

AI2011-5

**AIRCRAFT SERIOUS INCIDENT
INVESTIGATION REPORT**

NIPPON CARGO AIRLINES CO., LTD.

J A 0 1 K Z

August 26, 2011

Japan Transport Safety Board

The objective of the investigation conducted by the Japan Transport Safety Board in accordance with the Act for Establishment of the Japan Transport Safety Board (and with Annex 13 to the Convention on International Civil Aviation) is to prevent future accidents and incidents. It is not the purpose of the investigation to apportion blame or liability.

Norihiro Goto
Chairman,
Japan Transport Safety Board

Note:

This report is a translation of the Japanese original investigation report. The text in Japanese shall prevail in the interpretation of the report.

AIRCRAFT SERIOUS INCIDENT INVESTIGATION REPORT

**NIPPON CARGO AIRLINES CO., LTD.
BOEING 747-400F, JA01KZ
ABOUT 140 FEET OVER RUNWAY A OF
NARITA INTERNATIONAL AIRPORT, JAPAN
AT ABOUT 21:54 JST, JUNE 11, 2010**

August 5, 2011

Adopted by the Japan Transport Safety Board

Chairman	Norihiro Goto
Member	Shinsuke Endoh
Member	Toshiyuki Ishikawa
Member	Yuki Shuto
Member	Toshiaki Shinagawa

The following abbreviations are used in this report:

BSI	: Borescope Inspection
CVR	: Cockpit Voice Recorder
DFDR	: Digital Flight Data Recorder
DSR142	: Directionally Solidified René™ 142
EGT	: Exhaust Gas Temperature
EICAS	: Engine Indicating and Crew Alerting System
FF	: Fuel Flow
HPC	: High Pressure Compressor
HPT	: High Pressure Turbine
HPTR	: High Pressure Turbine Rotor
LPC	: Low Pressure Compressor
LPT	: Low Pressure Turbine
N1	: Low Pressure Compressor Speed (Fan Speed)
N2	: High Pressure Compressor Speed
NGV	: Nozzle Guide Vane
PF	: Pilot Flying
PM	: Pilot Monitoring
RN5	: René™ N5
S/B	: Service Bulletin
TBC	: Thermal Barrier Coating
TRF	: Turbine Rear Frame
V1	: Takeoff decision speed
V2	: Takeoff safety speed
VR	: Rotation speed
VSV	: Variable Stator Vane

René™ denotes a nickel-based heat-resistant alloy developed by General Electric.

1. PROCESS AND PROGRESS OF THE INVESTIGATION

1.1 Summary of the Serious Incident

The occurrence covered by this report falls under the category of “Damage of engine (limited to a major damage occurred inside the engine)” as stipulated in Clause 6, Article 166-4 of the Civil Aeronautics Regulations of Japan, and is classified as an aircraft serious incident.

On June 11 (Friday), 2010, a Boeing 747-400F, registered JA01KZ, operated by Nippon Cargo Airlines Co., Ltd., took off from Narita International Airport as a scheduled flight 166 (a cargo flight) of the company for Anchorage International Airport, the United States of America. At about 21:54 Japan Standard Time (JST: UTC+ 9hr, unless otherwise stated all times are indicated in JST on a 24-hour clock) just after the takeoff, an abnormal noise was heard from one of its engines and an instrument indicated a No.1 engine failure. Therefore, the flight crew shut down the No.1 engine after climbing to an altitude of 7,000 feet. After jettisoning its fuel, the aircraft turned back and landed at Narita International Airport at 23:08.

There were three persons on board, consisting of the Captain, one crewmember and an employee of the company, but no one was injured.

1.2 Outline of the Serious Incident Investigation

1.2.1 Investigation Organization

On June 14, 2010, the Japan Transport Safety Board designated an investigator-in-charge and another investigator to investigate this serious incident.

1.2.2 Representatives from Foreign Authorities

An accredited representative and an adviser of the United States of America, as the State of Design and Manufacture of the aircraft involved in this serious incident, participated in the investigation.

1.2.3 Implementation of the Investigation

June 15, 2010	Aircraft and engine examinations and flight crew interviews
June 16, 2010	Flight crew interviews
June 28 to 30, 2010	Teardown inspection of the engine at a facility designated by the engine manufacturer in Taiwan
July to September, 2010	A detailed examination of stage 1 high pressure turbine blades at a laboratory of the engine manufacturer in the United States of America

1.2.4 Provision of Factual Information to the Civil Aviation Bureau, the Ministry of Land, Infrastructure, Transport and Tourism

On July 9, 2010, factual information about the condition of damage to the inside of the engine was provided to the Civil Aviation Bureau of the Ministry of Land, Infrastructure, Transport and Tourism.

1.2.5 Comment from Parties Relevant to the Cause of the Serious Incident

Comments were invited from parties relevant to the cause of the serious incident.

1.2.6 Comments from the Participating State

Comments on the draft report were invited from the participating State.

2. FACTUAL INFORMATION

2.1 History of the Flight

At 21:54 on June 11, 2010, a Boeing 747-400F, registered JA01KZ (hereinafter referred to as “the Aircraft”), operated by Nippon Cargo Airlines Co., Ltd. (hereinafter referred to as “the Company”), took off from runway 16R (hereinafter referred to as “16R”) of Narita International Airport (hereinafter referred to as “the Airport”) as a scheduled flight 166 of the Company for Anchorage International Airport, the United States of America.

The outline of the flight plan was as follows:

Flight rules:	Instrument flight rules (IFR)
Departure aerodrome:	Narita International Airport
Estimated off-block time:	21:40
Cruising speed:	495kt
Cruising altitude:	Flight Level (FL) 310
Route:	GIRAF (reporting point) – Y808 (RNAV route) – (The rest is omitted)
Destination aerodrome:	Anchorage International Airport
Estimated flight time:	6 h and 34 min

At the time of the serious incident, in the cockpit of the Aircraft, the Pilot in Command (PIC) was in the left seat as the PF (pilot flying: pilot mainly in charge of flying) and the First Officer (FO) was in the right seat as the PM (pilot monitoring: pilot mainly in charge of duties other than flying).

The history of the flight up to the time of the serious incident is summarized below, based on the records of the digital flight data recorder (hereinafter referred to as “the DFDR”), ATC communication records and the statements of the flight crewmembers.

2.1.1 History of the Flight based on the DFDR Records and ATC Communication Records

21:52:28	The Narita Airport Traffic Control Tower (hereinafter referred to as “the Tower”) issued a take-off clearance for the Aircraft.
21:52:42	The throttle levers for all the four engines were advanced while the Aircraft was on the runway The low pressure compressor (LPC) rotation speed (hereinafter referred to as “N1”), the high pressure compressor (HPC) rotation speed (hereinafter referred to as “N2”) and the fuel flow (hereinafter referred to as “FF”) gradually increased for all the engines.
21:53:41	The Aircraft lifted off at an airspeed of about 190 knots.
21:53:48	N1 and N2 for No.1 engine declined fast at an altitude of about 140 feet and at an airspeed of about 200 knots. The exhaust gas temperature (hereinafter referred to as “EGT”) temporarily went up, but it gradually went down later along with a decline in FF.
21:53:50	The vibration of No.1 engine increased to a maximum value of 5 units. The heading of the Aircraft veered to the left by about 6 degrees, but the

veering was corrected later.

21:53:57 The Tower instructed the Aircraft to contact the Tokyo Departure Control (hereinafter referred to as “the Departure”).

21:53:58 The autopilot was engaged.

21:54:03 FO reported the occurrence of an engine failure to the Tower.

21:54:40 FO reported the occurrence of an engine failure to the Departure. FO also requested for a clearance for climbing to an altitude of 7,000 feet and reported to the Departure that it takes about an hour to jettison fuel.

Past 21:59 The Aircraft reached an altitude of about 7,000 feet and maintained a level flight.

22:00:36 The throttle lever for No.1 engine was retarded to the idle position.

22:00:45 The fuel control switch for No.1 engine was shifted to the Cut-off position.

23:08 The Aircraft landed at 16R of the Airport.

There was no record of a master warning activated during the flight.

2.1.2 Statements of Crewmembers

(1) PIC

I was sitting in the left seat and FO in the right seat. Another person was on aboard the Aircraft, but he was not in the cockpit. The takeoff weight of the Aircraft was heavier than usual for this flight. The Aircraft left an apron at 21:39, but I felt nothing unusual with the engines.

While the Aircraft was in takeoff rolling, EICAS*¹ displayed no message of failure. I performed a nose-up operation and when the Aircraft lifted off and started climbing, I heard a big abnormal noise and a message was indicated on EICAS that failure has occurred with No.1 engine.

I continued climbing while leaving the situation as it was and then, I shut down No.1 engine at an altitude of about 7,000 feet. I decided to return to Narita Airport and flew toward the reporting point of VENUS under the controller’s instructions to jettison fuel. While receiving radar vector at an altitude of 7,000 feet, the Aircraft jettisoned about 150,000 pounds of fuel in about 45 minutes and after that, the Aircraft landed on 16R of the Airport at 23:08. The Aircraft has four engines and despite the failure which occurred with one engine, the Aircraft was fully controllable. Therefore, I did not declare an emergency.

(2) FO

When we started the engines, these were all normal. We found no abnormality when we performed the checklist.

After the start of the takeoff rolling, when I called a V1 (Takeoff decision speed) at around a speed of 155 knots, the indications of the engine-related instruments were normal. The nose of the Aircraft was pulled up at a VR (Rotation speed) of 170 knots, and I called a V2 (Takeoff safety speed) at a speed of 180 knots. After confirming the positive climb rate, I raised the landing gear lever under PIC’s instructions. Just after that, I heard a boom sound and the N1 and EGT readings for No.1 engine clearly declined.

I think the engine failure occurred at a pressure altitude of about 300 feet over the runway, but I cannot remember the altitude exactly. PIC continued to climb straight and

*1 EICAS means an integrated system designed to display the operational conditions of the engines and part of other systems and notify the pilot of the occurrence of abnormalities by visual and auditory means

engaged the autopilot. Because I saw a message of No.1 engine failure on the EICAS display, I called “Engine Failure No.1.”

When we received instructions from the Tower to contact the Departure, I reported to the Tower of the occurrence of the engine failure. Later, I changed the radio frequency and also reported to the Departure of the occurrence of the engine failure. I requested the Departure to climb to an altitude of 7,000 feet under PIC’s instructions.

We flew to VENUS to jettison fuel and performed a checklist for “Engine Failure and Shutdown”, “After Takeoff” and “Fuel Jettison”. While we came close to VENUS, we started jettisoning the fuel and requested the controller for radar vector.

After the jettisoning of fuel, we returned to the Airport and landed on 16R normally with the remaining three engines.

This serious incident occurred at about 21:54 at an altitude of about 140 feet over runway A of the Airport (around Latitude 35°44’ 54” N, Longitude 140°23’12” E).
(See Figure 1 Estimated Flight Route, Figure 2 DFDR Records, Photo 1 Serious Incident Aircraft)

2.2 Injuries to Persons

No one was injured in the serious incident.

2.3 Damage to the Aircraft

2.3.1 Extent of Damage

The inside of No.1 engine was substantially damaged, but there was no other damage to the Aircraft.

2.3.2 Damaged Condition of No.1 Engine

A visual inspection and a borescope inspection (hereinafter referred to as “BSI”) revealed that damaged condition of No.1 engine was as follows:

- (1) There were no signs of foreign object debris (FOD) ingested from outside on the fan blades.
- (2) No abnormality was detected in the compressor and the combustion chamber.
- (3) The trailing edge of the nozzle guide vane (hereinafter referred to as “NGV”), located at the way out of the combustion chamber, was found partially damaged.
- (4) Four of the 80 stage 1 high pressure turbine (HPT) blades were found broken from the root, and all other blades had also been damaged.
- (5) The 74 stage 2 HPT blades were all damaged.
- (6) The low pressure rotor could not be rotated. Several low pressure turbine (LPT) blades had been broken in a section which can be observed near the BSI port.
- (7) Damage was found in the stage 5 LPT vane and the stage 5 LPT blades behind it, which can be seen from the back of the engine.
- (8) Numerous fine pieces of metal had been left in the turbine rear frame (hereinafter referred to as “TRF”).

2.4 Personnel Information

(1) PIC Male, Age 59
Airline Transport Pilot Certificate (Airplane)

July 28, 1987

Type rating for B747-400	June 23, 2003
Class 1 Aviation Medical Certificate	
Validity	July 13, 2010
Total flight time	16,274 h 36 min
Flight time in the last 30 days	11 h 46 min
Total flight time on the type of aircraft	3,079 h 00 min
Flight time in the last 30 days on the type of aircraft	11 h 46 min
(2) FO	Male, Age 30
Airline Transport Pilot Certificate (Airplane)	May 9, 2006
Type rating for B747-400	March 27, 2008
Class 1 Aviation Medical Certificate	
Validity	October 15, 2010
Total flight time	5,043 h 53 min
Flight time in the last 30 days	48 h 02 min
Total flight time on the type of aircraft	835 h 30 min
Flight time in the last 30 days on the type of aircraft	48 h 02 min

2.5 Aircraft Information

2.5.1 Aircraft

Type	Boeing 747-400F
Serial number	34016
Date of manufacture	May 16, 2005
Certificate of airworthiness	DAI-TOU-21-145
Validity	June 30, 2010
Category of airworthiness	Airplane, Transport T
Total flight time	20,730 h 59 min
Flight time since last periodical check (C03 check on April 25, 2009)	4,807 h 50 min

(See Figure 3 Three Angle View of Boeing 747-400F)

2.5.2 Engine

Model	General Electric	CF6-80C2B1F
	No.1 Engine	No.2 Engine
Serial Number	706704	706768
Date of manufacture	April 26, 2005	December 19, 2006
Total time	17,006 h 20 min	11,312 h 22 min
Total cycles	3,126 cycles	2,271 cycles
	No.3 Engine	No.4 Engine
Serial Number	706810	706887
Date of manufacture	January 20, 2007	October 30, 2007
Total time	10,152 h 44 min	6,672 h 08 min
Total cycles	2,072 cycles	1,373 cycles

(See Figure 4 CF6-80C2 Engine, Photo 2 Dismounted No.1 Engine)

2.5.3 History of No.1 Engine

For the combustion chamber and the stage 1 and stage 2 HPT blades, a BSI must be performed at an interval of 400 cycles or 2,000 flight hours, whichever comes first. This will be described later in 2.13.1.

A summary of the engine's history is as follows:

(1) On May 4, 2005,

The engine was installed on the Aircraft in position No. 4 as new after manufacture.

(2) As of February 9, 2009,

Total time 15,138 h 21 min, Total cycle 2,731 cycles

The engine was from the Aircraft to adjust its time in service and it was stored as a spare engine. By that time, BSIs, including inspections of the stage 1 and stage 2 HPT blades, had been regularly performed, and no damage was found.

(3) Between July and September of 2009,

A malfunction was found with the variable stator vane (VSV) in the compressor section of the engine which had been stored. Repair was made at a facility designated by the engine manufacturer, but at that time, a teardown of the turbine blades was not performed.

(4) On January 4, 2010,

The engine was installed to the Aircraft position No.1.

(5) As of January 11, 2010,

Total time 15,198 h 59 min, Total cycle 2,743 cycles

BSIs, including inspections of the stage 1 and stage 2 HPT blades, had been performed, and no damage was found.

(6) As of May 21, 2010,

Total time 16,752 h 18 min, Total cycle 3,070 cycles

BSIs, including inspections of the stage 1 and stage 2 HPT blades, had been performed, and no damage was found.

2.5.4 Weight and Balance

When the serious incident occurred, the Aircraft's weight was estimated to have been 802,880 pounds and the center of gravity was estimated to have been 21.9% mean aerodynamic chord (MAC), both of which are estimated to have been within the allowable range (maximum takeoff weight of 875,000 pounds, and 12.8 to 31.1% MAC corresponding to the weight at the time of the serious incident).

2.6 Meteorological Information

Aeronautical weather observations for the Airport around the time of the serious incident were as follows:

21:30	Wind direction	160°	Wind velocity	4 kt	Visibility	15 km	
	Cloud Amount	FEW* ²	Type	Stratocumulus	Cloud base	3,000 ft	
		Amount	BKN* ³	Type	Unknown	Cloud base	Unknown
	Temperature	18 °C	Dew point	16 °C			
	Altimeter setting (QNH)		30.00 inHg				

*² FEW (few) means a situation in which the visible proportion of areas covered with clouds to the entire skies is 1/8 to 2/8.

*³ BKN (broken) means a situation in which the visible proportion of areas covered with clouds to the entire skies is 5/8 to 7/8.

22:00	Wind direction	150°	Wind velocity	3 kt	Visibility	15 km	
	Cloud	Amount	FEW	Type	Stratocumulus	Cloud base	3,000 ft
		Amount	BKN	Type	Unknown	Cloud base	Unknown
	Temperature	18 °C	Dew point	16 °C			
	Altimeter setting (QNH)		30.00 inHg				

2.7 Information on DFDR and Cockpit Voice Recorder

The Aircraft was equipped with a DFDR (part number: 980-4700-042) made by Honeywell of United States of America and a cockpit voice recorder (hereinafter referred to as “the CVR”, part number: 2100-1020-00) made by L-3 Communications of United States of America. The DFDR retained data at the time of the serious incident, but CVR did not retain data at the time of the serious incident because the data was overwritten. The time was determined by correlating the DFDR recorded VHF transmission keying signals with the NTT (Nippon Telegraph and Telephone Corporation) speaking clock recorded on the ATC communication records.

2.8 Information on the Serious Incident Site

The Airport has an elevation of 135 feet and has two runways—runway A (16R/34L) with a length of 4,000 meters and a width of 60 meters to the southwestern side and runway B (16L/34R) with a length of 2,500 meters and a width of 60 meters to the northeastern side.

Runway checks were made just after the occurrence of the serious incident and after the Aircraft’s landing, but no broken pieces of the engine or the like were found.

Later, on June 18 and June 21, 2010, three metal pieces as large as about 2 centimeters were found under a takeoff course of 16R about 3 kilometers south of runway A. There was no report of damage caused with the metal pieces.

2.9 Service Bulletins Related to Stage 1 HPT Blades

Following are excerpts from service bulletins (hereinafter referred to as “S/B”) issued by the engine manufacturer regarding the stage 1 HPT blades (part number: 1538M90P12) which had been installed in the No.1 engine of the Aircraft:

- (1) S/B 72-721, issued on December 21, 1993

Compliance level: Do at customer convenience.

The S/B concerned the release of stage 1 HPT blades (part number: 1538M90P12) made from a DSR 142 material with Thermal Barrier Coating (TBC) applied on the outside surface of the blade airfoil.

- (2) S/B 72-1090, issued on March 19, 2003

Compliance level: GE recommends you do at customer option.

The S/B concerned the release of new stage 1 HPT blades (part number: 1639M70P11) made from a single crystal RN5 material. It said the supply of part numbers: 1538M90P11 and 1538M90P12 will be continued.

- (3) S/B 72-1269, issued on August 23, 2007

Compliance level: GE recommends that you do this Service Bulletin when the stage 1 HPTR blades are routed for repair

The S/B concerned added protective coating against type II hot corrosion (to be described later in 2.13.3) on the shank area (See Photo 8) below the platform of the stage 1 HPT

blade (part number: 1538M90P12).

- (4) S/B 72-1288, issued on May 8, 2008

Compliance level: GE recommends that you do this Service Bulletin at customer's convenience.

The S/B concerned the release of new stage 1 HPT blades (part numbers: 1538M90P18 and 1538M90P19) made of DSR142 material with an improved shape for the tip of the blades and vapor chromide coating applied to the shank area below the platform.

- (5) S/B 72-1296, issued on May 8, 2008

Compliance level: GE recommends that you do this Service Bulletin at customer's convenience.

The S/B concerned the release of new stage 1 HPT blades (part number: 1639M70P17) as a single crystal RN5 material with an improved shape for the tip of the blades and vapor chromide coating applied to the shank area below the platform.

2.10 Engine Maintenance Activities

According to the Company, the engine maintenance activities were as follows:

The Company has concluded a comprehensive maintenance contract with the engine manufacturer. This includes the collection and analysis of engine-related data. No clear shift in the trend which may indicate abnormalities in the data about the engine was shown since January 2010 when it was installed with the Aircraft as its No.1 engine.

The Company has regularly performed BSIs on the combustion chamber and the HPT blades in accordance with the maintenance manual, but no damage was found with the engine.

The Company was aware of S/B 72-1269 which recommended adding a corrosion protective coating on the shank area of the stage 1 HPT blades. In accordance with its compliance level, the Company planned to perform the S/B when the next overhaul will be made for the engine.

The Company received the Aircraft in June 2005. The Aircraft has operated mainly among cities in Asia, Europe and North America for about six years. There was no inclination to a specific area in its operations

2.11 Engine Data Left on DFDR

According to the records retained on the DFDR about the preceding seven takeoffs before the occurrence of this serious incident, there was no clear difference from engine to engine in the N1, the N2, the FF and the engine vibration value until just before this serious incident.

2.12 Tests and Researches for Fact-Finding

2.12.1 Teardown Inspection of Engine

For the investigation of this serious incident, a teardown inspection of No.1 engine of the Aircraft was performed at a facility designated by the engine manufacturer. The result was as follows:

- (1) There were no signs of FOD ingested into the fan.

The high pressure rotor could be rotated, but the low pressure rotor did not rotate because many metal pieces had got caught in the tips of the stage 5 LPT blades.

- (2) Contact wear was noted on the HPC rotor and part of the stator case which has 14 stages in total. There were traces of contact wear on the tips of all HPC blades and HPC vanes.

- (3) NGV had been partially broken with the trailing edge of most of the vanes damaged.

- (4) Four of the 80 stage 1 HPT blades (Blades numbered 2 to 4 and 12) had been fractured in the shank area below the platform. Other blades were damaged with the airfoils broken or lost. Three blades (numbered 5, 8 and 77), were found cracked in the shank area in the same place as the four fractured blades.

(Blade part number: 1538M90P12)

Blade	Serial Number	Condition
Blade 2	PCMDHBG9	Broken
Blade 3	PCMDGUD2	Broken
Blade 4	PCMDGUB4	Broken
Blade 5	PCMDGUD7	Cracked
Blade 8	PCMDHCT0	Cracked
Blade 12	PCMDGUB3	Broken
Blade 77	PCMDGSN8	Cracked

(See Photo 3 Stage 1 HPT Rotor, Photo 4 Damaged Stage 1 HPT Blades, Photo 7 Stage 1 HPT Blades with Cracks Found)

- (5) The stage 1 HPT shroud had been substantially damaged with holes made in some areas.
 (6) The stage 2 HPT vanes had been damaged and all the vane leading edges were damaged.
 (7) All of the 74 stage 2 HPT blades had been damaged with their tips lost.
 (8) In the LPT with five stages in total, the airfoils of the stage 1 to stage 3 blades had been fractured around the whole circumference at various heights. The stage 4 and stage 5 blades and the stage 1 to stage 5 vanes had been all damaged and part of them had been lost.

Dents caused by the collision of fragments were found at three places near the upper part of the LPT case, but these had not been passed through.

(See Figure 4 CF6-80C2 Engine)

2.12.2 Examination of HPT Blades

For the investigation of this serious incident, a metallurgical examination of the stage 1 HPT blades of the Aircraft's No.1 engine was performed at a laboratory of the engine manufacturer in the United States of America. The result was as follows:

- (1) Pitting (very small holes created by corrosion) was found on the convex and concave side surfaces of the shank area of all blades. Debris accumulated to the shank area included sodium, potassium and sulfates.
 (2) Fatigue cracks with multiple initiation sites were found on the convex side surface of the shank area for the four blades which had been fractured below their platforms (blade 2 to 4 and 12). The size of each crack was as follows:

Blade 2 (two cracks):

0.260 in × 0.080 in (6.6 mm × 2.0 mm) and
 0.370 in × 0.086 in (9.4 mm × 2.2 mm)

Blade 3: 0.280 in × 0.104 in (7.1 mm × 2.6 mm)

Blade 4: 1.23 in × 0.44 in (31.2 mm × 11.2 mm)

Blade 12: 1.27 in × 0.46 in (32.3 mm × 11.7 mm)

Areas in these four blades with no fatigue cracks had been fractured due to tensile rupture.

(See Photo 5 Fractured Portion of Blade 2 of Stage 1 HPT, Photo 6 Fractured Blades

3, 4 and 12 of Stage 1 HPT)

(3) Cracks were found on the convex of the shank area of the following five blades:

Blade	Serial Number	Size of Crack
Blade 5	PCMDGUD7	About 8 mm
Blade 8	PCMDHCT0	About 5 mm and 3 mm
Blade 68	PCMDDHL3	About 0.5 mm
Blade 70	PCMDGSM9	About 0.5 mm
Blade 77	PCMDGSN8	About 17 mm

(See Photo 7 Stage 1 HPT Blades with Cracks Found)

(4) All four fractured stage 1 HPT blades had widened fatigue cracks with multiple initiation sites. As a result, fracture occurred at the platform area due to overload. Type II hot corrosion (to be described later in 2.13.3) was found in the under platform area for all blades.

2.12.3 Teardown Inspection of Other Engines

Under the contract between the Company and the engine manufacturer, the No.1 engine (serial number: 706701, the total time: 18,862 hours, the total cycles: 3,500 cycles) of the Aircraft, which had the highest total time and total cycles in the Company, was removed from the Aircraft in January 2010 and then overhauled at a facility designated by the engine manufacturer in order to examine the engine in February of the same year.

As a result, the Company received a report that 66 of the 80 stage 1 HPT blades would have to be scrapped. Cracks were found in the shank area for most of the blades to be scrapped.

Later, 76 of the 80 stage 1 HPT blades were examined at a laboratory of the engine manufacturer in the United States of America, and pitting was found on the convex and concave side surfaces of the shank area for all blades.

2.13 Additional Information

2.13.1 Maintenance Manual

(1) The Company's maintenance manual includes the following remarks about BSIs for the stage 1 and stage 2 HPT blades of the engine (Excerpt):

Maintenance item number: 72-141-01

Maintenance item: combustion chamber, stage 1 and stage 2 high pressure turbine blades

Interval of maintenance: 400 cycles or 2,000 flight hours, whichever comes first.

Method of maintenance: Borescope Inspection (BSI)

2.13.2 S/B Issued by the Engine Manufacturer

The following descriptions were included in S/B 72-1269 issued by the engine manufacturer regarding underplatform corrosion protective coating for stage 1 HPT rotor blades (Excerpt):

CF6-80C2

SERVICE BULLETIN

ENGINE – HPT Rotor (72-53-01) – Introduction of Stage 1 HPT Rotor Blades with Underplatform Corrosion Protective Coating

REVISION 1 TO SERVICE BULLETIN 72-1269

1. Planning Information

A. Effectivity

CF6-80C2B1F (and dozens of other kinds)

The stage 1 high pressure turbine rotor (HPTR) blade assembly (stage 1 HPTR blades) 1538M90P11/P12 (and several other kinds) are affected by this Service Bulletin.

B. Description

This Service Bulletin provides rework instructions to add a corrosion protective coating to the shank area of the blade below the platform as a field rework.

C. Compliance

Category 6*⁴

GE recommends that you do this Service Bulletin when the stage 1 HPTR blades are routed for repair.

NOTE: This Service Bulletin can be accomplished in shop.

This Service Bulletin is offered to improve the reliability or performance of your GE product, or to help prevent the occurrence of the event or condition described in this Service Bulletin. If the operator elects not to participate in the bulletin, that decision will be taken into consideration by GE in evaluating future product performance issues that may arise in the operator's fleet.

E. Reason

(2) Condition:

Pitting and cracking of the stage 1 HPTR blades in the shank area due to type two hot corrosion has become one of the leading causes for blade scrap at exposure. Four operational events of stage 1 HPTR blade separations have occurred in the CF6-80C2 and CF6-80E1 fleet from corrosion initiated cracking in the shank area.

(3) Cause:

Type two hot corrosion of the base blade alloy is caused by the presence of sodium, sulfates and potassium and the operating temperature in the shank area of the blades.

(4) Improvement:

Material testing shows the protective coating significantly improves protection against type two corrosion attack.

2.13.3 Hot Corrosion

During the combustion of fuel in gas turbine engine, sulfur and salt like sodium chloride react with each other at high temperature and produce sodium sulfate (Na₂SO₄) and others. These products adhere to the HPT blades and other parts and when accumulated, react with the oxide layer of the blades in an environment with high temperature and sometimes corrode the base metals. This is called hot corrosion. In general, sulfur is introduced through the jet fuel, while salt is

*⁴ Category 6 is one of the nine compliance categories given in the engine manufacturer's S/B, which range from Category 1 (mandatory) to Category 9 (information only). Category 6 is a level which calls for the operator to do the S/B for the affected part usually when it is exposed and repair is planned.

introduced by ingesting air with high sodium content during over sea operations.

Hot corrosion is usually divided into two categories. Type I hot corrosion tends to occur at temperatures around 870 °C (about 1,600 °F). It does not depend on the amount of sulfur. But type II hot corrosion tends to occur at temperatures around 700 °C (about 1,300 °F), and the reaction speed increases as the amount of sulfur increases.

Type II hot corrosion creates pitting in a local area.

2.13.4 Comments from Engine Manufacturer

The following comments were presented by the engine manufacturer in response to the JTSB's questions:

- (1) GE believes the corrosion begins when corrosive elements from the operating environment (air pollution, de-icing fluid, etc.) are introduced/ingested into the engine. Since GE cannot determine the time or location of introduction/ingestion, we have no data to determine what is short term or long term. And, we cannot provide a recommendation for optimal replacement time for stage 1 HPT blades.
- (2) GE has no information to show that trend data can be used to develop a type II hot corrosion evaluation standard. Type II hot corrosion of the HPT blade shanks has no noticeable affect on engine parameters.
- (3) Detection of the presence of type II hot corrosion requires physical inspection of the blade shanks. GE has no knowledge of a method for detecting type II hot corrosion while the blades are installed in the engine. The shanks are not visible using BSI when installed in the engine.
- (4) Regarding the possibility of the HPT blades being fractured due to type II hot corrosion if they do not apply the SB72-1269, SB72-1288 or SB72-1296, GE has presented related type of information via Fleet Highlites articles, Field Service Engineering team and Customer meetings.
- (5) GE believes that the Category 6 classification for the SB 72-1269 was appropriate. The CF6-80C2 fleet has seen only three events of type II hot corrosion related HPT blade separation over eight years. GE has noted that this type of corrosion has been a piece-part inspection finding. GE has also noted that some customers have not seen corrosion, including another Japanese CF6-80C2 operator. The recommendation to complete inspection during the repair cycle is an appropriate recommendation.
- (6) GE believes the appropriate recommendations and materials have been provided to manage type II hot corrosion in the shank area of HPT blades. GE will continue to monitor the fleet to insure that corrosion protection programs are effective.
- (7) The temperature of the shank area of the stage 1 HPT blades can reach a temperature zone in which type II hot corrosion occurs.

3. ANALYSIS

3.1 Qualification of Personnel

PIC and FO held both valid airman competence certificates and valid aviation medical certificates.

3.2 Airworthiness Certificate of the Aircraft

The Aircraft had a valid airworthiness certificate and had been maintained and inspected as prescribed.

3.3 Relation to Meteorological Phenomena

It is considered highly probable that the meteorological condition at that time had no relation with this serious incident.

3.4 Damage to Engine

3.4.1 Partial Fracture of Stage 1 HPT Blades

As described in 2.12.2:

- (1) There were traces of fatigue in the fractured surfaces of all of the four fractured blades, blades numbered 2 to 4 and 12.
- (2) Cracks were found in the shank area of five blades.
- (3) Signs of type II hot corrosion were seen in the shank area of all blades.
- (4) Debris accumulated in the shank area included sodium, potassium and sulfates.

Judging from this, it is considered highly probable that pitting had been caused by type II hot corrosion occurred in the shank area below the platform of the stage 1 HPT blades and cracks occurred from there and then, fatigue cracks widened with cycle stress applied on the blades following the engine's operation, leading to the fractures of the blades.

As described in 2.12.2 (2), the fatigue cracks in blades 4 and 12 were seen widely in the fractured areas as a whole. Therefore, it is considered highly probable that initial fractures occurred in blades 4 or 12.

Traces of fatigue cracks were also seen in the fractured surfaces of blades 2 and 3, though less widely than in blades 4 or 12. Therefore, it is considered highly probable that following the fracture of blade 4, the neighboring blades 3 and 2, already weakened by existing fatigue cracks, fractured completely due to impact from blade 4.

3.4.2 Damage to NGV, Stage 1 HPT Shroud, Stage 2 HPT and LPT Sections

It is considered highly probable that the damage to the trailing edges of the NGV, as described in 2.3.2 (3) and 2.12.1 (3), occurred because broken blade fragments from the four fractured stage 1 HPT blades had damaged the stage 1 HPT shroud and at the same time, hit the trailing edge of NGV located just in front of the shroud.

It is considered highly probable that while damaging the outer circumference of the leading edges of the stage 2 HPT vanes, the broken pieces of the fractured blades hit the stage 2 HPT blades damaging them all. Then the broken blades fragments moved to the LPT with five stages in total damaging the LPT vanes and blades of each stage, eventually impacting and damaging the TRF before exiting the engine.

As described in 3.4.1, the propagation of the fatigue cracks in the shank area of the stage 1 HPT blades triggered the fractures of the blades. It is considered highly probable that damage to the NGV just in front of the stage 1 HPT blades, the stage 1 HPT shroud and components in the rear section had been caused secondarily.

3.4.3 Occurrence of Type II Hot Corrosion

Judging from the descriptions in 2.13.2 E (3), 2.13.3 and the following, it is considered probable that the shank area of the HPT blades had been in an environment where type II hot

corrosion easily occurs.

- (1) As described in 2.13.3, jet fuel contains a small amount of sulfur, while salt exists in the air.
- (2) As described in 2.12.2 (1), debris accumulated to the shank area had included sodium, potassium and sulfates.
- (3) As described in 2.13.4 (7), because the shank area of the stage 1 HPT blades is not directly exposed to combustion gas, it is considered somewhat likely that the temperature of the area was at around 700 °C, a temperature zone where type II hot corrosion easily occurs.

It is considered somewhat likely that type II hot corrosion had been caused not only by sulfur contained in jet fuel but also by corrosion-triggering substances ingested from the air into the engine, as described in 2.13.4 (1). But the cause for the phenomenon could not be determined because the Aircraft's operation had not concentrated on any specific region as described in 2.10.

3.4.4 Engine Control at the Time of Damage

It is considered highly probable that, since the Aircraft's No.1 engine flowpath hardware was substantially damaged, part of the stage 1 HPT blades got fractured, N1 and N2 suddenly dropped, and as a result, the engine control unit automatically worked to drop FF in accordance with the established controlling schedule and this led to a drop in EGT.

Judging from the DFDR data, it is considered highly probable that after the Aircraft reached an altitude of about 7,000 feet, the throttle lever for the No.1 engine was retarded, the fuel control switch was set to the cut-off position and then the engine was shut down.

3.5 Possibilities of Early Trouble Finding

In accordance with the maintenance manual as described in 2.13.1, BSIs had been performed for the engine, but as described in 2.13.4 (3), the shank area of the HPT blades is not visible in BSI. Therefore, it is considered highly probable that the Company's staff could not find indications of cracks or corrosion in the blade shanks.

As described in 2.10, there was no clear shift in the trend data which would have indicated abnormalities in the data about the engine since it was installed in the Aircraft as its No.1 engine. As described in 2.13.4 (2), it is considered probable that type II hot corrosion of the HPT blade shanks has no noticeable effect on engine parameters. Therefore, it is considered highly probable that only with the trend data, the Company and the engine manufacturer could not recognize a condition of deterioration of the shank area of the HPT blades.

Accordingly, the occurrence of type II hot corrosion and its progress cannot be easily confirmed in the shank area below the platform of the HPT blades. To confirm the condition of the area, all blades must be checked individually during a shop visit.

As described in 2.9, the engine manufacturer issued S/B 72-1269 recommending adding a corrosion protective coating on the shank area of the stage 1 HPT blades. As described in 2.13.2, the S/B had been given the compliance level of category 6, a less urgently needed level which does not require the operator to remove the engine prematurely to comply with the S/B. Since the S/B recommended doing the work when the stage 1 HPT blades are routed for repair, as described in 2.10, the Company had planned to perform the S/B 72-1269 during the next engine overhaul. As the compliance level is given as category 6, there may be no opportunity to confirm the condition of the shank area until it becomes necessary to repair the blades. But as described in 2.13.4 (5), the engine manufacturer believes that the Category 6 classification for the S/B was appropriate because the number of findings of similar phenomena was limited. As described in 2.9 (4) and (5), the engine

manufacturer had already released a new blade with corrosion protective coating applied for the shank area of the stage 1 HPT blades.

3.6 Responses after Engine Failure

Judging from the descriptions in 2.1.1 and 2.1.2, it is considered highly probable that, the crewmembers of the Aircraft did not feel any engine anomalies from the start of the engine to taxiing to the takeoff roll. They recognized the occurrence of a No.1 engine failure when they saw messages on the instrument displays just after the Aircraft lifted off.

Judging from the descriptions in 2.1.1 and the DFDR data, it is considered highly probable that: PIC engaged the autopilot after correcting the veering of the heading of the Aircraft to the left following the No.1 engine failure. Meanwhile, the master warning was not activated. Therefore, PIC did not judge that the situation requires any urgent step, as the engine was already in an idle condition and that he continued to climb without shutting down the engine until the Aircraft reaches a fully safe altitude.

4. PROBABLE CAUSES

It is considered highly probable that this serious incident occurred as follows: When the Aircraft took off, one or more of the stage 1 HPT blades from the No.1 engine fractured at the shank area below the platform and the blade fragments went downstream and damaged the NGV, the stage 1 HPT shroud and components in the rear section.

It is considered highly probable that the fractures of part of the stage 1 HPT blades had been caused by pitting due to type II hot corrosion in the shank area below the platform and cracks grew from there, which led to fatigue cracks in the blades with cycle stress applied on them.

5. ACTIONS TAKEN

Actions taken by the Company

After the occurrence of this serious incident, in order to prevent a recurrence of similar events, the Company has been gradually removing engines which have no corrosion protective coating on the shank area of the HPT blades and replacing these blades with new ones which have corrosion protective coating applied to the blade shank.

Figure 1 Estimated Flight Route

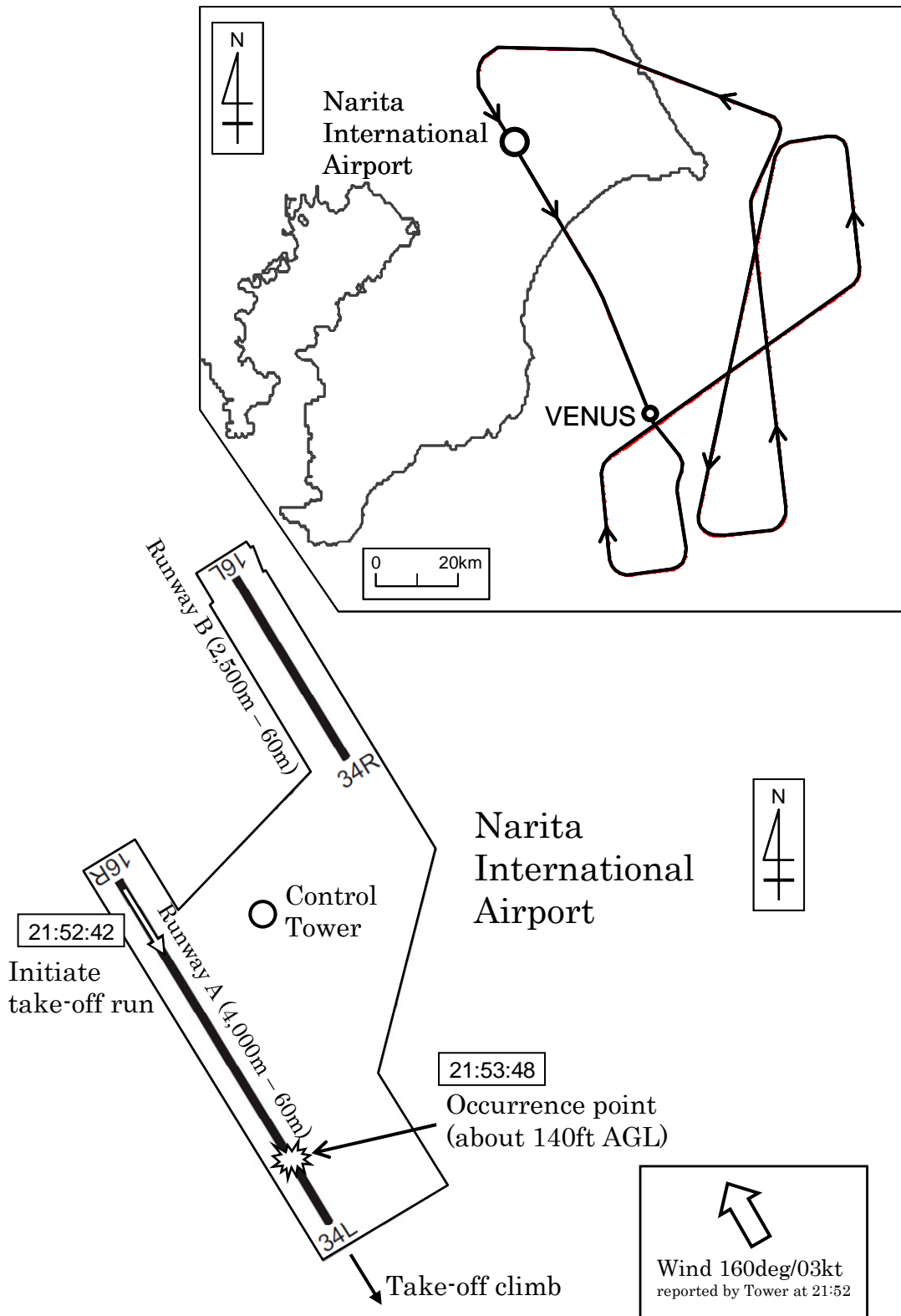


Figure 2 DFDR Records

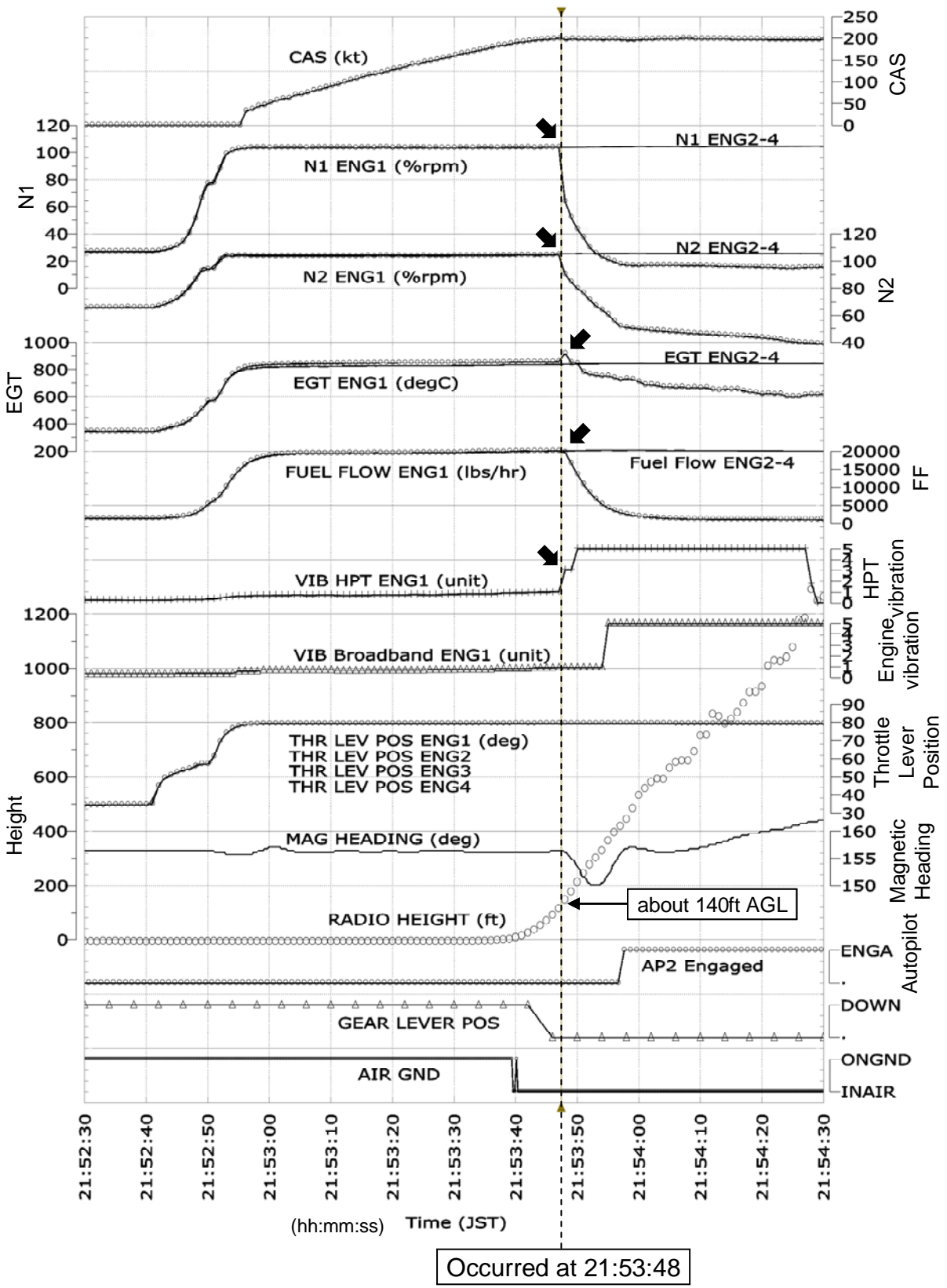


Figure 3 Three Angle View of Boeing 747-400F

Unit : m

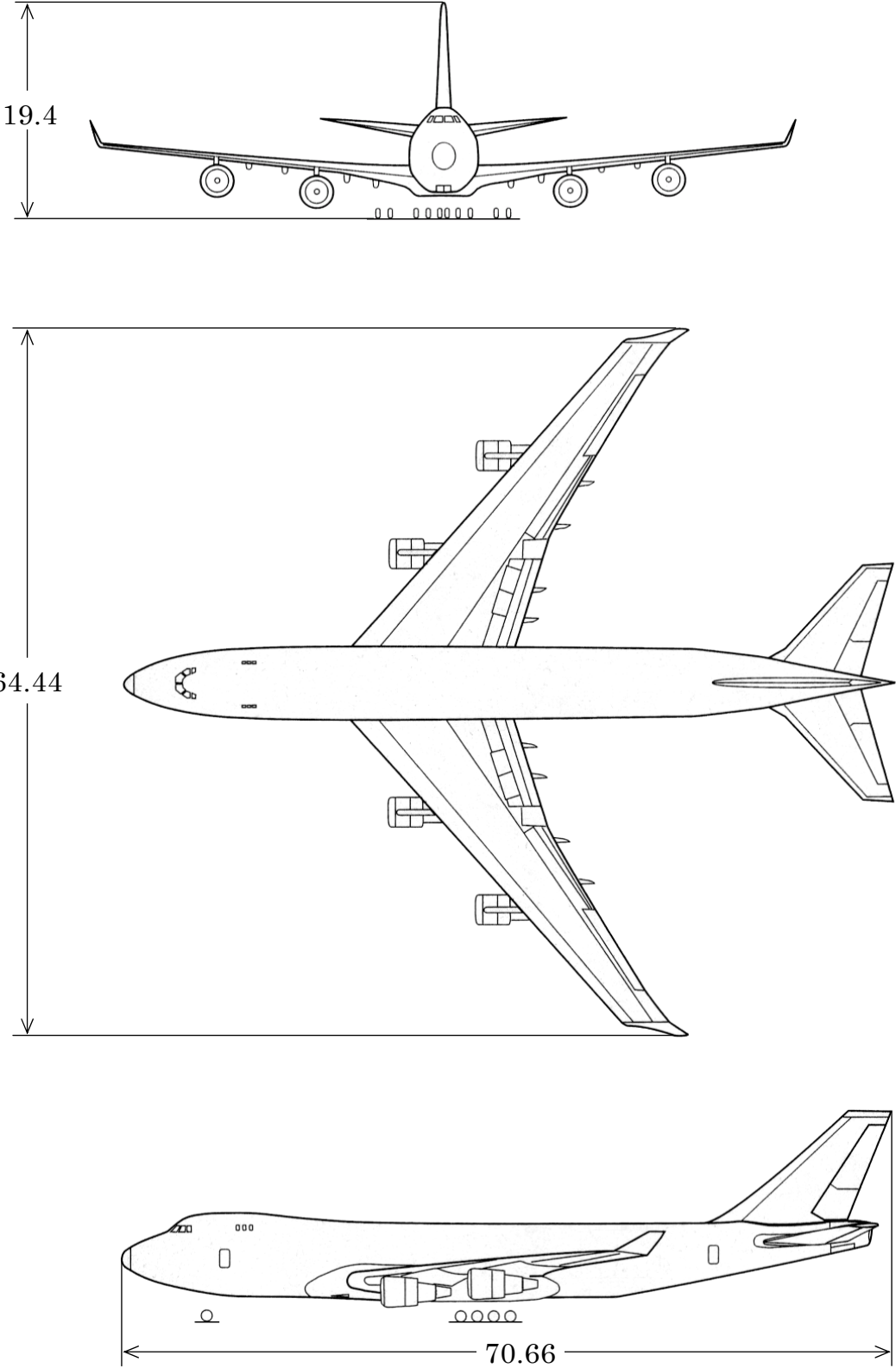


Figure 4 CF6-80C2 Engine

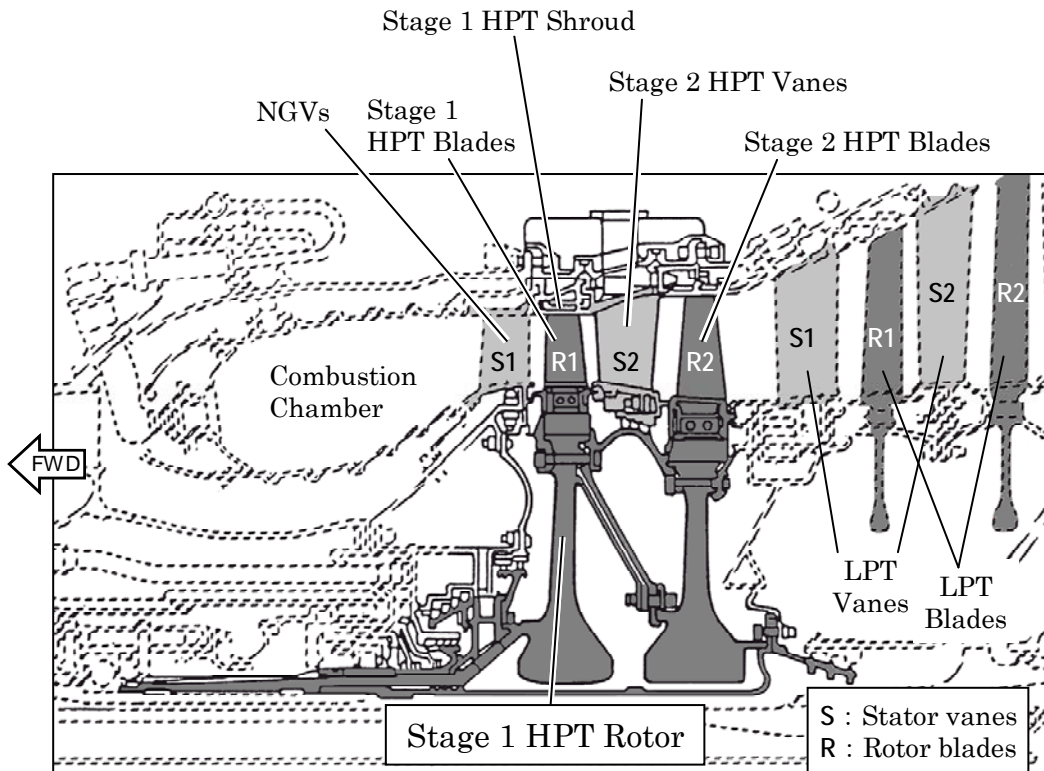
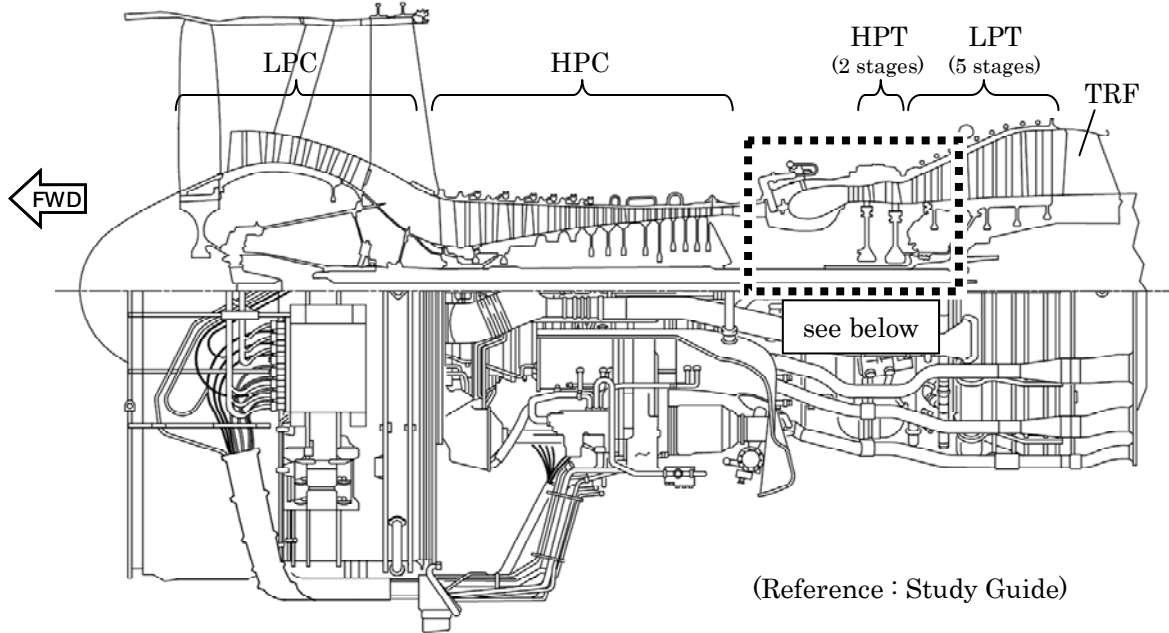


Photo 1 Serious Incident Aircraft



Photo 2 Dismounted No.1 Engine



Photo 3 Stage 1 HPT Rotor

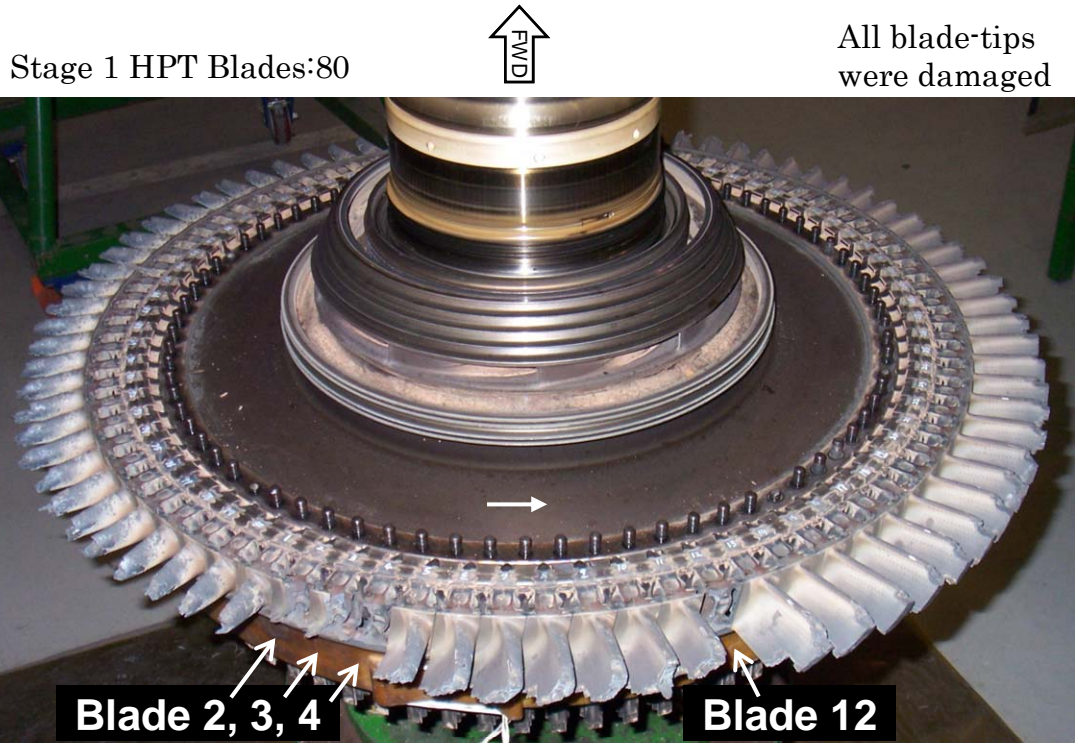


Photo 4 Damaged Stage 1 HPT Blades



Photo 5 Fractured Portion of Blade 2 of Stage 1 HPT

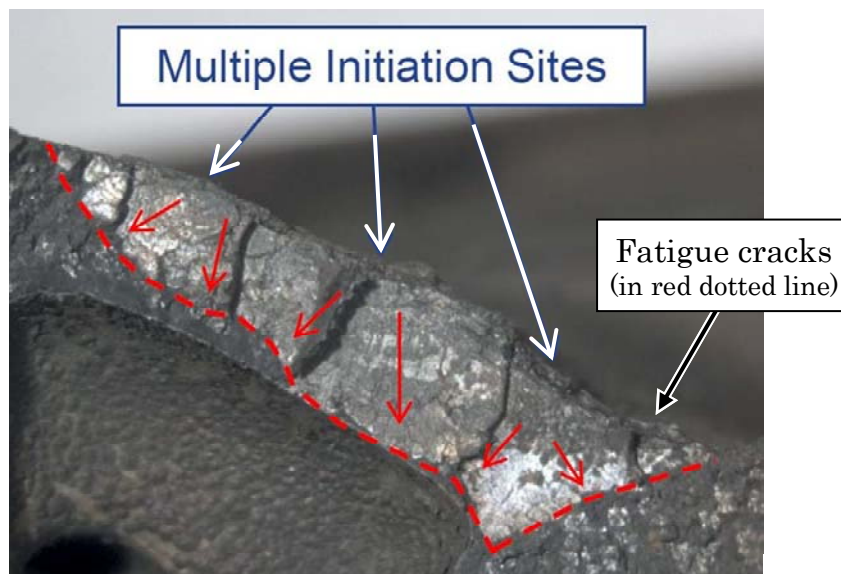
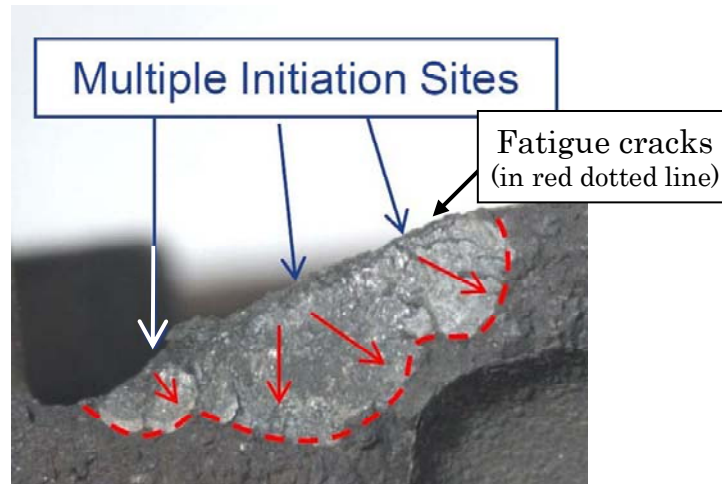


Photo 6 Fractured Blades 3, 4 and 12 of Stage 1 HPT

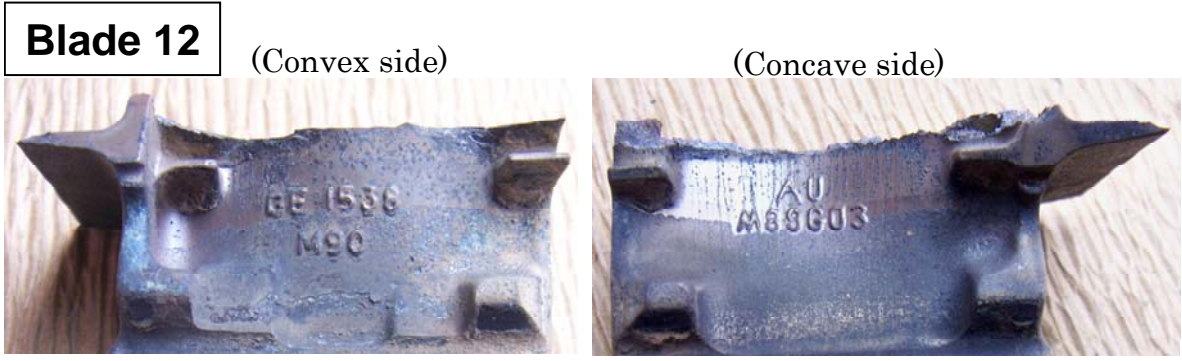
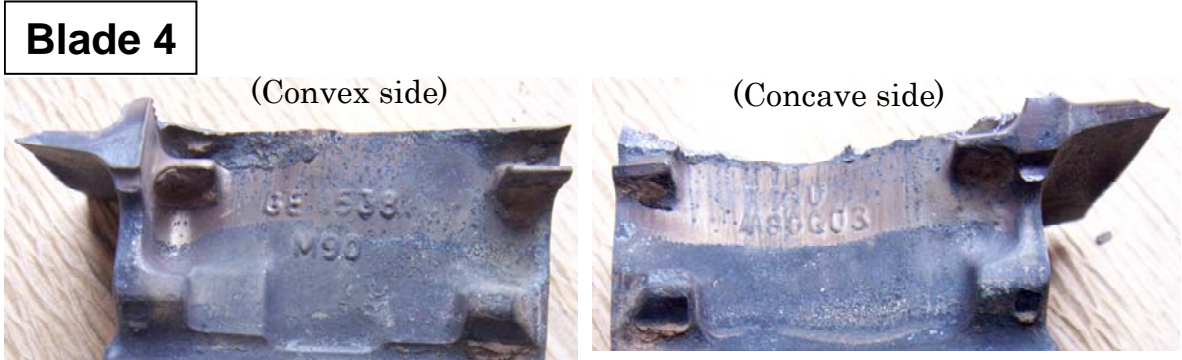
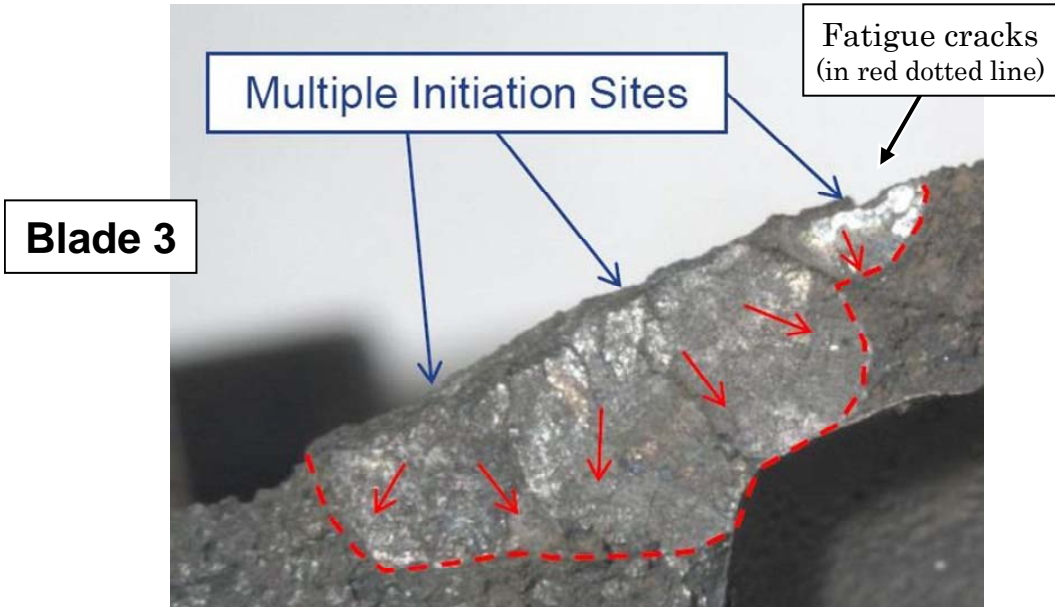


Photo 7 Stage 1 HPT Blades with Cracks Found

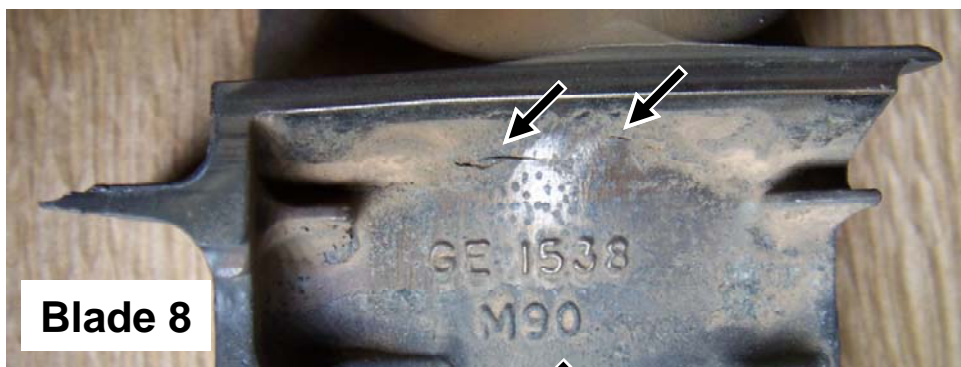
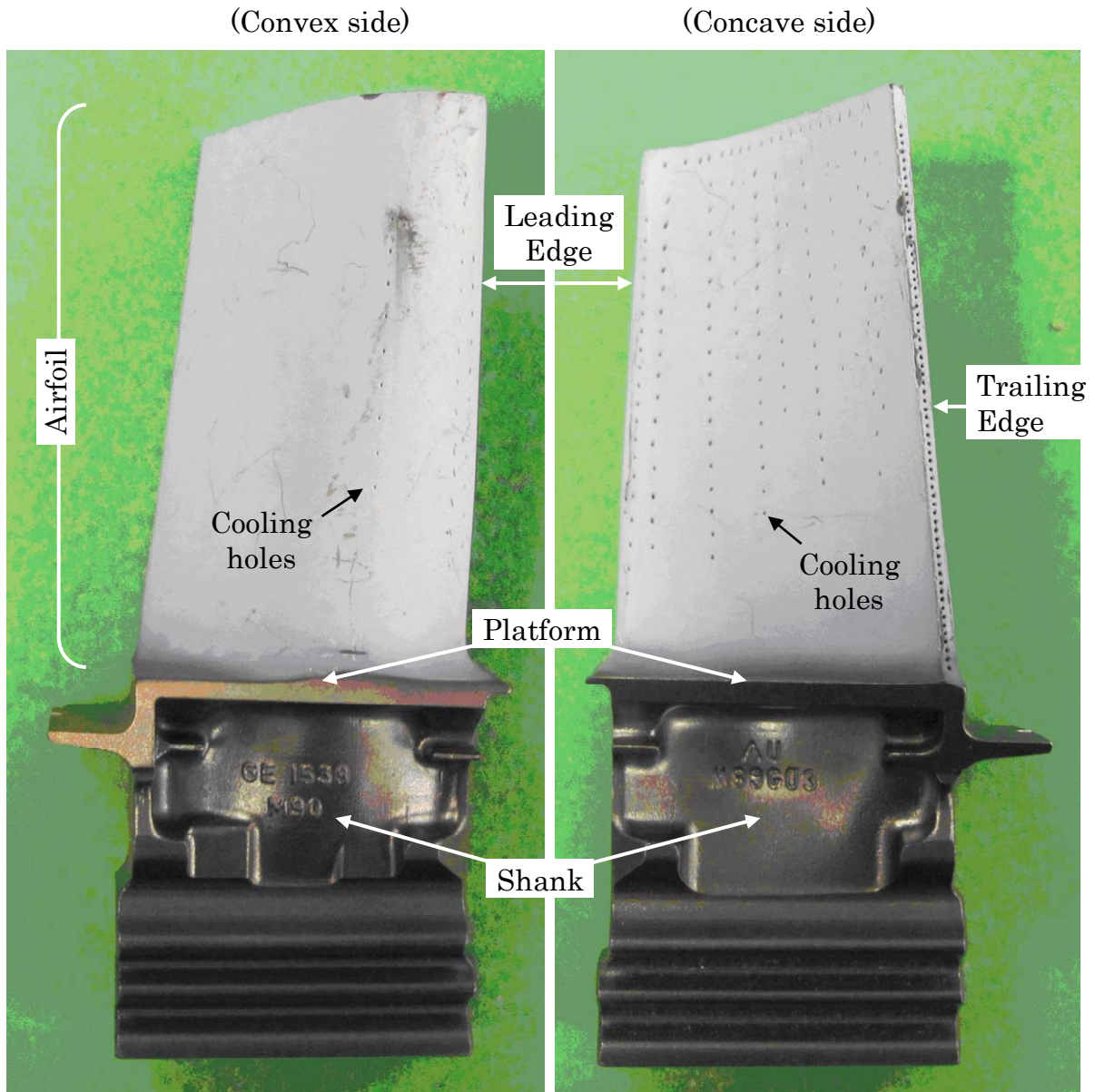


Photo 8 Stage 1 HPT Blade (Unused)



Part Number : 1538M90P12