - Guidance Material for Aircraft Engine Life-Limited Parts Requirements.

Macro-zones are inherent to the manufacturing process of Ti-6-4 forged parts and as such cannot be considered as defects. However, the presence of a macro-zone increases the risk of crack initiation under the cold dwell fatigue phenomenon.

There are no requirements, guides or other appropriate regulation chapters at the present time which deal with taking into account macro-zones in the demonstration of damage tolerance.

2.4.3 Knowledge of cold dwell fatigue phenomenon and taking it into account in design and certification

At the time of the certification of the GP7200 engine, the Ti-6-4 alloy was considered insensitive to cold dwell by the scientific community (see paragraph 1.18.4), the industry and the certification authorities. This was in part due to the significant number of service hours logged by Ti-6-4 components without any incidents, unlike those made from Ti-6242, IMI 685 and IMI 834, and in part due to its chemical composition placing it with α/β alloys (see paragraph 1.16.5).

Carrying out cold dwell fatigue tests requires a longer period of time than for fatigue tests, as the stress is held for a given duration. The cold dwell fatigue tests are generally carried out with a two-minute dwell time in order to make observations over a period consistent with take-off life-limited rotors. However the in-service use of climb life-limited engines generates markedly longer stress dwell times. The cold dwell fatigue tests carried out by P&W after the accident revealed that an increase in the dwell time significantly reduces the part's life. The stress/dwell time pairing significantly affects the cold dwell fatigue life. Given the stress level in the slot bottom, it is probable that the dwell time of this stress in service was a dominant factor in the initiation of the crack at this site. Unlike critical zones where the stresses are higher but with very small volume, the slot bottom, with a lesser stress, has a greater volume loaded. The volumetric character of the stress level associated with the presence of an intense macro-zone very probably contributed to a cold dwell fatigue incipient crack in this zone.

In addition, although there has been in-depth research into this phenomenon with respect to certain alloys, the mechanisms at the origin of the initiation of a cold dwell fatigue crack were still not completely understood at the time of the accident and are still not understood today. Numerous studies into cold dwell fatigue in the alloy Ti-6242 have been carried out. In particular, the FAA published in 2018[41] a study[42] which aimed to:

1) provide a fundamental understanding of the factors that promote the occurrence of dwell fatigue in Ti-6242 and other related titanium alloys,
2) develop a quantitative model to predict crack initiation based on the alloy microstructure. This model development has been supported by state-of-the-art microstructure characterization involving electron, optical and x-ray techniques. In addition, a new method for determining microstructures that are susceptible to dwell fatigue, by using non-destructive acoustic emission methods, has been demonstrated on test specimens. Its validity on full-size engine rotor components has yet to be proven.

(41) At this date, the F-HPJE hub failure mechanism was not known.

(42) FAA, The Evaluation of Cold Dwell Fatigue in Ti-6242, February 2018, Final report http://www.tc.faa.gov/its/worldpac/techrpt/tc17-57.pdf

One of the significant results of this study is the development of a model for predicting stress-strain behaviour and crack initiation in Ti-6242 parts. The authors of this study worked with GE Aviation in order to compare the models' prediction capability with the test results. According to the authors, the macroscopic models developed for Ti-6242 can be easily incorporated into commercial finite element calculation codes. This study does not cover Ti-6-4.

In 2010, a manufacturer developed its own criteria for taking into account cold dwell fatigue in Ti-6-4. This manufacturer's findings were not shared with the engine manufacturer community before the cold dwell fatigue failure was revealed by the fan hub examination. During the investigation, EASA was made aware of this manufacturer's observations and informed the BEA.

Without prejudicing a manufacturer's efforts and technological lead, a fair level of shared information, in a presentation to international work groups, would benefit all of the community. This would permit the overall safety level to be increased. It would be up to the manufacturer at the origin of the finding to define the content and detail of information to be shared.

Pratt & Whitney is working in partnership with a group of manufacturers from the civil and military aeronautic industry and from the titanium industry, on the development of a macro-zone quantification and classification model and the relationship between macro-zones and the cold dwell fatigue life. This work applies to Ti-6-4 and to Ti-6242[43] and is still on-going. A cold dwell fatigue life analytic model is being developed for these two alloys. This model predicts the influence of a macro-zone's characteristics on the fatigue life.

Fatigue models have been validated by comparison with experimental results where the specimen microstructure and macro-zones had been characterized. The difference between the predicted life and the experimental result life was less than a factor of 2.

Work must be continued in order to be able to predict the presence of macro-zones and their characteristics in an actual part with dimensions other than those of the specimen's. This scale effect must be taken into account so that the models developed using specimens can be used on an actual part. The advances made in this study were presented in the 14th world conference on titanium, at Nantes, in 2019.

[43] Data Driven Tools and Methods for Microtexture Classification and Dwell Fatigue Life Prediction in Dual Phase Titanium Alloys, Vasisht Venkatesh, Ryan Noraas, Adam Pilchak, Sesh Tamirisakandala, Kayla Calvert, Ayman Salem, Thomas Broderick, Michael Glavicic, Ian Dempster, Vikas Saraf, 2019.

2.5 Production precautions

2.5.1 Presence of macro-zones in titanium parts

The presence of macro-zones is inherent to the manufacturing process of forged titanium parts. They appear during the process to convert an ingot into a billet (see paragraph 1.18.2). They are then reduced during the part forging process by means of various successive thermomechanical treatments.

The size and severity of macro-zones depend on the size of the billet during the β recrystallization stage, and on the level and direction of the thermomechanical processing during the subsequent steps. The smaller the billet, the greater the forging strain required to obtain the geometry of the final part, reducing the risk of having large intense macro-zones. Engine developments have had a tendency to increase the bypass ratio and thus to increase the size of the fans. The size of the billets required to make the titanium hubs has also increased, increasing the risk of having large intense macro-zones if the deformation level during the forging and conversion steps is not sufficient.

The macro-zone present in the subsurface of the slot bottom of the engine No 4 fan hub, in which the crack started, was, according to the manufacturer, one order of magnitude (x10) larger and more intense than the average MTR (maximum size 1.1×10^6 μm^2 with an intensity of 6.58). The average magnitudes of the macro-zones observed by the manufacturer were situated around 2×10^5 μm^2 for the average size and their intensity was generally less than 1. No standard or practice defines the acceptability criteria of a macro-zone.

2.5.2 Detection of macro-zones in production

In production, macro-zones can only be detected by destructive methods: cutting the part and then carrying out an EBSD analysis. No non-destructive test method is available to detect macro-zones in the part, whatever the stage of the manufacturing process. The ultrasonic detection methods are constantly progressing to improve the detection threshold. However, at the current time, the macro-zone ultrasonic return is too low to distinguish it from measurement noise.

2.6 Operational aspects

2.6.1 Information available to crew when there is severe damage

When the crew detected the failure of the right outer engine and applied the associated procedure, they decelerated to the greendot speed and descended to the driftdown indication (EO MAX FL) on the FMS. They were surprised that they were unable to hold this level at a constant speed, and they were not able to estimate the altitude which the aircraft could hold. They started a step down descent to finally stabilize around 7,000 ft below the expected level. The increased drag resulting mainly from the damage to engine No 4 explains the difference in stabilization level. The Engine Fail procedure does not refer to possible degraded aerodynamic characteristics in the event of a severe failure. When the engine failure generates degraded aerodynamic performance, the crew must be aware that the driftdown indication calculated by the FMS is a ceiling. The operational documentation does not contain information which would allow a flight crew to estimate the consequences of severe aircraft or engine damage on aircraft performance (ceiling, speed and range). Producing such information in a usable format is not an easy task as the range of possible damages to be considered would be quite large.

The aeroplane was outside radar coverage during the step down descent to FL 270. No nearby traffic was reported and there were no obstacles on the planned route. The driftdown to a flight level around 7,000 ft lower than the flight level envisaged by the crew did not therefore create any particular risk. However, for any aeroplane, in the event of an engine failure, descending below the driftdown level considered during flight preparation may be critical when obstacles are present on the route or due to traffic when the aeroplane is not able to hold a flight level assigned by air traffic control. It had been a concern for the crew who were surprised that they could not hold the theoretical driftdown indication calculated by the FMS and displayed in the cockpit. The step down descent initiated by the crew was the only effective means available to determine the actual stabilization level that the aircraft could maintain at constant targeted speed.

Moreover, the final flight level was not the crew's sole concern. They also had to determine the targeted descent speed taking into account the additional drag. The theoretical CAS to be targeted was higher than the greendot speed. However without any possibility for the crew to evaluate the optimized descent speed, targeting greendot was appropriate.

The Engine Fail procedure did not specify that this level was a ceiling which only takes into consideration simple engine failures. The calculation takes the hypothesis that the engine is windmilling and does not take into account other possible situations leading to aerodynamic effects, notably increased drag from seized engine rotors or substantial engine damage.

2.6.2 CVR preservation by crew

The engine failure was not present in the CVR data as this recorder's total recording time is 2 h 04 min, in accordance with the regulations in force. The aeroplane landed around two hours after the separation of the fan. The taxiing time was extended due to the inspections required to collect the parts which had fallen from the engine onto the runway. The crew shut down the engines 2 h 37 min after the engine failure.

The analysis of the data confirmed that the CVR automatically stopped five minutes after the last engine was shut down which corresponds to the Airbus CVR end of recording logic.

The preservation of the CVR had been anticipated while en route by the captain who asked FO/1 to get ready to take the necessary steps once on the ground. The latter had looked for the associated procedure in the operational documents in order to not lose time on arrival. The position of the breaker indicated in the documents available to FO/1 was incorrect which meant that he was not able to find the breaker and pull it. Prior to this, access to the avionics bay had already been slowed down by the access door being locked with a key, and its position requiring the removal of a bench.

Despite the CVR recording time being increased to 25 h for certain newly built aeroplanes, it is probable that the preservation of the audio data in the case of a safety investigation will still principally rely on the CVR preservation procedures for the older aeroplanes not equipped with these new recorders, due to the extended operation time of aircraft.

2.6.3 Three-person crew

In order to comply with the flight time limitation and rest requirements, Air France flight crews may be augmented and include three pilots for certain long haul flights (captain and two FOs). The relief pilots can be present in the cockpit and ensure active monitoring of the flight.

When a serious incident or accident occurs during a flight, the flight crew on duty manage the flight and process the incident while the relief pilot has no defined task or role. It is however probable that he will try and help the two pilots flying in such a situation.

As the division of tasks is planned for two pilots, the role of the relief pilot will depend on the circumstances and initiative of the crew members. On the one hand, the duty pilot might disrupt the performance of operational procedures designed for two crew members. On the other hand, although they could reduce the crew's workload, the resources of the third pilot might not be used in the absence of instructions or training. The help provided by FO/1 was considered beneficial by the crew flying.

Operators could anticipate the interaction of an augmented crew by providing a guide which would, for example, supplement the Decide and Execution parts of the FOR-DEC, by specifying the possible areas in which the relief pilot can intervene.

Furthermore, the relief pilot's access to the cockpit should not be implicit right after the occurrence of the incident, if not justified by any defined role.

The F-HPJE crew flying did not consider that FO/1's differed arrival in the cockpit was detrimental to the tasks carried out.

2.6.4 Method for processing onboard incidents

The decision making method, called FOR-DEC by Air France, proved to be an effective tool for processing the incident. It ensured, via a shared framework known to all of the crew, the adequate temporal management of the occurrence. All the decisions taken by the captain were first discussed and agreed. His leadership created a climate of trust conducive to the performance of the safety tasks and the reassurance of the passengers.

3 - CONCLUSIONS

The conclusions are solely based on the information which came to the knowledge of the BEA during the investigation. They are not intended to apportion blame or liability.

3.1 Findings

- The aeroplane had a valid airworthiness certificate.
- The crew held the necessary licenses and ratings to accomplish the flight.
- In cruise climb over Greenland, strong vibrations appeared with the aeroplane simultaneously yawing to the right; several failure messages concerning the engine installed in the right outer position were displayed on the ECAM "*ENG 4 STALL*" and "*ENG 4 FAIL*".
- The front part of engine No 4, including the fan hub, air inlet and the associated fairings had separated.
- Debris struck the wing, airframe and trimmable horizontal stabilizer without any significant consequences.
- The fan hub had accumulated 3,534 cycles.
- The speed started to decrease and the aeroplane to descend.
- The captain took the controls and became PF again.
- The crew started processing the failure in accordance with the FOR-DEC decision making method used by Air France.
- FO/1 who had been in the crew rest station, came into the cockpit to help the crew flying.
- The crew observed the damage to engine No 4 from photos taken by a passenger. FO/1 confirmed the actual and visible nature of the damage from the cabin.
- The crew were surprised and concerned by the need to stabilize the aeroplane at a level lower than the driftdown indication displayed on the FMS.
- The descent was made to the driftdown level FL 270, around 7,000 ft below the level calculated by the FMS. This did not lead to conflicts with other aircraft and did not lead to a significant risk due to the absence of obstacles on the route.
- The crew chose to divert to Goose Bay aerodrome in agreement with the Air France CCO.
- They made a GNSS approach to runway 26.
- The aeroplane landed without any other anomaly.
- FO/1 tried to stop the CVR on the arrival of the aeroplane but was unable to do so due to an error in the onboard aircraft documents.
- The CVR automatically stopped five minutes after the last engine was shut down in accordance with the Airbus end of recording logic.
- The Goose Bay air terminal cannot simultaneously handle the number of passengers in an A380, consequently they were authorized to leave the aeroplane in small groups before returning to the aircraft in which they stayed for around 16 h. They were then re-routed to their final destination on two aeroplanes.

Later

- ❒ A fragment of the fan hub was found in south Greenland, under the ice, 21 months after the accident.
- ❒ The fragment was analysed and revealed that the failure, which originated in a macro-zone in the subsurface of a blade slot bottom, occurred due to a cold dwell fatigue phenomenon. The crack progressed for around 1,650 cycles until the total failure of the hub.
- ❒ The predicted number of life cycles for the hub was 15,000 cycles.
- ❒ This failure was neither anticipated nor prevented by an operational or maintenance action.
- ❒ The hub production inspections did not reveal any anomaly.
- ❒ The macro-zone where the crack was initiated was of an order of magnitude larger and more intense than the average MTR observed by the manufacturer, both in other zones of the engine No 4 hub and in hubs from the same billet.
- ❒ The cold dwell fatigue phenomenon brought to light by this accident was taken into account neither in the engine certification nor in the engine design.
- ❒ At the time of the part design and engine certification, it was accepted by the scientific community, the industry and the certification authorities that Ti-6-4 was not sensitive to the cold dwell fatigue phenomenon.

3.2 Contributing factors

Scenario

The crew took off from Paris-Charles-de-Gaulle for a flight bound for Los Angeles, onboard an A380 equipped with EA GP7270 engines. The aeroplane suffered an uncontained failure of the engine installed in the right outer position (engine No 4) while in cruise climb to FL 380, overhead Greenland. No forewarning had preceded this failure. The fan separated from the engine bringing about the separation of numerous pieces of debris. The loss of these parts followed a random path but did not cause any substantial damage to the aeroplane. During the descent, the crew were surprised that they could not hold the driftdown level calculated by the FMS. They adopted a strategy consisting in a step down descent to finally stabilize in speed and at a flight level around 7,000 feet below the expected level, known as EO MAX FL. The operational documents do not remind flight crews that this EO MAX FL is a ceiling that is achievable with a failed engine in that is windmilling, and that it may not be achievable in other situations. Without means to estimate the reduction in performance consecutive to the observed severe damage, the crew was not able to anticipate the final stabilization level. The diversion was continued to Goose Bay aerodrome (Canada) where the aeroplane was able to land without any other difficulty.

The engine debris fell into a deserted area in Greenland and the main elements were only found around 21 months after the accident. It was only possible to determine the failure process once a hub fragment had been found. In the meantime, inspection actions concerning the fleet in service had been carried out based on the information available. The scenario of damage occurring during a maintenance operation involving the removal of fan blade lock ring had been considered the most likely. The examination of the fan hub fragment located in Greenland found that a cold dwell fatigue phenomenon caused the development and progression of a crack in the subsurface of a blade slot bottom. Neither the manufacturer nor the certification authorities had anticipated this phenomenon in this alloy during the design of the engine.

Contributing factors

The following factors may have contributed to the failure of the fan hub on engine No 4:

- ❒ engine designer's/manufacturer's lack of knowledge of the cold dwell fatigue phenomenon in the titanium alloy, Ti-6-4;
- ❒ absence of instructions from the certification bodies about taking into account macro-zones and the cold dwell fatigue phenomenon in the critical parts of an engine, when demonstrating conformity;
- ❒ absence of non-destructive means to detect the presence of unusual macro-zones in titanium alloy parts;
- ❒ an increase in the risk of having large macro-zones with increased intensity in the Ti-6-4 due to bigger engines, and in particular, bigger fans.

4 - MEASURES TAKEN SINCE OCCURRENCE

4.1 Preservation of flight recorders

The difficulties encountered by FO/1 when he wanted to preserve the CVR revealed that the onboard operational documents did not correspond to the aeroplane's configuration. In particular, the position of the breaker indicated in the procedure was not correct.

Air France homogenized the CVR preservation procedures on all of the fleet on 4 January 2018. From now on, it is indicated that the crews must contact the maintenance personnel in order to carry out the appropriate CVR preservation actions. If the maintenance personnel cannot be contacted, the crew must pull the breaker as specified in the rest of the procedure.

The A380 technical information was updated on this occasion for the Air France fleet, taking into account each aircraft model.

4.2 Inspection of GP7270 fan hubs just after accident

The failure of F-HPJE engine No 4 occurred suddenly, without any early warning signs being either picked up by the crew or recorded. Close attention was not specifically paid to the monitoring of the engine due to the number of cycles and operating hours it had accumulated at the date of the accident. The consequences of this uncontained high-energy failure are potentially catastrophic and risk minimization is required under CS-25.901(c) and CS-25.903(d)(1).

Several Service Bulletins (SB) were published by the manufacturer after the accident requiring in-service inspections to be carried out. These inspections focused on the detection of potential damage in the fan hub regions which the manufacturer had identified as critical in terms of stress levels. The manufacturer's SBs were adopted by ADs issued by the FAA and EASA.

These inspections were decided on without knowing the failure mechanism identified later during the examination of the fan hub fragment from engine No 4, on the assumption that the origin of the crack was on the surface and on the front face of the part. In the scope of the continuing airworthiness of A380 aeroplanes equipped with GP7270s and given the information and tools available at this point, the fan hubs were to be inspected to check that there were no cracks. Some of these inspections were extended to engines equipping other A380s.

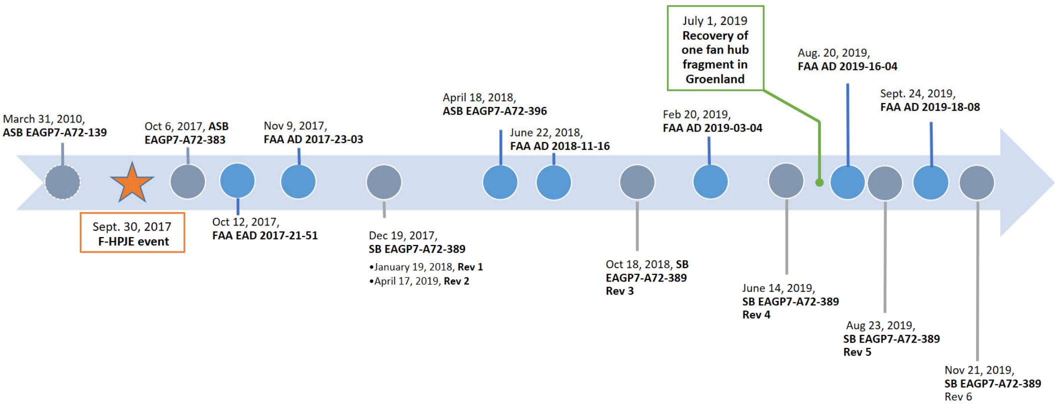


Figure 27: publication sequence of documents concerning fan hub inspection (time scale not respected)

The visual inspections required by the first SB published after the accident detected several cases of damage on the front faces of fan hubs. In particular, damage 0.30 mm (0.012 inches) deep was found in a scallop. This zone is subject to the highest stresses in service. This damage was most certainly due to an object striking the front face of the hub. The use of inappropriate tools during blade removal maintenance actions could cause similar damage.

In March 2020, Engine Alliance indicated that all the visual inspections required by SB A72-383 had been carried out. In total, 19 hubs had damage. Of these 19 hubs, two were scrapped as the damage was outside the repair limits, the others returned to service after being repaired.

The ECIs and visual inspections required by SB A72-389 led to the inspection of a total of 30 hubs with more than 3,500 cycles after its initial publication and 20 additional hubs after revision 1. After revision 2, 58 hubs with a service life of more than 3,500 cycles (out of 62) were inspected. After revision 3, two hubs were withdrawn from service, one due to the result of the ECI and the other following the visual inspection. The supplementary examinations carried out on the first hub did not find any particular damage. An indication 0.13 mm (0.005 inches) deep was identified on the second hub, without any transfer of foreign matter. A replica identified damage of 0.025 mm (0.001 inches) only.

The part was repaired and returned to service. Following the issue of revision 5, nearly all the initial inspections had been carried out and the periodic inspections were in progress. As of 10 March 2020, no crack had been detected.

4.3 Design of a new fan blade lock ring

During the investigation and before the examinations of the fan hub fragments found in Greenland, the failure simulations combined with the in-service inspection results gave rise to a scenario in which a maintenance operation to remove the fan blade lock ring could be at the origin of the damage observed on the front face of the fan hub, leading to the hub failure. The ring removal operation was described as difficult by the operators because of its stiffness. The marks found during the hub in-service inspections were attributed to the use of inappropriate tools.

The engine manufacturer has designed a new blade lock ring. The new ring is more elastic which facilitates the maintenance operations. Its deployment in the fleet started on 25 November 2019.

4.4 Inspections since examination of engine No 4 fan hub

Once the failure mechanism had been determined, EA published a new Service Bulletin (revision 6 of SB A72-389) on 21 November 2019. It required, in particular, the performance of periodic ultrasonic inspections in addition to the ECIs.

These frequent inspections (every 330 cycles) require the removal and installation of the fan blades. The increased risk of damaging the hub front face, in particular, during the removal and installation of the blade lock ring, has been reduced by the modification to the design of this ring (see paragraph 4.3).

5 - SAFETY RECOMMENDATIONS

Note: in accordance with the provisions of Article 17.3 of Regulation No 996/2010 of the European Parliament and of the Council of 20 October 2010 on the investigation and prevention of accidents and incidents in civil aviation, a safety recommendation in no case creates a presumption of fault or liability in an accident, serious incident or incident. The recipients of safety recommendations report to the authority in charge of safety investigations that have issued them, on the measures taken or being studied for their implementation, as provided for in Article 18 of the aforementioned regulation.

5.1 Titanium rotor-grade critical parts

Up until the failure of the (GP7270) engine No 4 fan hub, the titanium alloy, Ti-6-4 was not considered sensitive to the cold dwell fatigue phenomenon. Certain alloys such as IMI 685 or Ti-6242 had already shown predispositions to this phenomenon in the 1970s, whereas Ti-6-4 had accumulated significant in-service experience without the occurrence of any incident identified as being linked to this phenomenon.

Sizing

The investigation was able to show that the maximum stress level observed in the fracture zone of the F-HPJE fan hub (slot bottom) was less than 80% of the material's yield strength. The investigation also brought to light that the failure of the fan hub occurred after a number of cycles four times less than the hub's minimum life. The methods for estimating the pure fatigue life developed by the engine manufacturer and accepted by the FAA, forecast an incipient crack at twenty times the number of cycles of engine No 4, without taking into account the cold dwell fatigue. It was accepted that cold dwell fatigue was not significant at these stress levels.

However, the volume of the test specimens for cold dwell fatigue along with the dwell times applied in tests are not sufficiently representative of an actual part to activate large macro-zones. In fact, to reduce test times, the specimens are subject to shorter dwell times and greater stress compared to actual parts. It is not known what effect these different dwell times and stress levels have on the part's life. The actual in-service stresses and dwell time seen by the part are significantly different. Lastly, the initiation of a cold dwell fatigue crack generally occurs in a macro-zone. The probability of having an intense macro-zone in a test sample is by nature less than in a larger part. The service life debits obtained by dwell effect during tests on specimens are therefore, at the current time, difficult to transpose to in-service parts.

A lack of knowledge of both the activation envelope of the cold dwell fatigue phenomenon on Ti-6-4 and the conditions conducive to the appearance of intense macro-zones meant that a cold dwell fatigue crack was initiated at a stress level lower than that accepted up until now by only taking into consideration pure fatigue, and at a significantly lower number of cycles.

Manufacturing processes

The investigation found that a crack started and then progressed in the subsurface of a slot bottom, in a macro-zone quantified as being one order of magnitude (x10) larger and more intense than the average MTR observed by the manufacturer. Its unusually large size and its orientation, perpendicular to the hoop loads, probably contributed to the initiation of a crack although the stress levels were below 80% of the yield strength.

Cold dwell fatigue cracks are initiated in macro-zones, the presence of which is inherent to the manufacturing process of forged titanium parts. The macro-zones generally appear during the process to convert an ingot into a billet and are then reduced during the subsequent forging process, by means of various successive thermomechanical treatments.

The risk of macro-zones appearing increases with the size of the billets. For small billets, the considerable plastic deformation (strain hardening) during the conversion and forging phases reduces the size and intensity of the macro-zones. Large engines with a high bypass ratio require larger diameter fan hubs to improve effectiveness; these hubs require larger billets. The parts forged from large billets may not benefit from the same deformation levels as those parts which come from smaller billets. This may contribute to the risk of macro-zones of a large size and intensity being present.

Production check

At the present time, it is not possible to detect in a reliable way, the presence of macro-zones using non-destructive methods, whatever the stage of the manufacturing process. The EBSD technique characterizes the grain crystallographic orientation and thus reveals a macro-zone, but this is a destructive examination. The suspected zone has to be isolated, removed and prepared by polishing before the examination.

Methods for predicting the presence of macro-zones in finished parts by digital simulation are starting to emerge but are not yet reliable enough. It is currently possible to predict macro-zones in a test sample but transposing this prediction to an actual part is still in progress.

Ultrasonic measurements are carried out during the part manufacturing process in order to principally detect α based type anomalies or process induced cracks. To date, the ultrasonic inspection method does not detect macro-zones.

Thus, today, macro-zones may be naturally present in forged critical parts made of Ti-6-4, and are not covered by rejection criteria as no reliable non-destructive detection method exists, and because the current manufacturing processes do not reliably control the risk of them appearing.

The tendency to increase the size of engine fans to reduce engine fuel consumption may lead engine designers to try and substantiate higher acceptable stress levels, to limit the weight of these engines. This may lead to an increase in the risk of a cold dwell fatigue incipient crack in a macro-zone. The size criteria during the design phase, for forged critical parts made of Ti-6-4 should thus be adapted to improve the control of the cold dwell fatigue phenomenon, taking into account the risk of macro-zones appearing in production, given that these macro-zones may contribute to this phenomenon, and the limits of the macro-zone detection capabilities.

In-service monitoring

The presence of an intense macro-zone in a titanium part, not detected during production, may lead to the initiation of a crack in service. The current non-destructive inspection methods detect subsurface cracks or voids.

The initiation of a cold dwell fatigue crack can only be predicted by taking into consideration both the characteristics of the macro-zone (size, position and orientation, intensity) and local loading (stress level, dwell time, temperature). A crack may start in a zone with low stress due to the presence of an intense macro-zone or due to the length of dwell time.

The continuing airworthiness of critical parts made of the titanium alloy, Ti-6-4, which undergo a manufacturing process likely to lead to the presence of intense macro-zones and for which the risk of failure due to a cold dwell fatigue phenomenon has not been sufficiently taken into account during design, may require the implementation of appropriate means to detect in-service cracks before the failure of the part.

Consequently, the BEA recommends that:

- **EASA and the FAA ensure that the design and sizing criteria and methods along with the manufacturing processes and in-production checks of engine rotor-grade critical parts made of α/β titanium alloy, and in particular the titanium alloy Ti-6-4, are such that the risk of failure of these parts due to the cold dwell fatigue phenomenon is controlled.**
  EASA: [Recommendation FRAN 2020-006]
  FAA: [Recommendation FRAN 2020-007]

- **EASA and the FAA carry out a review of engine rotor-grade critical parts made of α/β titanium alloy, and in particular the titanium alloy Ti-6-4, which undergo a manufacturing process likely to lead to the presence of intense macro-zones and for which the risk of failure due to a cold dwell fatigue phenomenon has not been sufficiently taken into account during the certification. EASA and the FAA will subsequently make sure, where appropriate, that an adapted in-service inspection programme is implemented to detect possible incipient cracks which might lead to the failure of the part.**
  EASA: [Recommendation FRAN 2020-008]
  FAA: [Recommendation FRAN 2020-009]

6.1 Appendix 1 FDR parameters

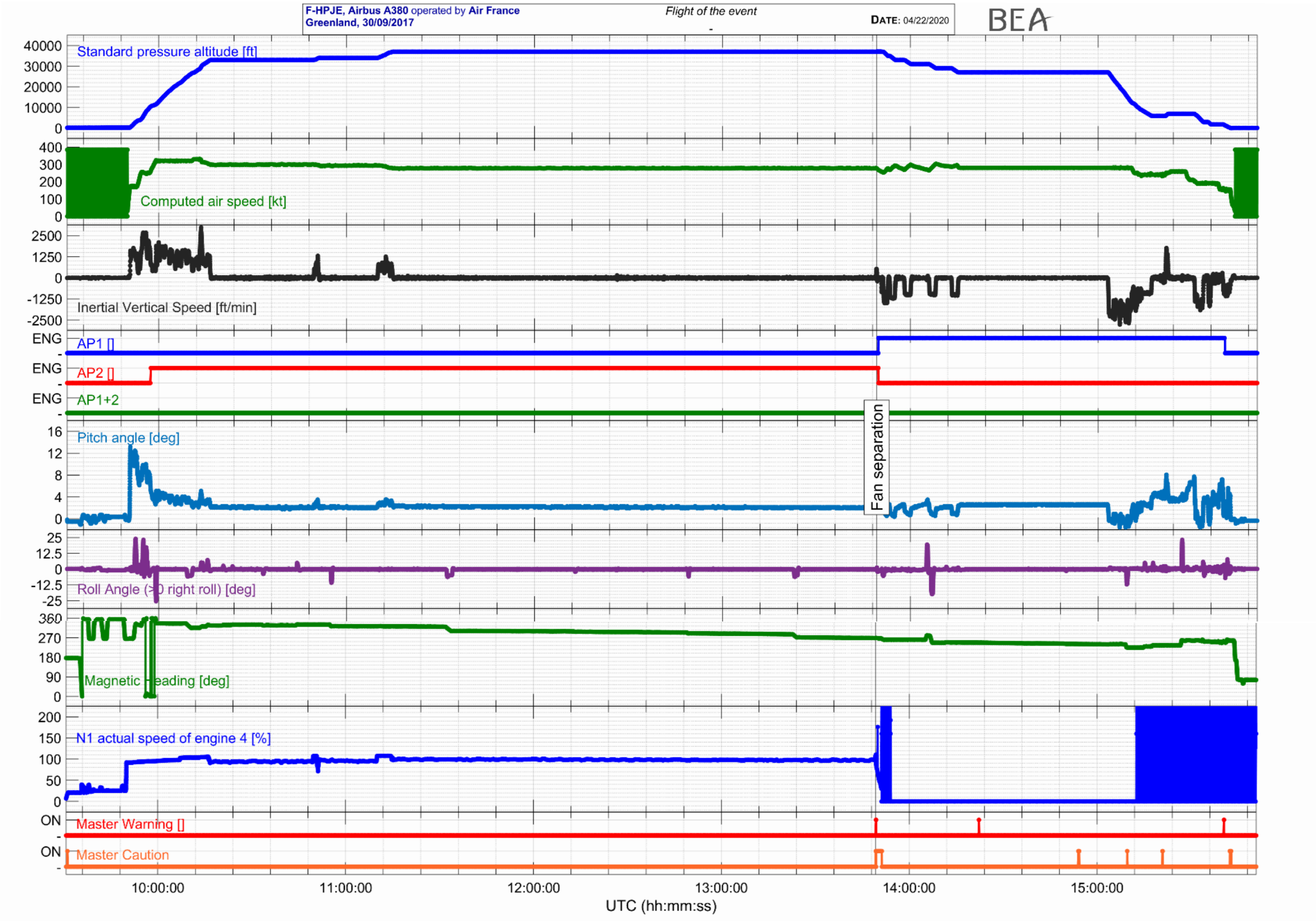


Figure 28: Occurrence flight

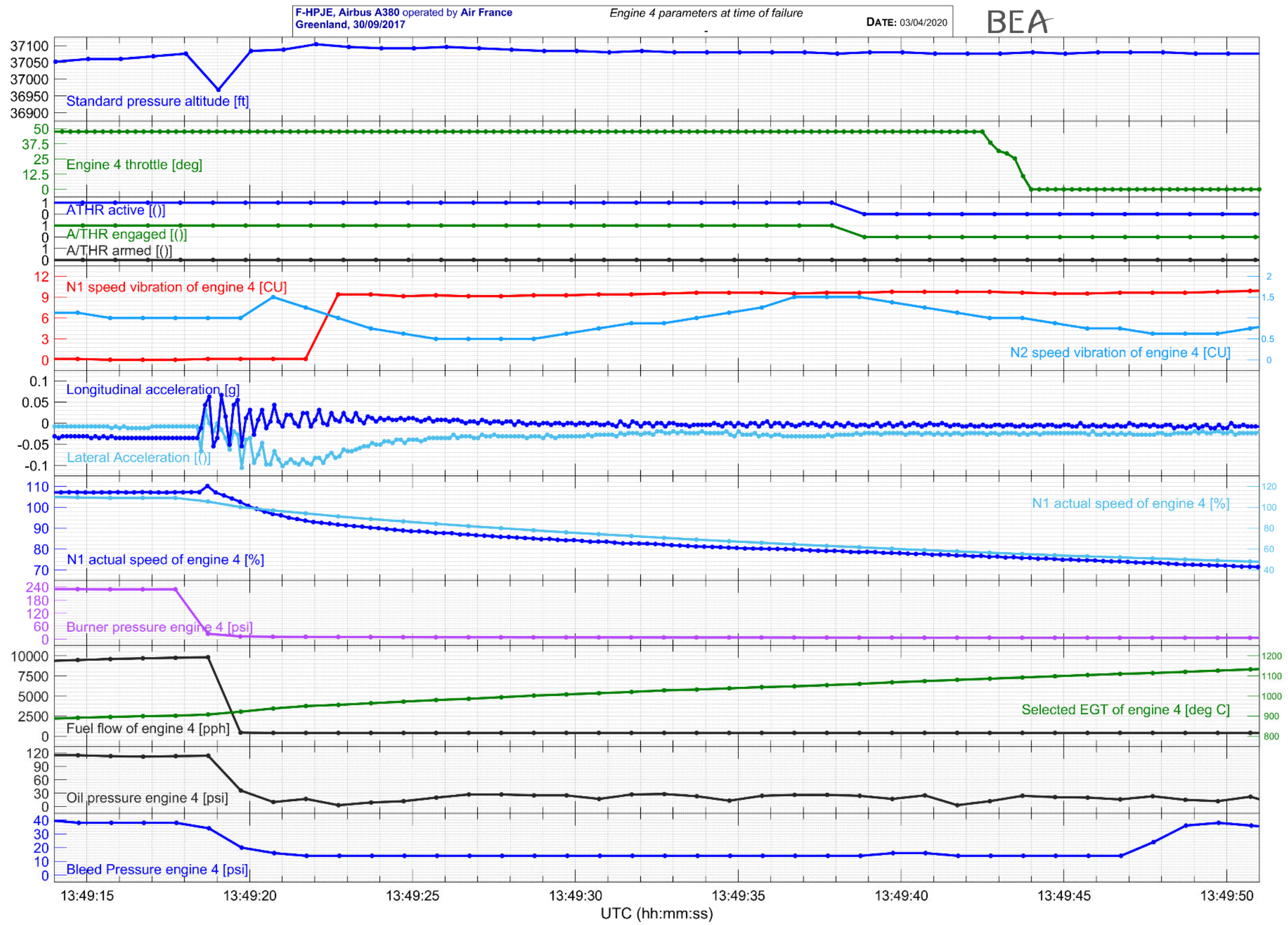


Figure 29: Engine No 4 parameters at time of failure

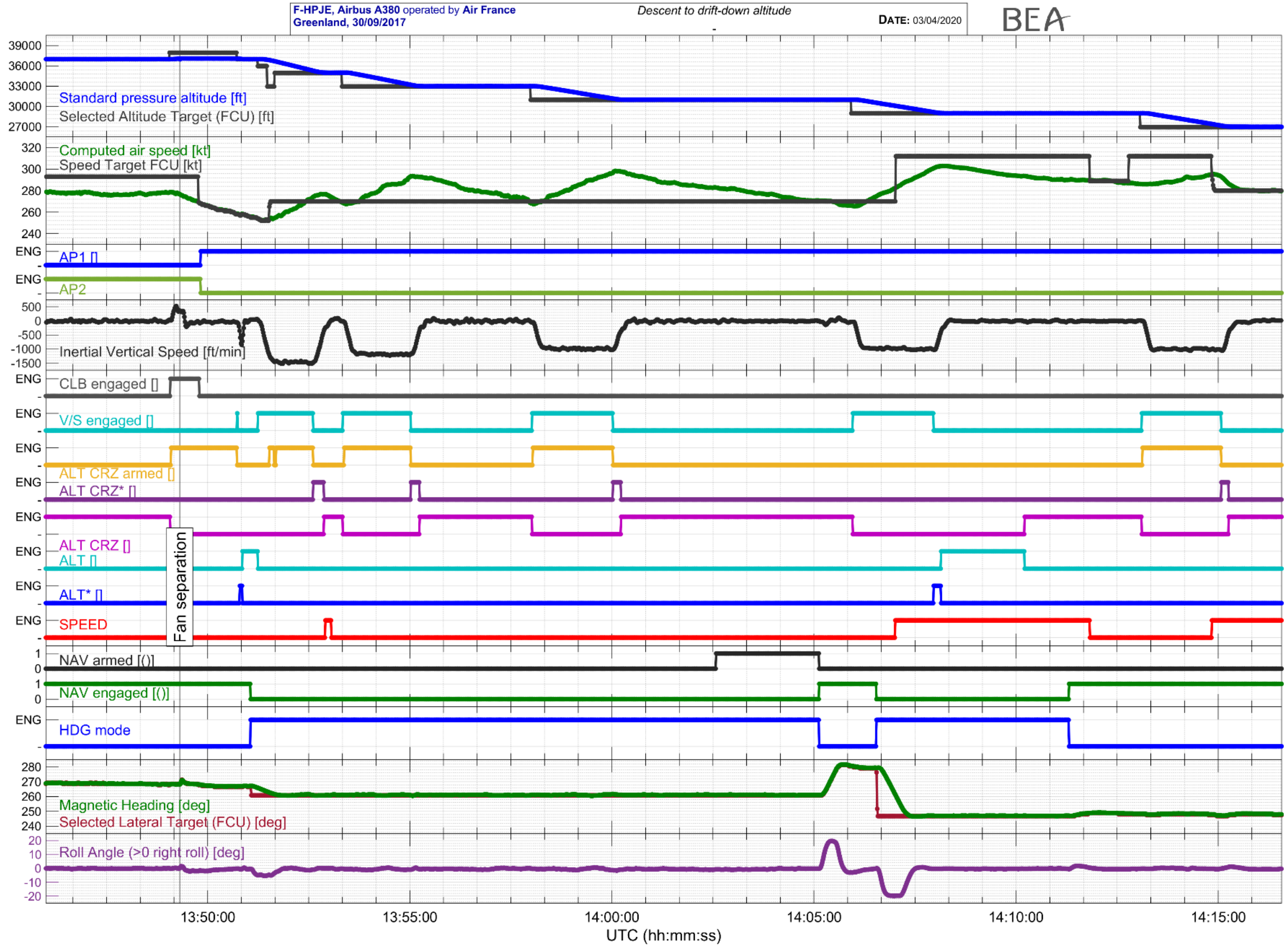


Figure 30: Descent to FL 270

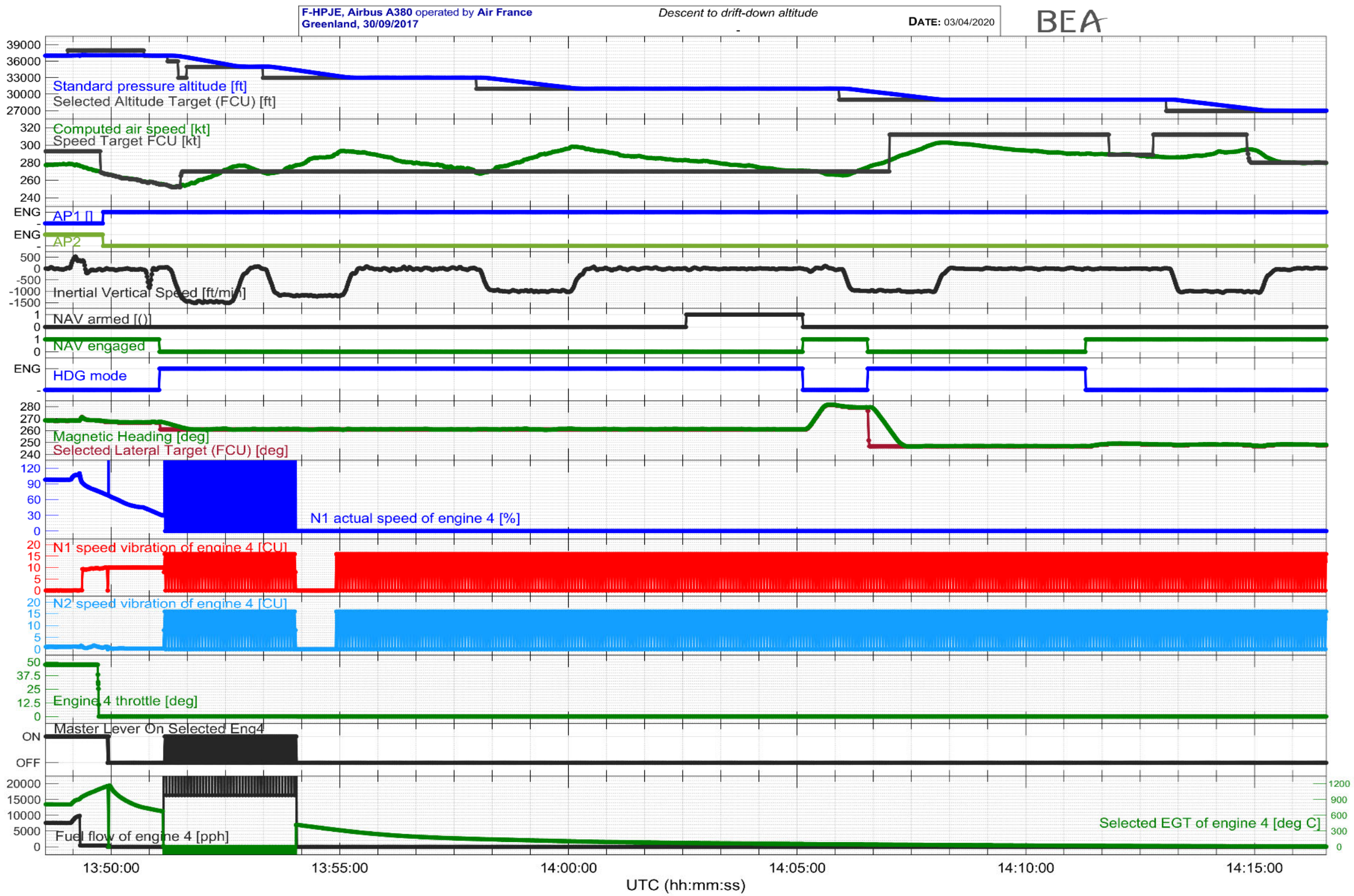


Figure 31: Engine 4 parameters during descent to FL 270

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7.2 BEA reports

TECHNICAL REPORT - Accident to the Airbus A380 registered F-HPJE and operated by Air France on 30 September 2017 en route over Greenland, October 2017 - June 2018, Search phases I and II

TECHNICAL REPORT- Accident to the Airbus A380 registered F-HPJE and operated by Air France on 30 September 2017 en route over Greenland, July 2018 - July 2019, Search phase III

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