AI2009-1

# AIRCRAFT SERIOUS INCIDENT INVESTIGATION REPORT

SKYMARK AIRLINES Co., Ltd.

J A 7 6 7 B

January 23, 2009

Japan Transport Safety Board

The investigation for this report was conducted by Japan Transport Safety Board, JTSB, about the aircraft serious incident of SKYMARK AIRLINES, B767-300 registration JA767B in accordance with Japan Transport Safety Board Establishment Law and Annex 13 to the Convention on International Civil Aviation for the purpose of determining causes of the aircraft serious incident and contributing to the prevention of accidents/incidents and not for the purpose of blaming responsibility of the serious incident.

This English version of this report has been published and translated by JTSB to make its reading easier for English speaking people who are not familiar with Japanese. Although efforts are made to translate as accurately as possible, only the Japanese version is authentic. If there is any difference in the meaning of the texts between the Japanese and English versions, the text in the Japanese version prevails.

Norihiro Goto,
Chairman,
Japan Transport Safety Board

[Abbreviation]

Abbreviations etc. applied in this reports are as follows.

| | | |
|---|---|---|
| AD | : | Airworthiness Directive |
| ACCS | : | Active Clearance Control System |
| AIP | : | Aeronautical Information Publication |
| AMM | : | Aircraft Maintenance Manual |
| AOM | : | Airplane Operating Manual |
| APU | : | Auxiliary Power Unit |
| BPTO | : | British Petroleum Turbo Oil |
| BSI | : | Bore Scope Inspection |
| CAS | : | Computed Airspeed |
| CCCV | : | Core Cooling Control Valve |
| CCV | : | Concave side (凹) |
| CDP | : | Compressor Discharge Pressure |
| CDU | : | Control Display Unit |
| CESM | : | Commercial Engine Service Memorandum |
| CRF | : | Compressor Rear Frame |
| CVR | : | Cockpit Voice Recorder |
| CVX | : | Convex side (凸) |
| DFDR | : | Digital Flight Data Recorder |
| DSR | : | Directionally Solidified Rene |
| eCA&V | : | e-Characteristic Accountability & Verification |
| EDS | : | Energy Dispersive X-Ray Spectroscopy |
| EGT | : | Exhaust Gas Temperature |
| EICAS | : | Engine Indication and Crew Alerting System |
| ENG | : | Engine |
| ESN | : | Engine Serial Number |
| FAA | : | Federal Aviation Administration (U.S.A.) |
| FADEC | : | Full Authority Digital Electronic (or Engine) Control |
| FL | : | Flight Level |
| FPI | : | Fluorescent Penetrate Inspection |
| GND | : | Ground |
| HMU | : | Hydraulic Mechanical Unit |
| HPC | : | High Pressure Compressor |

ＨＰＴ　　　: High Pressure Turbine
ＨＰＴＣ　　: High Pressure Turbine Cooling
lb　　　　　: Pound
ＩＦＳＤ　　: In Flight Shut Down
ＩＬＳ　　　: Instrument Landing System
in　　　　　: Inch
ＬＰＣ　　　: Low Pressure Compressor
ＬＰＴ　　　: Low Pressure Turbine
ＬＰＴＣ　　: Low Pressure Turbine Cooling
ＭＡＣ　　　: Mean Aerodynamic Chord
ＭＣＤ　　　: Magnetic Chip Detector
mil　　　　: 1/1,000inch
μm　　　　: Micro Meter ($10^{-6}$ m)
mm　　　　: millimeters
ＭＳＧ　　　: Message
ＮＤＩ　　　: Non Destructive Inspection
ＮＴＳＢ　　: National Transportation Safety Board (U.S.A.)
Ｎ１　　　　: Low Pressure System
(Low Pressure Compressor + Low Pressure Turbine) rpm (%)
Ｎ２　　　　: High Pressure System
(High Pressure Compressor + High Pressure Turbine) rpm (%)
ＮＧＶ　　　: Nozzle Guide Vane
Ｐ４９　　　: High Pressure Turbine discharge Pressure or Low Pressure Turbine inlet Pressure
ＰＦ　　　　: Pilot Flying
ＰＮＦ　　　: Pilot Not Flying
ＰＳ３　　　: High Stage compressor discharge total Pressure
psi　　　　: Pound per Square Inch
psig　　　: Pound per Square Inch Gauge
ＱＡＲ　　　: Quick Access Recorder
ＱＮＨ　　　: Altimeter Setting indicating the atmospheric pressure above Mean Sea Level
ＲＮＡＶ　　: Area Navigation
rpm　　　　: Revolutions Per Minute
ＳＥＭ　　　: Scanning Electron Microscope

ＳＢ　　　　　：Service Bulletin
ＳＮ　　　　　：Serial Number
ＴＡ　　　　　：Turn Around
ＴＬＡ　　　　：Thrust Lever Angle
ＶＡＡＣ　　　：Volcanic Ash Advisory Center
ＶＳＶ　　　　：Variable Stator Vane
ＸＲＤ　　　　：X-Ray powder Diffraction

# AIRCRAFT SERIOUS INCIDENT INVESTIGATION REPORT

SKYMARK AIRLINES CO., LTD.

BOEING 767-300

JA767B

AT ABOUT 16:45 JST, DECEMBER 1, 2005

OVER KAGOSHIMA AIRPORT RUNWAY,

December 10, 2008

Adopted by Japan Transport Safety Board

(Aircraft-Sub Committee )

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# 1. PROCESS AND PROGRESS OF AIRCRAFT SERIOUS INCIDENT INVESTIGATION

## 1.1 Summary of the Serious Incident

The event covered by this report falls under the category of "Flame in Engine Designated Fire Zone" as stipulated in Item 9, Article 166-4 of the Civil Aeronautics Regulations of Japan (at the time of the incident; it is now stipulated as Item 10 as the Regulations were amended on October 1, 2006) and, as such, was treated as an aircraft serious incident.

The Boeing 767-300 JA767B, operated as Skymark Airlines scheduled Flight 306, took off from Kagoshima Airport at 16:45 (Japan Standard Time, UTC+9h) on December 1, 2005 (Friday). Immediately after takeoff, vibration started on the right engine , and at about 16:48 the fire warning on the right engine was activated. The flight crew shut down the right engine and returned to the Kagoshima Airport at 17: 04 for a successful single engine landing.

Of the total of 90 persons on board, consisting of a pilot in command (PIC), a co-pilot, nine flight attendants, and 79 passengers, no one was injured. The aircraft suffered minor damage. After takeoff, the grass area alongside the runway caught fire.

## 1.2 Outline of the Serious Incident Investigation

### 1.2.1 Investigation Organization

On December 1, 2005, the Aircraft and Railway Accidents Investigation Commission (ARAIC) appointed an investigator-in-charge and two investigators to the serious incident.

For this serious incident, an expert advisor was appointed to investigate the following: Investigation on the fracture of turbine blades

Dr. Hiroshi HARADA
Managing Director of High Temperature Materials Center,
Independent Administrative Institution, National Institute for Materials Science
(Appointed on September 29, 2006)

### 1.2.2 Foreign Representative

An accredited representative of the United States of America, the State of the Design and Manufacture of the incident aircraft and the engine, participated in the investigation.

### 1.2.3 Implementation of Investigation

| | |
|---|---|
| December 2 – December 7, 2005 | Investigation of the aircraft and interviews |
| December 2, 2005 – February 28, 2006 | Analysis of flight data recorder, quick access recorder and cockpit voice recorder |
| December 13 and 14, 2005 | Investigation of the engine |
| January 9 – 13, 2006 | Teardown examination of the engine (The investigation was carried out in cooperation with the National Transportation Safety Board (NTSB) of the United States of America.) |
| February 13 – June 1, 2006 | Metallurgical evaluation of the turbine |

| | |
|---|---|
| | blades<br>(With NTSB) |
| October 31 – November 3, 2006 | Investigation of design and manufacturing of the turbine blades<br>(With NTSB) |
| April 3, 2007 | Investigation of the response time of the fire detectors<br>(With NTSB) |

1.2.4 Notification of Factual Information to the Civil Aviation Bureau

The following information obtained from the fact-finding investigation was provided to Civil Aviation Bureau.

(1) Through investigation of the engine, the fracture of the blade of stage 2 high-pressure turbine and the damage to other blades were confirmed (provided on December 14, 2005).

(2) Disassembling the engine revealed that grayish white deposits had adhered to the compressor and the turbine (provided on January 17, 2006).

1.2.5 Interim Report

On November 24, 2006, the interim report based on the results of the fact-finding investigation up to that date was submitted to the Minister of Land, Infrastructure Transport and Tourism, and made public.

1.2.6 Comments from Parties Relevant to the Cause of the Serious Incident

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

1.2.7 Comments from Participating State

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

# 2. FACTUAL INFORMATION

## 2.1 History of the Flight

On December 1, 2005, the Boeing 767-300, JA767B (hereinafter referred to as “the Aircraft”) operated by SKYMARK AIRLINES CO.,LTD. (hereinafter referred to as “the Company,” the Company changed its name to SKYMARK CO.,LTD. on October 1, 2006) was scheduled to fly from Kagoshima Airport to Tokyo International Airport as the Company’s scheduled Flight 306.

The outline of the flight plan submitted by the Company to the Tokyo Airport Office (JCAB : Japan Civil Aviation Bureau ) was as follows.

Flight rules: Instrument flight rules; Departure aerodrome: Kagoshima Airport; Estimated off-block time: 16:35; Cruising speed: 461kt; Cruising altitude: FL370; Route: MELLY (Position Reporting Point) ～Y752 (RNAV[*1] route)～MADOG (PRP)～M750 (RNAV route) ～ SOPHY (PRP) ～ W28(airway) ～ SPENS(PRP) ～ Y211(RNAV route) ～ WESTN(PRP); Destination aerodrome: Tokyo International Airport; Estimated flight time: 1h and 16 min.; Fuel-loading in terms of flight duration: 3 h and 09 min.; Number of people on board: 90

At the time this serious incident occurred, in the cockpit of the aircraft, the PIC sat in the left seat as Pilot Not Flying (primarily responsible for non-maneuvering tasks) and the co-pilot sat in the right seat as Pilot Flying (primarily responsible for aircraft maneuvering).

### 2.1.1 Flight History and Engine Data

The flight history of the Aircraft, from the time it took off from Runway 34 (elevation 892 ft) of Kagoshima Airport to the time it returned and land at the airport in the course of this serious incident, and the data on the operating condition of the right engine, based on the records of a digital flight data recorder (hereinafter referred to “DFDR”), a quick access recorder (QAR), a cockpit voice recorder (CVR), the air traffic control (ATC) communications and the radar tracking, are summarized as below,.

Note that words in $\boxed{\quad\quad}$ in this report indicates the names of checklists, and those in [ ] indicate the letters of EICAS[*2] indications.

(1) History of the flight

| | |
|---|---|
| Around 16h37m | Started the engines. |
| 16h44m11s | Received the take off clearance from the Kagoshima Aerodrome Control Facility (hereinafter referred to as “the Tower”) |
| 16h44m58s | Started takeoff rolling |
| 16h45m05s | Set TLA[*3] at takeoff position (left 74.5°, right 74.4°) |
| 16h45m17s | Started rotation |
| 16h45m19s | The AIR/GND sensor of the main gears detected AIR. |

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*1 RNAV means an area-navigation for navigating any flight route within the coverage of the navigation aid facilities or within the capacity of the self-contained navigation system, or by both.

*2 EICAS stands for the Engine Indicating and Crew Alerting System, an electronic indication and warning device that displays engine data, and warns operating crewmembers, should an abnormality be found, of its location, and the like by indication of messages.

*3 TLA is the thrust lever angle.

*4 Vibration levels of the engine fans, LPTs, and N2, the highest of which is displayed on the EICAS in

| | |
|---|---|
| | Vibration levels*4 of the low-pressure turbine (hereinafter referred to as "LPT*5") of the right engine increased to 2.82 units |
| 16h45m20s | Co-pilot uttered "Well…." (sounded like he noticed that something was abnormal). The right EGT*6 indicated 926˚C, the maximum value. |
| 16h45m21s | The right engine LPT vibration increased to 4.96 units, the right fuel flow decreased to 16,544 1b/h. |
| 16h45m22s | The left and right TLA started to increase. (left 75.1˚, right 74.7˚) |
| 16h45m23s | The heading veered to the right at the maximum (angle of 4˚), then gradually returned to the original direction.<br>The PIC called, "Oh, I have control."<br>The vibration level of the right engine reached the upper measurement limit of 4.99 for the fan, N2*7 and LPT units, and the level lasted until 16h46m46s. |
| 16h45m24s | The co-pilot responded "You have control." |
| 16h45m26s | The PIC directed "Gear up." |
| 16h45m27s | The co-pilot repeated "Gear up," and the gear- lever was set to UP position. |
| 16h45m31s | The PIC said, "We have a problem with the right engine," to which the co-pilot replied "Yeah." |
| 16h45m35s | The PIC told the co-pilot to "Follow the engine failure reference procedures." The co-pilot responded "Roger." |
| 16h45m37s | The auto throttle was disengaged. |
| 16h45m44s | The Tower told the Aircraft, "Skymark306, contact Departure, good day." |
| 16h45m45s | TLAs were stabilized(left 77.7˚, right 77.3˚), the right fuel flow and the right EGT decreased (8,864 lb/h and 790˚C.) |
| 16h45m50s | The Aircraft informed the Kagoshima Terminal Radar Control Facility (hereinafter referred to as "the Radar"), "Kagoshima Departure, Skymark 306, engine failure, follow engine failure procedure." |
| 16h46m01s | The Radar told the Aircraft, "Skymark306, Kagoshima Departure, radar contact, say again please." |
| 16h46m03s | At about 1,700 ft above the ground level (pressure altitude 2,320 ft), the right TLA retarded to 68°, the right fuel flow was about 8,500 lb/h, the right EGT 780˚C, and the right oil pressure 52 psi. |

analog form and in digital form in units of in/sec. or Mils. Inspection is required if the level exceeds 4 units.

*5 LPT is the Low-Pressure Turbine, and be interlocking with Low-Pressure Compressor.

*6 EGT is the engine exhaust gas temperature, which is measured between the HPT high-pressure turbines and LPT low-pressure turbines. The maximum allowable value at takeoff is 960˚C.

*7 N2 is the rotation speed of the high-pressure compressor and high-pressure turbine of the engine, indicated with the number of rotations at 9,827 rpm, which is close to the maximum engine thrust, as 100 % for the Aircraft.

| | |
|---|---|
| 16h46m06s | The Aircraft reported to the Radar, "Right engine failure, following reference procedure." |
| 16h46m07s | The right EGT and the right oil pressure started to drop (738˚C and 49 psi). |
| 16h46m14s | The Radar asked the Aircraft, "Skymark 306, break, will you please speak in Japanese? I could not catch you.(in Japanese language)" |
| 16h46m20s | The Aircraft reported to the Radar, "Roger. The right engine failed, and we are going to follow the reference procedure*8 of Kajiki VOR/DME 141 (in Japanese language)." |
| 16h46m30s | The Radar told the Aircraft, "Skymark 306, roger (in Japanese language). We are standing by for your next intention." |
| 16h46m31s | Started right turn at a pressure altitude of about 3,060 ft. Continued climbing turn with an average bank angle of about 23 ˚ and average pitch angle of about 12˚. |
| 16h46m34s | The Aircraft told to the Radar, "Stand by." |
| 16h46m42s | The right TLA became at idle position (about 33.9˚). |
| 16h46m49s | Right engine LPT vibration level decreased to 4.2units, with the right N2 vibration level, the right P49*9 and the right oil pressure. (4.99 units, 18.0 psi, 31 psi.) |
| 16h46m51s | The right fuel flow stabilized (1,728 1b/h.) |
| 16h46m53s | The right EGT which had continued to lower since 16h45m20s. rose by around 30˚C and stabilized there for about 20 seconds (See (3)). |
| 16h47m13s | The PIC said, "I'm go on a try to put power on again." The right TLA was at 36.9˚. |
| 16h47m14s | The right TLA, the right fuel flow, the right EGT, the right engine N2 vibration and the right oil pressure and the like increased. (45.5˚, 2,048 lb/h, 548˚C, 4.99 units, 34 psi.) |
| 16h47m19s | The PIC said, "Very heavy vibration, yes?" |
| 16h47m22s | The co-pilot answered, "Yes." |
| 16h47m23s | The right P49 increased (33 psi). |
| 16h47m27s | The right PS3*10 increased (138 psi). |
| 16h47m29s | The right PS3 rapidly decreased (71 psi). |
| 16h47m30s | The right fuel flow and the right EGT increased (6,144 1b/h , 694 ˚C). |
| 16h47m31s | The right TLA increased, the right fuel flow decreased, the right N2 decreased, the right EGT decreased, the right oil pressure decreased and the right PS3 decreased. (67.1˚, 5,824 lb/h, 75.6 %, 606˚C, 34 psi, 32.5 psi). |
| 16h47m34s | The right TLA rapidly decreased to the idle position, and the |

*8 Reference procedure means the flight procedure, described in the Company's Operation Manual, which sets the route to take in the event of an engine failure, etc., for individual airports.

*9 P49 means the high-pressure turbine outlet pressure or low-pressure turbine inlet pressure.

*10 PS3 is the total pressure at the high-pressure compressor outlet.

| | |
|---|---|
| | right N1[*11] and N2 decreased.(33.9˚, 19.8%, 52.8%) |
| 16h47m35s | The right oil pressure rapidly decreased, the right PS3 and right P49 decreased. (18 psi, 19 psi, 13.4 psi) |
| 16h47m41s | The EICAS [R ENG OIL PRESS] (yellow) MSG was indicated.[*12] |
| 16h47m43s | All of the right engine vibration levels dropped to 0 unit[*13], the right PS3 and P49 became almost equal to P0[*14]. (15 psi, 12.8 psi, and 12.5 psi). |
| 16h47m50s | The right fuel flow (672 lb/h) |
| 16h47m55s | Started turning to left from right. |
| 16h48m40s | The Radar told the Aircraft, "Skymark 306, maintain at or below 10,000, we are standing by your intention." |
| 16h48m42s | The pressure altitude was about 5,070 ft, the master warning light illuminated, the fire bell[*15] sounded, the EICAS [R ENGINE FIRE] (red) MSG indicated, the right engine fire switch light illuminated, the right engine discrete fire warning light illuminated , and the right engine fuel control switch warning light illuminated. |
| 16h48m47s | The fire bell sounded again |
| 16h48m48s | The PIC declared "Engine fire" to the co-pilot. |
| 16h48m49s | The Aircraft told the Radar, "Stand by." |
| 16h48m52s | The PIC ordered the co-pilot "Right engine fire, fuel cut off, Pull." From then on, crewmembers continued procedures to shut down the right engine. |
| 16h48m56s | The right engine fuel valve closed, the fuel flow became 0. |
| 16h48m58s | The EICAS [R ENG SHUTDOWN] (yellow) MSG indicated. |
| 16h49m10s | The EICAS [R ENGINE FIRE] (red) MSG disappeared, the fire switch, the discrete fire warning light, the fuel control switch warning light, all went off. |
| 16h51m14s | The pressure altitude reached 6,000 ft. |
| 16h51m19s | The Aircraft told the Radar that the right engine had been shut down. |
| 16h52m00s | The Aircraft told the Radar that the aircraft had intention to return to Kagoshima Airport. |
| 16h53m04s | The Aircraft set the transponder to emergency code, under the Radar control, flew the final approach course and started ILS |

[*11] N1 means the rotation speed of the fans, low-pressure compressor and low-pressure turbines of the engine, indicated with the number of rotation at 3,280rpm which is close to the maximum engine thrust as 100 % for the event Aircraft.

[*12] The engine oil pressure warning illuminates, for example, if the oil pressure drops below 10 psi when N2 is 55%.

[*13] The engine vibration value stands at 0 when the engine rotation drops below a certain value. See 2.11.1 (7).

[*14] P0 means the outside air pressure adjacent the engine suction port.

[*15] The Fire Bell of the Aircraft can stop either when the fire is went out, the Master Warning switch is pressed, or the Fire Switch is pulled. The fire warning indication (other than the Fire Bell) remains illuminated so long as there is a fire signal, which operates when the temperature rises above a certain temperature.

approach. The operating crewmembers explained the situation to the flight attendants and passengers.

17h04m01s The Aircraft landed at Kagoshima Airport, shut down the left engine at Spot 3. Passengers were deplaned.

(2) The right engine data before and after airborne

| Time | AIR/GND sensor | Fuel flow (1b/h) | N1 (%) | N2 (%) | EGT (°C) | PS3 (psi) | Vibration (units)* |
|---|---|---|---|---|---|---|---|
| 16h45m15s | GND | 18,048 | 99.8 | 103.1 | 810 | 396.5 | 0.41 |
| 16h45m16s | GND | 18,048 | 99.8 | 103 | 810 | - | - |
| 16h45m17s | GND | 18,080 | 99.8 | 103.1 | 810 | 358.8 | 0.28 |
| 16h45m18s | GND | 18,080 | 99.8 | 103 | 810 | - | - |
| 16h45m19s | AIR | 18,336 | 81.0 | 103 | 858 | 249.5 | 2.82 |
| 16h45m20s | AIR | 17,728 | 64.4 | 102.6 | 926 | - | - |
| 16h45m21s | AIR | 16,544 | 60.3 | 100.1 | 924 | 223.0 | 4.96 |
| 16h45m22s | AIR | 15,488 | 57.9 | 97.1 | 902 | - | - |
| 16h45m23s | AIR | 14,432 | 56.3 | 94.8 | 886 | 200.5 | 4.99 |
| 16h45m24s | AIR | 13,408 | 54.6 | 92.9 | 868 | - | - |

(* Based on the DFDR and QAR data. The PS3 and vibration levels are recorded every other second.)

(3) The right engine data before and after the EGT increase (Values in shade: EGT increase)

| Time | AIR/GND Sensor | Fuel flow (1b/h) | N1 (%) | N2 (%) | EGT (°C) | PS3 (psi) | Vibration (units) |
|---|---|---|---|---|---|---|---|
| 16h46m45s | AIR | 2,336 | 25.1 | 72.4 | 504 | 54 | 4.99 |
| 16h46m46s | AIR | 2,208 | 24.3 | 71.3 | 498 | - | - |
| 16h46m47s | AIR | 2,080 | 23.5 | 70.3 | 490 | 48.5 | 4.99 |
| 16h46m48s | AIR | 1,952 | 22.8 | 69.1 | 484 | - | - |
| 16h46m49s | AIR | 1,824 | 21.9 | 67.9 | 476 | 48 | 4.99 |
| 16h46m50s | AIR | 1,760 | 21.4 | 67.3 | 478 | - | - |
| 16h46m51s | AIR | 1,728 | 21.1 | 67.3 | 488 | 48.5 | 4.99 |
| 16h46m52s | AIR | 1,728 | 21.0 | 67.3 | 498 | - | - |
| 16h46m53s | AIR | 1,728 | 20.9 | 67.3 | 504 | 48 | 4.99 |
| 16h46m54s | AIR | 1,728 | 20.9 | 67.3 | 506 | - | - |
| 16h46m55s to 47m00s | AIR | 1,728 | 20.9 | 67.4 | 506 | 48 | 4.99 |
| 16h47m01s to 47m04s | AIR | 1,728 | 20.9 | 67.4 | 504 | 48 | 4.99 |
| 16h47m05s to 47m10s | AIR | 1,728 | 20.8 | 67.4 | 502 | 48 | 4.99 |
| 16h47m11s to 47m13s | AIR | 1,728 | 20.9 | 67.4 | 502 | 48~56 | 4.99 |

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 16h47m14s | AIR | 1,856 | 21.3 | 68.9 | 518 | - | - |

The average CAS[*16] from the takeoff (airborne) of the Aircraft to the disappearance of the engine fire warning was about 158kt.

2.1.2 Statements from the Operating Crewmembers and the Persons Concerned

The history before and after this serious incident happened, based on the statements from the operating crewmembers and the persons concerned, is summarized below.

(1) The PIC

The flight from Tokyo to Kagoshima was made without anomalies. At the departure from Kagoshima Airport, I started the engines during pushback without anomalies. After the towing car was removed, I gave the controls to the co-pilot. At the time of airborne during take off from Runway 34, a high-amplitude low-frequency airframe vibration started. As it was a critical stage of flight time requiring full crew attention, I immediately took over the controls, and I ordered the co-pilot to pull up the landing gear lever. Although I initially suspected airframe problem, I found that the right engine N1 was as low as 50 to 60%. I told the co-pilot that we were going to fly according to the engine failure reference procedures, directed him to report to the ATC, and I started to retard the thrust lever of the right engine. As I retarded the thrust lever close to the idle position, the vibration became almost normal level.

I increased a little the thrust lever of the right engine, which had been set almost at the idle position, to check the engine response. At about that time, the engine fire warning was activated. I immediately initiated the $\boxed{\text{ENGINE FIRE OR SEVERE DAMAGE}}$[*17] procedure, and reconfirmed it with the checklist. When I pulled the Engine Fire Switch (hereinafter referred to as "the fire handle"), the fire indication went out, so it was not necessary to discharge the extinguishing agent. Having already told the ATC that the engine failure reference procedure course would be followed and to stand by for our further flight, while climbing on the course, I set the transponder to emergency code to declare an emergency to the ATC. I notified the flight attendants of the situation, and the co-pilot and I made announcements to the passengers in turn.

The weather was Visual Meteorological Condition (VMC), so I told the ATC that I would stop the climb at 6,000 feet, and immediately return to Kagoshima Airport by radar vectoring. I made a left turn to AMORI (PRP) (See Fig 1.) directly and made an ILS approach to runway 34 with left engine only. The landing was smooth and we could take taxiway T4. As there was no fire indication, I made a normal ramp-in to Gate 3. No bird strike, and as the weather was favorable, no lightning strike. Everybody stayed calm. I considered an emergency escape but decided against it, because the engine was shut down and the fire was out, moreover, executing an emergency escape could have triggered a panic among passengers, which might cause injuries such as broken bones, and I judged that normal disembarkation would be safer.

(2) The co-pilot

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[*16] CAS is the computer-processed airspeed obtained by precisely computing the temperature, altitude, compression errors, etc; it is displayed on the cockpit panel.

[*17] In this report, the check list of $\boxed{\text{ENGINE FIRE OR SEVERE DAMAGE OR SEPARATION}}$ is expressed as $\boxed{\text{ENGINE FIRE OR SEVERE DAMAGE}}$.

No anomalies were found during the exterior inspection before departure. I took the controls and started a take off with flaps 5° and DERATE-2[*18] thrust . A moment after the rotation, I felt a slow and large vibration. As the nose of the aircraft veered to the right, I countered the movement with the rudder. I was told by the PIC, “I have control.”, so I handed the control to him and the PIC directed me to make the gears up, and I did it. When the PIC reduced the right engine down to idle, the vibration became quiet.

After we found that we were able to fly, when the PIC examined the engine response, the fire warning of the engine activated. We executed the ENGINE FIRE OR SEVERE DAMAGE checklist. We set the transponder to emergency code. We explained to the passengers that there was nothing to worry about as we had undergone enough training to handle such situations. I thought that the PIC considered no necessity of an emergency evacuation.

(3) A mechanic of the company at Kagoshima Airport

When the aircraft arrived at Kagoshima Airport from Tokyo as the flight 305, the crewmembers did not mention any problems with the aircraft equipment. During a check between flights, I did not find any abnormalities with the engines, such as air inlets, exhaust pipes. At the departure, the engines started normally. After its takeoff, I heard over the radio that the Aircraft would return due to an engine trouble. After the Aircraft ramped in, I asked about the situation to the crewmembers over the interphone, they said, “We experienced vibration after takeoff. It was such a heavy vibration so we reduced the power. Then we tried to increase the power a little, the engine fire warning activated, that we reduced the power again and pulled the fire handle. Then, the fire warning went out, so we did not discharge the extinguishing agent.”

(4) Chief purser

Immediately after takeoff, we had the kind of vibration that occurs once in a while as the gears were retracted, but I thought that this vibration was different from what I had experienced before; then the cabin lights were turned down a bit. I confirmed with the PIC and he told me that we were having engine trouble. After that we were briefed that we were flying back to Kagoshima Airport, I made an announcement to the passengers. The landing was as usual. After spot in, the passenger decks were connected, I confirmed with the PIC and led the passengers to disembark the plane.

(5) Passenger 1

At the takeoff, I saw a burst of many thin flames spew from the engine for a moment. After that, the Airplane was jolting and I saw thin white streams like cloud or smoke coming out of the engine. It seemed to me flying slower and climbing slower than usual. There was no odor or smoke in the cabin.

(6) Passenger 2

The Aircraft kept rattling at takeoff. I felt something was different from usual. After a while from the take off, white smoke was flowing out from the right engine, and later, I do not remember when, there were flames just like those from a burner emitted a few times, and then they were gone. The cabin crew explained the situation, then we received the PIC announcement that one of the engines was failed and we would land at Kagoshima Airport.

(7) Tower controller

Having received a report from the Aircraft that it was ready to takeoff, I gave it

[*18] DERATE-2 means the takeoff thrust which is set approximately at 80% of the maximum takeoff thrust.

clearance to take off from Runway 34 as usual. The wind direction was 270°, and the velocity was four to five kt. After the Aircraft took off, I transferred it to the Departure Control (the Radar) at usual position. I watched the Aircraft until its gears were retracted, but I heard the engine sounds quite normal, and I did not notice any abnormalities. Then from the communication between the Radar and the Aircraft I kept monitoring voluntarily, I knew that they had some trouble with the engine. I watched the bright display*19 of the control tower, and assumed that the Aircraft would return to Kagoshima Airport via traffic pattern or by radar vector, so I instructed a commuter airplane that was to takeoff next, to stand by. After that, as the Aircraft was flying at 4,000 to 5,000 ft, I coordinated with the Radar to decide if the airplane can be cleared for takeoff. The Radar told me to fly it by "RUNWAY HEADING." At this point, I saw some smoke from the grass area to the east of the runway. I hesitated a moment and told the commuter airplane about the smoke. The pilot of the commuter airplane reported that the smoke would not hamper the takeoff, so I gave a clearance for take off. The commuter airplane took off. From the retrospect, there was a relation between the smoke occurrance and the engine failure, but at that point, I could not imagine the relationship with these phenomena. I saw on the bright display that the Aircraft was sending an emergency code. After that, I thought that another commuter airplane could be cleared for take off and I made it take off. It was twilight, and not raining. The Aircraft landed without problem, and ramped in via taxiway T4. The runway check done after the landing of the Aircraft revealed that fragments were scattered, so we closed the runway.

(8) Eyewitness on the ground

When the Aircraft took off, I was near Spot 3, but nobody heard any abnormal sounds. After a while, I heard that the Aircraft might return, and looked up the sky from around Spot 3, then the Aircraft was making a right turn but I did not see anything like smoke from the Aircraft. After the Aircraft completed the right turn, and flew over the Kagoshima Airport, I could see a thin brownish line trailing from the right engine. So I thought something was wrong with the right engine.

This serious incident occurred, as described later, over Kagoshima Airport Runway (latitude 31°48´ North and longitude 130°43´ East) at about 16:45. (See Figures 1, 3, and 18, and Photos 1, 2, 4 and 5.)

## 2.2 Injuries to Persons

None

## 2.3 Damage to the Aircraft

The investigation of the airframe revealed that the outboard flap and inboard aileron of the right wing and the right engine were damaged, details of which are as follows. No other parts were damaged. Note that hereinafter angular positions from the engine rotation shaft are expressed by positions of the hour hand of a clock viewing the fuselage from aft to forward.

(1) High- pressure turbine (hereinafter referred to as " HPT"*20) and LPT

*19 Bright display is the radar display unit installed in the control tower and used by the controllers.

*20 HPT stands for a high-pressure turbine toward which combustion gas is blown. The HPT is made up of two-stage nozzle guide vanes and blades.

① The HPT 1st stage nozzle guide vanes were not damaged.
② The HPT 1st stage blades exhibited slight traces of tip rub.
③ The HPT 2nd stage nozzle guide vanes were not damaged.
④ Of the 74 blades of the HPT 2nd stage, Blade #58 was fractured at the shank*21 located between the root (dove tail) and the platform. Blade #57 was fractured in the middle of its airfoil. Blade #56 was fractured near the tip of its airfoil.
⑤ Other blades and the engine interior flow path of the HPT 2nd stage were partially damaged.
⑥ Almost all of the LPT blades were fractured and the LTP parts including the stator*22 and engine case were damaged.

(2) LPT case

① Of the LPT 1st stage, at about 6h30m position, there was a hole which was 15 mm long and 7 mm wide, and at about 9h position, there was another hole which was about 10 mm long and wide.(See photo 9.)
② The surrounding areas of these holes were covered with soot.
③ On the inner diameter of the core cowl*23 near these holes, there were no traces of penetration or contact marks of part such as turbine blades.

(3) Fuel supply tube ( pigtail) #6

① Fuel supply tube #6 at 2h30m position was fractured near the B nut(Built in Nut) coupling with the fuel nozzle.
② The retaining ring*24 inside the fuel nozzle shroud can*25 (hereinafter referred to as "shroud can") covering the B nut coupling joint of the fuel supply tube with the fuel nozzle, and the O-ring sealing the gap between the fuel supply tube and the shroud can, were displaced from their normal positions and left on the upstream part of the fuel supply tube.(See photo 12.)
③ The mounting base (ferrule) of the O-ring of the fractured fuel supply tube was dislodged from the shroud can, making a crescent-shaped opening centered at 3h position of the engine (core cowl side). The U-shaped portion of the fuel supply tube was bent a little between its bend and the fuel manifold.

(4) Burning soot condition on the exterior surface of the engine case

① The lower half of the rear surface of the accessory gearbox heat shield*26 mounted on the lower exterior of the engine case was discolored to blue between 5h and 6h30m positions, while its upper half was covered by soot. According to the designer and manufacturer of the engine (hereinafter referred to as "the engine manufacturer"), the shield turns to blue at 427° to 482°C and above, and

*21 "Shank" See photo 15.

*22 The stator is a static airfoil placed between the turbine blades of each stage of the engine; stators are fixed on the outside case of the engine and don't rotate.

*23 "Core cowl" means an aluminum alloy cover which envelopes the high-pressure part (core) of the engine, including the high-pressure compressors, combustors, HPTs and LPTs.

*24 The retaining ring is a circular hook with which to hold the fuel supply tube inside the fuel nozzle shroud.

*25 The fuel nozzle shroud can is a cover over the fuel nozzle supply tube to prevent fuel from leaking outside, should the fuel supply tube be separated.

*26 The heat shield is a titanium alloy installed in between the combustor and accessories to protect the accessories at the outside bottom of the engine case from high heat , in the occurrence of a fire.

soot adheres at surface temperatures of 370˚C and below.

② Engine case condition

From the thermal damages found on the right hand exterior surface of the engine case and on the core cowl, areas of sprayed and burned fuel leaked from the fractured fuel supply tube are indicated with white strings in Photo 11. And, the condition of the exterior of the engine case is shown in the table below.

| Engine case | Position | Surface condition (ex.Soot) | Damage (ex. to Wiring) | Remarks |
|---|---|---|---|---|
| Right-hand surface | Fuel supply tube fwd. upper portion | Heavily sooted | No damage | |
| | Fuel supply tube fwd. lower portion | Nearly not sooted | Partly damaged<br>No discontinuity of the wiring | |
| | Fuel supply tube aft portion | Heavily sooted | At lower aft portion, fire detector burned, fire detector isolator melted, no discontinuity of the wiring | Fire detector isolator melts at about 428˚C |
| Left-hand surface | General | Sooted | | |
| | Case lower portion | Heavily sooted | Burned, discontinuity of one EGT probe<br>No other wiring discontinuity | |
| Bottom surface | General | Heavily sooted<br>Traces of flames | | Traces looked like remnants of liquid |

(5) Core cowl

The right hand surface of the core cowl was melted and burned from 3h to 4h position, leaving a hole whose approximate size is 150 cm in length and 25 to 60 cm in width. The forward part of the core cowl at about the position of the stiffener ring was melted in a round shape at its top and bottom. The aluminum alloy outer panel of the core cowl melts at 642˚C. (See photo 10.)

The outer surface of the core cowl was heavily sooted around the hole on it.

The inner surface of the core cowl was entirely covered with dense soot, and there were traces of remnants of liquid similar to (4)②.

(6) Although the blowout door[*27] installed at 5h position of the core cowl was closed, the edge of the door was slightly deformed outwards.

(7) A number of very tiny metallic fragments were adhered to the engine MCD[*28] of the lubrication system. From their materials, those fragments were found to be from the engine sump seal and not from the bearings.

(8) There were many tiny holes in the lower surfaces of the outboard flap and of the

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[*27] The blowout door is fixed on the outside of the lower core cowl, and opens to reduce the difference in air pressure from outside when the pressure in the core cowl rises abnormally high, to protect the accessories, etc. mounted outside of the engine case. It is also known as the pressure relief door.

[*28] The engine MCD detects magnetic debris in the lubrication system inside the engine.

inboard aileron of the right wing, however, these holes did not affect their functions. (See Figures 4, 5, 6, 7, and 9, and photos 1 through 17.)

## 2.4 Damages to Items Other than the Aircraft

Fragments including engine blades were scattered on the runway (3,000 m long) at an approximate distance from 800 to 1,400 meters from its south end, and over the grass area to its east side. In addition, the grass area of around 100 $m^2$ was scorched to the east side of the runway and at a distance of 1,100 m from the south end of the runway, where most of the fragments including engine blades were scattered. The Kagoshima Airport was closed for about one hour to collect those fragments, the total weight of which was about 10 kg. (See Photo 4.)

## 2.5 Pilot Information

(1) PIC Male, Age 57 years

| | |
|---|---|
| Airline transport pilot certificate (airplane) | September 16, 1992 |
| Type rating for Boeing 767 | September 16, 1992 |
| 1st class aviation medical certificate | |
| Validity | Until February 8, 2006 |
| Total flight time | 15,440 h and 43 min |
| Flight time in the last 30 days | 58 h and 25 min |
| Flight time on the aircraft type | 7,853 h and 19 min |
| Flight time in the last 30 days | 58 h and 25 min |

(2) Co-pilot Male, Age 28 years

| | |
|---|---|
| Commercial pilot certificate (airplane) | December 19, 2002 |
| Type rating for Boeing 767 | November 11, 2004 |
| Instrument rating | December 27, 2002 |
| 1st class aviation medical certificate | |
| Validity | March 7, 2006 |
| Total flight time | 905 h and 00 min |
| Flight time in the last 30 days | 61 h and 08 min |
| Flight time on the aircraft type | 672 h and 07 min |
| Flight time in the last 30 days | 61 h and 08 min |

## 2.6 Aircraft Information

### 2.6.1 Aircraft

| | |
|---|---|
| Type | Boeing 767-300 |
| Serial number | 27617 |
| Date of manufacture | October 4, 1998 |
| Airworthiness certificate | To- 17-370 |
| Validity | October 29, 2006 |
| Total time in service | 20,791 h and 45 min |
| Time since the last periodical check (C maintenance on February 16, 2005) | 2,598 h and 17 min |

(See Figure 2.)

2.6.2 Engine

| | | |
|---|---|---|
| (1) | Right engine | |
| | Model | General Electric CF6-80C2B6F |
| | Serial Number | 704741 |
| | Date of manufacture | June 30, 1998 |
| | Total time in service | 18,419 h and 38 min |
| | Total cycles*29 | 13,429cycles |
| | Time since overhaul | 5,646 h and 17 min |
| | Cycles since overhaul | 4,126 cycles |
| (2) | Left engine | |
| | Model | General Electric CF6-80C2B6F |
| | Serial Number | 704759 |
| | Date of manufacture | August 13, 1998 |
| | Total time in service | 16,841 h and 00 min |
| | Total cycles | 12,252 cycles |
| | Time since overhaul | 6,048 h and 18 min |
| | Cycles since overhaul | 4,418 cycles |

2.6.3 History of Maintenance of the Right Engine

According to the company's maintenance records, history of maintenance since manufacture of the engine is shown as below.

| | |
|---|---|
| July 17, 1998: | Installed as left engine of JA767A |
| July 10, 2000: | Removed from JA767A |
| November 28, 2000: | Installed as right engine of the Aircraft (JA767B) |
| March 5, 2001: | Removed from the aircraft due to a bird strike |
| May 25, 2001: | Disassembled, repaired, re-assembled and tested at Maintenance Company D |
| July 6, 2001: | Installed as right engine of the Aircraft |
| November 7, 2003: | Removed from the Aircraft due to the damage of HPC*30. The lever arm of HPC 5th stage VSV*31 was fractured. The HPC 1st stage blades suffered possible some foreign object damage. Repaired at Maintenance Company D.<br>In accordance with the Service Bulletin (hereinafter referred to as "SB") 72-0910R4 issued in May 2001 to replace HPT 2nd stage blades with those of DSR142*32 material, all HPT 2nd stage blades were replaced with those of part number 1881M52G05. |

*29 In this report, a cycle is used to indicate the number of flights, where one cycle is from takeoff to landing. However, the cycle of the metallic fatigue means the number of load stresses applied on the metal.

*30 HPC stands for the high pressure compressor of the engine.

*31 VSV is a variable stator vane system for improving the engine efficiency. It shifts the angle of stator vanes according to the rotation speed of the LPC so as to shift the direction of flow-in air and optimize the blades for air flow.

*32 DSR142 stands for Directional Solidified Rene 142 material, which is a heat- resistant alloy material.

| | |
|---|---|
| | In accordance with SB73-0326, the clamps of the engine and the thickness of the fuel manifold were inspected. |
| March 17, 2004: | Installed as right engine of the Aircraft |
| November 13, 2005: | Conducted BSI to check the surface of HPT 1st and 2nd stages blades in accordance with the Company's maintenance manual. |

2.6.4 Record of Discrepancies

According to the Company's maintenance records, records concerning the vibration of the right engine for a year prior to this serious incident are as follows.

| | |
|---|---|
| February 10, 2005: | Adjusted the gap at the forward portion of fan cowl due to out of limit. |
| March 1, 2005: | The fan vibration level during climb (3.7 units). After the check, the vibration level was within the limit (4 units), and resumed flight. |
| March 2, 2005: | During climb the fan vibration level (3.7 units). Test-run on the ground, confirmed normal. |

2.6.5 History of Maintenance of Related Issues of the Right Engine HPT 2nd Stage Blades

On November 7, 2003, due to HPC damage, the engine was removed from the Aircraft. On January 4, 2004, at Maintenance Company D, a repair was made for the above mentioned damage, at the same time, all HPT 2nd stage blades were replaced with those of part number 1881M52G05. The installed blades are classified into the following two groups:

① Blades overhauled: 3 pieces

Manufactured by Blade manufacturer B, overhauled by Maintenance company E and stored at Maintenance company D.

| | Time in service | Cycles |
|---|---|---|
| Since New | 23,739 h | 7,536 cycles |
| Since Overhaul | 5,646 h | 4,126 cycles |

② Blades with minor repair on their tips : 71 pieces

Manufactured by Blade manufacturer A, coated by Coating company F, removed at the time of engine overhaul, minor repaired on the tips, and stored at the Maintenance Company D. Blade #58 is one of those 71 blades.

| | Time in service | Cycles |
|---|---|---|
| Since New | 11,513 h | 9,546 cycles |
| Since Repair | 5,646 h | 4,126 cycles |

2.6.6 EGT Comparison

The monthly averages of EGT during cruising for four months prior to this serious incident were compared with those in the same months of the previous year, and which resulted in no remarkable difference.

| | | | |
|---|---|---|---|
| August 2005 | 650°C | August 2004 | 650°C |
| September 2005 | 638°C | September 2004 | 648°C |

| | | | |
|---|---|---|---|
| October 2005 | 637°C | October 2004 | 643°C |
| November 2005 | 626°C | November 2004 | 629°C |

At the time of this serious incident, the Company was operating the type of the Aircraft with four roundtrips (eight flights) per day between Kagoshima and Tokyo.

2.6.7 Weight and Balance

At the time of the serious incident, the Aircraft weight was calculated at about 245,300 1b, and its center of gravity at a 22% mean aerodynamic chord (MAC), both of which are estimated to have been within the allowable limits (299,000 1b for the maximum takeoff weight; and between 7 and 37% MAC for the allowable center of gravity based on the calculated aircraft weight at the time of the serious incident).

2.6.8 Fuel and Lubricating Oil

The fuel was aviation fuel Jet A-1 and lubricating oil was BPTO2197.

According to the NTSB's document, the natural ignition point of the aviation fuel Jet A-1 is 225°C and its flash point[*33] is 40°C.

## 2.7 Meteorological Information

(1) The observation data in the aviation routine weather report at Kagoshima Airport at around the time of the serious incident was as follows:

16h00m Direction of wind: 270°; Velocity of wind: 8 kt; Prevailing visibility: 30 km;
Clouds Amount: 1/8, Type: stratus, Ceiling: 3,000 ft,
Temperature: 14°C; Dew point: 6°C;
Altimeter setting (QNH): 30.11 inHg

17h00m Direction of wind: 280°; Velocity of wind: 5 kt; Prevailing visibility: 30 km;
Clouds Amount: 1/8, Type: stratus, Ceiling: 3,000 ft,
Amount: 2/8 ~3/8, Type: unknown, Ceiling: unknown;
Temperature: 14°C; Dew point: 6°C;
Altimeter setting (QNH): 30.11 inHg

(2) Volcano eruption and ashes information

According to the Tokyo VAAC[*34],

the information on airspaces surrounding Japan was released 257 times in total in 2005. With regard to the volcano on Mt. Sakurajima, six eruptions were recorded in 2005, but no volcanic activities were recorded on the day of this serious incident.

(3) According to the Meteorological Agency, yellow sand dust were observed for a total of 43 days in Japan in 2005. During a one-month period prior to this serious incident, yellow sand dust were observed from November 7 to 9 in Kagoshima Prefecture.

## 2.8 Information on the DFDR, QAR and CVR

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[*33] A flash point is the lowest temperature at which a certain size of flame brought close to a heated fuel catches on fire by the vapor of the fuel.

[*34] VAAC is a Volcanic Ash Advisory Center designated by the ICAO that provides information when ash clouds or ashes from a volcano within the center's monitoring area affect or are predicted to affect the flight of aircrafts.

The Aircraft was equipped with a DFDR (P/N 980-4700-042) manufactured by Honeywell Inc. of the United States of America and a CVR (P/N S200-012-00) manufactured by L-3 Communications Corp. of the United States of America.

The DFDR, QAR, and CVR retained all the data recorded during the serious incident. Concerning the engine vibration level, the data from QAR were used because it had more records than the DFDR.

## 2.9 Information on Fire and Firefighting

According to Security and Disaster Prevention Division of the Kagoshima Airport Office (JCAB), the fire and firefighting information at the time of the serious incident is as follows.

| | |
|---|---|
| About 16h48m | Upon receiving the information of engine failure, three fire engines set out |
| About 16h52m | Found a fire on the grass area near taxiway T3. With permission from the tower, fire fighters entered the runway and put out the fire with water. Then, returned to the holding position near T3 and held for the landing of the Aircraft. |
| About 17h04m | When the Aircraft landed, 3 fire engines followed the Aircraft up to Spot 3. No fire or smoke from the airframe. |
| About 17h10m | Having confirmed all crew and passengers were safe, lifted alert. |

A firefighter who followed the Aircraft made the following statement about the fire on the grass area near the runway which was found at about 16:52:

> We did not receive any report about the fire in the grass area. When we arrived at the designated place to stand by for the emergency landing of the Aircraft, we found the fire and put it out.

The Air Traffic Control Operation Information Officer (hereinafter referred to as "Information Officer") , of the Kagoshima Airport Office (JCAB), who found the fire and went there, made the following statement about the fire:

> I found the fire from Operation Room, so I promptly left the Room and headed for the scene, and asked the firefighters nearby to extinguish the fire. It was already twilight and I did not notice fragments such as engine debris on the runway. At that time, we did not make any special inspection of the runway nor investigation on the cause of the fire. (See Photos 4 and 5.)

## 2.10 Fact-Finding Tests and Researches

### 2.10.1 Engine Disassembly Research

To investigate the cause of this serious incident, the right engine of the Aircraft was disassembled at the engine manufacturer's Overhaul / Repair shop in Prestwick, Scotland in the United Kingdom in January 2006, the findings of which are as follows.

(1) Fire detector inspection

The wirings of the fire detectors and overheat detectors installed in the engine designated fire zone were inspected for continuity, with the result of no discontinuity.

(2) Core cowl

The inner surface was entirely covered with soot. On the outer surface, soot was markedly adhered near the border (front end ) to the reverser cowl.

(See Figure 5 and Photos 10 and 11.)

(3) Result of disassembly inspection of the engine

① Fan

-No abnormality was observed with the fan.

-Abrasion was found on the labyrinth seal[*35] of #3 bearing.

② Compressor

-No abnormality was observed with the LPC[*36] and HPC.

-Grayish white deposits were observed on the CCV[*37] surface of almost all blades.

-Grayish white deposits were observed from the blade platform up to about 50 % of the HPC stator vane airfoil as well as the inside of the bleed air port.

-N2 shaft was seized and could not be rotated.

(See photos 30 and 32.)

③ CRF[*38] Hub and combustor section

-The inner surface of the CRF(Compressor Rear Frame) and the hub of the CRF were covered with grayish white deposits identical to those on the HPC.

-Small cracks were observed in the cooling holes of the inner liner of the combustor.

(See Photo 31.)

④ HPT

-On the CVX[*39] surface of 1st stage blades, grayish white deposits were observed from the platforms to the airfoils.

-Some of 1st and 2nd Stage nozzle guide vanes[*40] were covered with grayish white deposits and had damaged trailing edges.

-2nd Stage Blade #56 was fractured in its airfoil three inches from the platform.

-2nd Stage Blade #57 was fractured in its airfoil one inch from the platform (roughly half of blade height).

-2nd Stage Blade #58 was fractured in the shank about 1.3 inches(33mm) from its root, and its airfoil was completely missing. (The normal blade height: 5.5 inches(140mm).) Because uniform contact pattern was observed on the CCV and CVX surfaces of the dovetail, the blade is considered to be properly positioned within the disk.

-All of the 2nd stage blade airfoils and the inner surface of the engine case were partially damaged and generally covered with grayish white deposits.

(See Figure 14 and photos 15.)

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[*35] A labyrinth seal is a sealing method for preventing lubricating oil from leaking outside by means of the pressure difference created between the bearing with low-pressure and the high-pressure steam extraction from the compressor led to the outside.

[*36] LPC stands for a low-pressure compressor of the engine.

[*37] CCV in this report means the concave side of the blade.

[*38] CRF is a frame located at the rear end of the compressor, which is a case covering the combustor. The temperature of the CRF rises from the heat of the combustor while the engine is working.

[*39] CVX in this report means the convex side of the blade.

[*40] Nozzle guide vanes are the circularly aligned static wings installed in front of the HPT blades for rectifying the combustion gases in a manner in which they hit the turbine blades at the optimum angle.

⑤ LPT

- All stator vanes had damages or missing parts. The case had sustained damage on many parts when the fragments such as fractured blades, hit it. And, low pressure rotor system could not be rotated.
- The airfoils of 1st stage blades exhibited impact marks with contact from internal engine debris.
- The airfoils of 2nd stage blades were fractured at about 50% of their length on average.
- The airfoils of 3rd and 5th stage blades were fractured at various lengths.
- The airfoils of 4th stage blades were fractured at about 75% of their length from their roots on average.

(See photos 16, 17, 28, and 29.)

(4) Fuel leak position

There are 30 fuel supply tubes on the entire circle of the manifold and each tube is connected to a fuel nozzle. The fuel supply tube #6 at 2h30m position was fractured inside the shroud can. Furthermore, the retaining ring and the O-ring normally located within the shroud can were found dislocated from their proper positions, and fractured fuel supply tube was out of the the shroud can. The O-ring sustained heat damage.

The pressurized nitrogen gas was used to check if there were any leaks in the fuel system, but no leaks were found other than the area.

The thickness of the fractured fuel supply tube was found to be not constant.

(See Figure 6 and Photos 12, 13 and 14.)

(5) Engine vibration accelerometer inspection

The insulation of the engine vibration accelerometers installed near the fan and the HPT were checked, and no abnormality was observed. The ground wire of the fan accelerometer had discontinuity. Other functions were found in normal.

(6) Engine lubrication system inspection

Pressure test was conducted with oil return line, and no leaks were found.

2.10.2 Metallurgical Research of Engine Parts

In cooperation with NTSB, a metallurgical examination was made of the fractured turbine blades and fuel manifold supply tube.

(1) HPT 2nd stage blades

All remaining blades were cut off to investigate the occurrence of cracks in the internal cooling air passage[*41] (hereinafter referred to as “passage”), and fractured Blade #58 and the non-fractured Blades #10, #20, #25, #35, #49, #54, #56, and #73 manufactured by Blade manufacturer A, and non-fractured Blade #2 manufactured by Blade manufacturer B were investigated to find out the cause of the fracture. The results of the comparison are as follows.

The metal composition of all investigated blades were found in conformity with the DSR142 material standard.

①Occurrence of cracks in passage

10 out of the 74 blades had cracks. All cracks originated on the CVX

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[*41] Cooling air passage made inside the blade so that the temperature of the HPT blades doesn’t rise too high, and through which air from the compressor is passed to cool the inside of the blades, and creates a layer of cooling air on the surface of the blades by the air blown from small holes, so as to protect the HPT blades from the direct impact of the highly heated combustion gases.

surface of the passage (hereinafter, curvature of blade airfoil surface is used even to indicated the interior of the blades), and their locations were limited to the circular portion of the fillets (hereinafter referred to as "TA area", which connect the end of the rib (passage partition wall) at which the cooling air turns around (hereinafter referred to as "TA") with the surface of the passage (blade wall) with roundness. And, no cracks were found on the three blades of Blade manufacturer B.

(Units: locations)

| CVX | LE passage | Mid passage | TE passage |
|---|---|---|---|
| Blades with cracked coating | 1 | 10 | None |
| Blades with cracked base metal | 0 | 4 | None |

(LE: Leading Edge, TE: Trailing Edge)

② Blade #58 (Fractured)

The origin of the crack was found in TA area of the CVX side in the LE passage and was in broad region. Hot corrosion was not found in the origin region and the precise cause of the crack could not be determined. Details are given below.

- Immediately before the blade separation, a fatigue crack reached about up to 50% of the cross section of the shank.
- The thickness of the blade coating[*42] in TA area on the CVX side in the LE passage was 0.0017 in(0.0432mm) at most, which was thinner than those of other examined blades (0.0021 in~0.0033in(0.0533mm~0.0838mm).

| | |
|---|---|
| Specification of passage coating thickness | 0.0005~0.0030in(0.0127mm~0.0762mm) |

- Hot corrosion blisters were formed on the passage surfaces.
- No evidence was found to show significant hot corrosion penetration into base metal in the origin region.
- The depths of hot corrosion blisters were confined to the coating additive layer.
- The hot corrosion blisters were analyzed with EDS[*43], which resulted that existence of sulfur (S) was recognized near the surface and in the blisters.
- The crack origin region was broad and multiple origins consistent with crack initiation occurring from a prior coating crack in the TA area.
- The fracture surface was consistent with low-cycle fatigue.
- The thickness of the base metal of the LE passage on the CVX side in the examined blade shank was a little thinner (0.010in(0.254mm)) than other blades. According to the Engine manufacturer, such thickness variation can exist and be related to a minor shift of core[*44] die during casting.
- The wall of the blade shank which includes the origin of the fracture was cut

[*42] Coating is a thin film coated on the surface of the base metal for protecting it from heat or oxidation to keep it from deterioration.

[*43] EDS is an energy-dispersive X-ray spectroscopy for analyzing material compositions using X rays.

[*44] The core is the matrix used for punching holes in the cast.

and polished for observation, the result of which showed that the fractured surface was heavily oxidized with significant alloy depletion[*45], indicating that the fractured surface had been exposed to engine operation for an appreciable length of time, but the DSR142 gamma prime[*46] associated with high temperature strength still remained.

-The wall thickness of the LE passage in the shank was 0.075 in(1.905mm), thinner than the nominal[*47] model definition. The thickness of Blade #58 was compared to Blade #54 (normal) at the same position, and found thinner on the CVX side and thicker on the CCV side.

| Thickness of the shank of master blade : specification nominal minimum value | |
|---|---|
| TA area of LE passage on CVX side | 0.079 in (2.01mm) |
| TA area of Mid passage on CVX side | 0.070 in (1.78mm) |

(See Figure 14 and Photos 18 through 23.)

③ Blade #49 (Sample of crack and sulfur in the base metal)

Blade #49, which had a crack in the Mid passage, was cut to analyze the cracked surface.

-Depth of the crack was 0.0055 in(0.1397mm), of which 0.0035 in(0.0889mm) was in the coating and 0.002 in(0.0508mm) was in the base metal.

-The cracked surface was entirely oxidized, and especially in the coating layer it was severely oxidized.

-By the EDS analysis, sulfur was detected in the deepest part of the crack of the base metal.

(See Photos 26 and 27.)

④ Blade #20 (Sample of crack in the base metal)

Blade #20, which had a crack in the Mid-passage, was cut and polished for analysis.

-Depth of the crack in TA area of Mid passage on the CVX side was 0.00529 in in total. The crack propagated through the coating (thickness 0.00329 in(0.0836mm)) up to the depth of 0.002 in(0.0508mm) of the base metal. The crack in the rib (passage partition wall) propagated through the coating thickness (0.0034 in(0.0864mm)) up to the depth of 0.0006 in(0.0152mm) of the base metal, and had shallow distress on its surface.

(See Photo 25.)

⑤ Blade #25 (sample of crack in the base metal)

Blade #25, which had a crack in the Mid- passage and was manufactured by Blade manufacturer A, was cut and polished to check the presence of cracks and for analysis.

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[*45] Alloy depletion occurs when aluminum, one of the compositions of alloy, reacts with oxygen after being exposed to high temperatures without protective coating. Depleted aluminum loses DSR142 gamma prime, further reducing the high-temperature resistance.

[*46] Gamma prime is a state in which the gamma phase, the mother base of the alloy in the atomic level, becomes like a rectangular net and less susceptible to being broken.

[*47] Nominal model definition is the standard values of blades, such as length, that the engine maker specifies to the blade maker.

| Location | Coating thickness | Feature |
|---|---|---|
| TA area of LE passage | 0.0028in (0.0711mm) | A slight pitting (dent) in the coating layer only |
| TA area of Mid passage | 0.0024 in (0.0610mm) | Cracks of maximum 0.0004 in(0.0102mm) into the base metal |

⑥ Comparison of radii of rising part of TA area by blade manufacturers

The difference of the radii of rising part of TA area (hereinafter referred to as “TA radius”) as shown from A to B in Photo 24, and occurrence of crack were examined.

-The TA radius of LE passage on the CVX side of the blades made by Blade manufacturer A was smaller than the nominal model definition.

-The TA radius of LE passage on the CVX side of the blades made by Blade manufacturer B was larger than the nominal model definition.

| Location | Nominal model definition of A | Blade manufacturer A | Blade manufacturer B |
|---|---|---|---|
| LE passage on CVX side | 0.042 in(1.067mm) | 0.033in ~ 0.040in (0.838mm~1.016mm) | 0.183 in. (4.648mm) |

(See Photo 24.)

(2) HPC and HPT blades

The grayish white deposits were found all over the surfaces of the HPC and HPT blades. The deposits were also found in the small ports of the passages of the CRF, blades and nozzle guide vanes.

(See Photos 28 through 32.)

(3) EDS analysis of the deposits found on the blades and the volcanic ashes from Mt. Sakurajima.

① The white deposits on the HPC 10th stage blades were mostly sodium sulfate ($Na_2S0_4$), and as a result of composition analysis, mainly contained sulfur (S), sodium (N), potassium (K) and oxygen ($0_2$).

② The sample of deposits collected from the surface of the HPT blade shanks contained, as a result of composition analysis, oxygen, silicon (Si), sodium, sulfur, calcium(Ca) and nickel (Ni), which compose quartz, albite, and plaster, and did not thoroughly match the sample of ①.

③ The sample of volcanic ashes from Mt. Sakurajima contained, as a result of composition analysis, mostly sodium, silicon and calcium, and only partially matched with the sample of ①.

According to the engine manufacturer, although sodium sulfate was not a factor which caused this serious incident, it can possibly induce hot corrosion of the engine turbine section. (See Photo 32.)

(4) Hot corrosion

The engine manufacturer classifies hot corrosion of turbine blades into the following two types:

*(1) Type 1 hot corrosion: It occurs over 1,550-1,850°F (843° to 1,010°C) and*

*corrosion rate is dependent of temperature but not of sulfur level.*
*(2) Type 2 hot corrosion: It occurs over 1,100-1,350°F (593° to 843°C) and corrosion rate is dependent of temperature and sulfur level.*
(See Figure 15.)

(5) Examination of fractured fuel supply tube (pigtail) # 6

The tube was fractured and dislodged at the bead weld*48, where the wall thickness of the tube was about 1.45 mm at its thickest and about 0.55 mm at its thinnest.

The other fuel supply tubes were cut off to observe their wall thickness, which was found to be uniform on the entire circumference at about 0.8 mm.

According to the engine manufacturer, the cracks in the fractured fuel supply tube originated from several points on the outer diameter of bead weld where the wall was the thickest. Through the investigation, it was found that the tube was fractured because the high-stress load of tension and compression was repeatedly applied in certain amplitude. The crack originated from the outer surface of the tube propagated to the inner surface under about 2,500 times repetitive load. The repetitive load caused by the HPT imbalance is calculated to induce vibration with acceleration about 45G. Moreover, the retaining ring and the O-ring attached at the end of the shroud can were dislocated from their proper positions, which caused the fuel supply tube to be dislodged from the shroud can.

(See Photos 12, 13, and 14.)

2.10.3 Researches of Manufacture of Blades and Fuel Supply Tube (pigtail)

In cooperation with the NTSB, ARAIC made the investigation on the design and manufacture of the engine blades and the fuel supply tube.

(1) Design and manufacture of turbine blades

Although the engine manufacturer defines the blades such as exterior shape, it is the blade manufacturer to design interior detail of the blade such as the TA area, and the engine manufacturer approves it. Concerning the TA radii of the blades used at the time of this serious incident, Blade manufacturer A was using smaller value than Blade manufacturer B, but the engine manufacturer had approved both values.

Since 2001, blade manufacturers have started to utilize the eCA&V*49 system to control product standards by inspecting each step from die manufacturing to completion of product in order to maintain quality at the stages of design and manufacturing. Especially concerning the exterior shape of blade, 1,177 referencing points are specified to link the casting die and the finished blade, and based on these measurements shape of a blade is confirmed to be in compliance with the standard. In addition, during inspection of the finished blades, the passage standard is confirmed by means of NDI*50 such as the microscope, X ray, or FPI*51.

(See Photo 33.)

(2) Rate and position of coating crack occurrences by blade manufacturers and coating

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*48 Weld bead is the swollen wall-thickness made when a tube is welded to be connected.

*49 eCA&V, an e-Characteristic Accountability & Verification, is a web-based system led by the engine makers and used by most suppliers. It refers to the information management system for quality assurance throughout the entire manufacturing process of blades, and so on.

*50 NDI stands for non-destructive inspection.

*51 FPI stands for fluorescent permeation inspection, a method of detecting a crack by permeating a fluorescent penetrate in a crack and applying the ultraviolet rays to cause the crack to emit light.

suppliers

After this serious incident, the engine manufacturer cut off 938 blades of HPT 2nd stage, all of which had been discarded in Japan by reasons of expired life limit and the like, and returned to the manufacturer, and conducted a research on cracks in their passages as follows.

① Rate of coating crack by Blade manufacturers

Cracks were found in the coating of the passage surfaces of 28 blades. One of the cracks propagated 0.002 inches into the base metal, whose blade was used about 9,000 cycles since new.

The Rate of crack occurrence by blade manufacturers is given below.

| Blade manufacturer | Rate of crack occurrence | Remarks |
|---|---|---|
| Blade manufacturer A | 3.3% | Manufactured in and after 1999 |
| Blade manufacturer B | 2.2% | Manufactured in and before 1999 |

The Rate of crack occurrence by blade manufacturers and coating suppliers is given below.

| manufacturer (period) Number of blades / Coating supplier | Blade manufacturer B (1997 to1999) 45,000 blades | Blade manufacturer A (1999 to 2006) 300,000 blades | Total |
|---|---|---|---|
| Supplier F | Inspected: 123<br>Cracked: 6<br>Rate: 4.9% | Inspected: 369<br>Cracked: 21<br>Rate: 5.7% | Inspected: 492<br>Cracked: 27<br>Rate:5.5% |
| Supplier G | Inspected: 146<br>Cracked: 0<br>Rate: 0.0% | Inspected: 300<br>Cracked: 1<br>Rate: 0.3% | Inspected: 446<br>Cracked: 1<br>Rate:0.2% |

The rate of crack occurrence for the total number of inspected blades was 3.0%, and rate of cracks which penetrated into the base metal was 0.1%.

Blade #58, fractured in this serious incident, was made by Blade manufacturer A and Coating supplier F.

② The position of cracks by blade manufacturers

The blades made by Blade manufacturer A had more cracks on the CVX side of the passage.

The blades made by Blade manufacturer B had more cracks on the CCV side of the passage.

(Unit: Number of blades)

| Region / Blade manufacturer | CVX | | | CCV | | | Total |
|---|---|---|---|---|---|---|---|
| | LE passage | Mid passage | TE passage | LE passage | Mid passage | TE passage | |
| manufacturer A | 16 | 4 | 1 | 0 | 1 | 0 | 22 |

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| manufacturer B | 2 | 0 | 0 | 1 | 1 | 2 | 6 |

(3) Stress in the TA area by computer simulation

Using computer simulation, the engine manufacturer calculated stress in the TA areas of blades made by Blade manufacturers B and A, in comparison with the stress and temperature of LE passage of a blade made by Blade manufacturer A. As a result, it became clear that there is less stress on the blades made by Blade manufacturer B than that made by Blade manufacturer A. The engine manufacturer suggests that the result had some correlations with the cracks in the blades.

(CVX side)

| Region | LE passage | | Mid passage | | TE passage | |
|---|---|---|---|---|---|---|
| manufacturer | A | B | A | B | A | B |
| Relative stress (reference= 1) | 1 | 0.81 | 1.39 | 1.02 | 1.10 | 0.88 |
| Temperature difference from reference | ±0° F | ±0° F | +53° F | +53° F | +93° F | +93° F |
| Rate of cracked parts | 2.2% | 0.9% | 0.3% | 0% | 0.2% | 0% |

(See Figure 13.)

(4) Measures against fuel leakage from the fuel supply tube

As described in 2.11.7 (2), an incident of a fuel supply tube failure similar to this serious incident occurred in October 2003. The engine manufacturer took the following measures to prevent fuel leakage caused by such as the fuel supply tube fracture due to engine vibrations. The operators were notified of all of these measures through Service Bulletins.

① Fuel Nozzle to Fuel Manifold - - New Snap Rings (SB73-0337)

Issued in November 2004, its content includes replacement of retaining rings with snap rings in order to increase the capacity of retaining the tube at the end of the shroud can, at the occasion of engine shop visit after November 30, 2004. The incident engine had not yet reached the time.

② Improvement of fuel manifold clamp (SB73-0326)

Issued in May 2003, its content includes replacement of material, from synthetic fiber to poly tetra fluoride ethylene, of the cushion of the P clamps supporting fuel manifold in order to alleviate the engine vibrations transmitted to the fuel supply tubes. The incident engine was replaced it in November 2003. (See Figures 12 and 16.)

## 2.11 Other Information

### 2.11.1 Description of CF6-80C2 Engine

(1) Summary

A part of the air flowing in the fan, which goes to LPC (then to HPC) is called primary air. The other part of air which flows through the fan mainly to generate thrust is called fan air. A part of the fan air is used for HPTC[*52] and LPTC[*53]. The fan, LPC and LPT

[*52] HPTC means to cool the high-pressure turbine case.

[*53] LPTC means to cool the low-pressure turbine case.

rotate on the identical shaft, and this rotation speed is referred to as N1, whereas the HPC and HPT rotate on the other identical shaft and that rotation speed is referred to as N2.

(2) FADEC functions

The primary function of FADEC*[54] is to control the fuel flow to get N2 necessary for generating N1 which corresponds to the TLA set by the pilot as well as to keep the EGT within its limit. That is, if N1 drops below the value set by TLA, the PS3, which is the HPC discharge pressure (hereinafter referred to as "CDP"), also decrease. The FADEC, upon detecting the decrease in PS3, increases the fuel flow to increase N2 so that N1 rotates as specified by the TLA. However, like a case of this serious incident, when PS3 rapidly drops despite that FADEC controls to increase N2 in response to decrease in N1 detected by PS3, the FADEC decreases the fuel flow to avoid the engine from stopping.

(3) Heat control measures of engine blades

In order to alleviate the high temperature on the HPT blades which are exposed to combustion gas, a part of the CDP air flows in the passage in the blades to cool the blades. The CDP air flows from the bottom of the blades along the internal cooling passage, used for convection cooling*[55] and film cooling*[56].

(4) Cooling air flow in the core cowl

Because the temperature inside the core cowl rises high due to the heat released from the engine case and accessories such as a generator, a part of fan air is introduced by a duct to make an air flow inside the core cowl (hereinafter referred to as "ventilation flow"). Cooling of HPC case and accessories are controlled by CCCV*[57] (20psig*[58]). Cooling of HPT case is controlled by the FADEC using HPTC valve (14.7 to 29.7psig), it is called the ACCS*[59]. Cooling of LTP case is made at all times by fan air blown onto it. The ventilation flow combined with these cooling air is discharged from the air vent between the exhaust nozzle and the aft end of core cowl in the aft part of the engine. The speed of the ventilation flow is normally about 6 to 12 kt according to the engine manufacturer, but the flow speed decreases as N1 decreases.

(5) Engine fire-control measures

① Fire wall

A fire wall is installed in the engine pylon to prevent fire from spreading to wing fuel tanks in case of engine fire. The compressor, combustor, turbine and accessory areas of an engine are the designated fire zone, and constructed with fire resistant materials. The heat shield installed at the lower forward part of

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*[54] FADEC stands for Full Authority Digital Engine Control. Engine is thoroughly controlled by an exclusive computer installed in the engine case, and a pilot inputs the necessary N1 thrust as TLA using the thrust lever.

*[55] Convection cooling is a method of cooling blades by cooling air convectively flowing inside the blade wings.

*[56] Film cooling is to blow cooling air out of the small holes onto the blade surface to cover and cool the surface with the film of the cooling air.

*[57] CCCV is a valve for controlling the cooling air for the engine core.

*[58] Psig is a gauge pressure, and means the difference from the air pressure (14.7psi) at the sea surface.

*[59] ACCS refers to a system for improving turbine efficiency at cruising, and works as follows: cooling air is sent into the air manifold attached to the outer circumference of the turbine case so that the cooling air blown to the outer surface of the turbine case compresses the case, so that the gap between the end of the blades and the case is correctly narrowed. It normally works at altitude approximately 20,000 ft or higher.

the engine case protects lines of fuel, oil and hydraulic systems from the heat released from the combustor and high pressure turbine and forms of the upper part of the fire zone.

② Fire detector

The table below shows the location of four sensor tubes installed in the designated fire zone of engine for the prompt detection of fire and overheating of the engines of the Aircraft.

| Location | | Function | Remarks |
|---|---|---|---|
| FWD | Upper | Overheat | Engine pylon |
| | Lower | Fire | Lower part of HPC |
| AFT | Upper | Fire | Upper part of HPT |
| | Lower | Fire | Lower part of HPT |

According to the data of the aircraft designer and manufacturer (hereinafter referred to as "the aircraft manufacturer"), each sensor has a certain length of tube in which hydrogen and helium are sealed. As a fire detector, when the entire tube is heated to about 426˚C or higher, the helium gas expands and activates the pressure switch at the end of the tube. In addition, in the center of the tube, there is another thin tube, and when heated locally to about 635˚C or higher, the sealed hydrogen gas expands, and activates the pressure switch.

According to the aircraft manufacturer, fire detector is not installed between 3h and 4h positions in the aft part of engine, because there are cooling air ducts for the HPTC and LPTC there, not allowing enough space for fire detector. The engine is equipped with two fire-extinguishing (halon) bottles in case of a fire. The temperature inside the core cowl sometimes rises up to about 300˚C while the engine is running.

(6) EGT sensing system

Eight almel-chrome thermocouple probes are installed in the gas path near the LPT 2nd stage of the engine to measure the flow path gas temperature between the HPT and LPT.

(7) Engine vibration sensing system

The vibration levels of the fan, LPT and N2 are calculated from outputs of the accelerometers attached near the fan and HPT, and the N1 and N2 data. Of these levels, the highest is automatically displayed with its location on the EICAS. When the N1 and N2 data are not available, the vibration levels sensed by the fan and HPT accelerometers are displayed. The vibration levels are indicated as zero when N1 is less than 20% or N2 is less than 40%.

The vibration level is indicated in the range of zero to five units. The maintenance manual of the Company mandates an inspection in case the vibration level becomes 4 units or higher.

(8) Fuel pressurization

The fuel sent from the fuel tank is pressurized to about 1,000psi[*60] by the main fuel pump attached to the accessory gear box of the engine. From the main pump it is then sent to the HMU for metering to the fuel manifold.

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[*60] Based on *the Investigation Summary/Closure Report* written by the Engine Manufacturer, October 31, 2006.

(See Figures 4, 7, 8, 10 and 17.)

2.11.2 Engine Washing Standard

(1) CESM[*61] of engine manufacturer

According to the CESM No.18 issued on October 28, 1992, by the engine manufacturer, dirt, oil and salt are washed off by washing with water or coke. The average EGT drops by about 10.1˚C when washed with water, and by about 9.9˚C with coke, improving the engine reliability and fuel efficiency.

The engine manufacturer recommends periodical washing to extend engine life. However, its implementation is at the discretion of individual operator.

Regaining the EGT margin reduces the engine operating temperature and thus improves hot section life.

(2) Maintenance manual of the Company

The Company had no standard of time in service or degree of contamination to indicate the time of water washing, nor did not possess the washing facilities. In response to this serious incident, the Company established the standard for washing, and is preparing its implementation.

2.11.3 HPT Blade Maintenance Standard

*The WorkScope Planning Guide* provided by the engine manufacturer stipulates the HPT blade maintenance as follows, and the Company was conducting maintenance according to the Guide.

*-Perform an on-condition[*62] check at every 400 cycles.*

*-Blades have no service time limit.*

The Company's maintenance manual, approved by Japan Civil Aviation Bureau, does not stipulate blade life limit.

2.11.4 Regulations on the Fire Detector

With regard to engine fire detection, Annex 1 to Article 14 of the Civil Aeronautics Regulations of Japan corresponds to detailed airworthiness code chapterⅢ "Transport category airplane".

5-9-12-1 (FAR25.1203(a))

*There must be approved, quick acting fire or overheat detectors in each designated fire zones, and in the combustion, turbine and tailpipes of the turbine engine installations, in numbers and locations ensuring prompt detection of fire in those zones.*

2.11.5 Descriptions in AOM Concerning Engine Failure and the Like

The AOM procedures of the Company concerning engine failure and the like, describe as follows:

(1) Engine Failure After V1 Procedure (partly omitted)

A crewmember who finds engine fire (severe damage or separation) or engine failure shall call out "Engine Fire" or "Engine Failure."

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[*61] CESM stands for Commercial Engine Service Memorandum issued by engine makers. It carries useful articles on operations and maintenance and distributed to companies using the engines by the makers.

[*62] On condition means to continuously monitor the conditions of aircraft equipments by visual observation, measurement, testing or by other means without disassembling or overhauling the aircraft, to maintain the airworthiness.

| *PF* | *PNF* |
|---|---|
| *Continue flight* | |
| *Monitor air speed, and rotate smoothly at $V_R$* | *Call out "$V_R$,".*<br>*Call out"$V_2$", and "Positive (Climb)".* |
| *After verifying positive rate of climb, call out "Gear Up".* | *Set the* ***Landing Gear Lever*** *UP*<br><br>*When GEAR Light and DOORS Lights extinguish, set the* ***Landing Gear Lever*** *OFF* |
| *Climb at $V_2$ to $V_2$ +15.*<br>***The maximum bank angle is***<br>*30° when speed is at or above $V_2$+10, and 15° when below $V_2$ +10.* | *Monitor engine and flight instruments* |
| *When the desired altitude is achieved,** accomplish the memory items of the CHECKLIST*<br>*(Omitted hereinafter)* | |

***: at or above obstacle clearance altitude or 400 ft AFE, which is higher.*

(2) ENGINE FIRE OR SEVERE DAMAGE (partly omitted, tentative translation))
(Items in boxes must be reconfirmed by reading them from the checklist after crewmembers have conducted a quick-set up. The same procedure should adopt for items in (4).)

*Condition: fire is detected in the affected engine, or airframe vibrations detected with abnormal engine indications, or the engine has separated.*

*AUTOTHROTTLE ARM SWITCH ........................ OFF PF*
*[Autothrottle use not recommended under engine inoperative conditions.]*
*THRUST LEVER........................................ CLOSE PF*
*[Assist in recognition of affected engine.]*
*FUEL CONTROL SWITCH....................... CUT OFF PNF*
*ENGINE FIRE SWITCH.............................. PULL PNF*
*If engine fire warning light remains illuminated:*
*ENGINE FIRE SWITCH....................... ....ROTATE PNF*
*(Omitted hereinafter)*

(3) ENGINE FAILURE OR SHUTDOWN

*Condition: Engine has failed, engine has flamed out, or when prescribed by another non-normal procedure.*
*AUTOTHROTTLE ARM SWITCH........................OFF PF*
*[Autothrottle use not recommended under engine inoperative condition.]*
*THRUST LEVER..............................................CLOSE PF*
*[Assist in recognition of affected engine.]*
*Engine condition permitting, operate at idle for two minutes to allow engine to cool*

*and stabilize.*
*FUEL CONTROL SWITCH...............................CUT OFF PNF*
*APU[*63] SELECTOR*
*(If APU available) ............ ............START, THEN ON PNF*
*[Provide an additional source of electrical power.]*
(Omitted.)
*Plan to land at the nearest suitable airport.*
(Omitted hereinafter.)

(4) ENGINE LIMIT OR SURGE OR STALL (the items in boxes are conducted by memory)

*Condition: Engine EGT or RPM are abnormal or are approaching or exceeding limits, abnormal engine noise are heard, or there is no response to thrust lever movement.*

| |
|---|
| *AUTOTHROTTLE ARM SWITCH............................ OFF PF* |
| *[Allows thrust lever to remain where manually positioned]* |
| THRUST LEVER .............................................. RETARD PF |
| [Helps identify the event engine.] |
| Retard until indications remain within normal limits or the thrust lever is closed. |

*If indications abnormal or EGT continues to increase:*
*Accomplish ENGINE FAILURE OR SHUTDOWN Checklist.*
*If the engine indications appear normal after shutdown, the engine may be restarted.*
*If indications stabilized and EGT stabilized or decreasing:*
*THRUST LEVER ..........................................ADVANCE PF*
*Advance slowly. Check that RPM and EGT follow thrust lever movement.*
(Omitted hereinafter)

(5) The Company reference procedure set for the flight procedure at the time of engine failure at Kagoshima Airport (part)

*For a takeoff from Runway 34*
*Climb via RWY track to KGE(Kajiki VOR/DME) 2.7D, then turn RIGHT and intercept KGE R141 to the Pacific Ocean.*
(See Figure 18.)

2.11.6 Information on Airport Management

Based on the ATC communication records, the Flight Operation Records and the Daily Business Records kept by the Kagoshima Airport Office(JCAB), the runway usage from the takeoff to the landing of the aircraft, was outlined mostly as follows. Situations of the two commuter airplanes that took off after the aircraft are indicated with asterisk ∗.

16h44m58s The Aircraft started take off roll on Runway 34.
∗16h45m07s The Tower gave a clearance to a commuter airplane to lineup on

[*63] APU stands for the Auxiliary Power Unit installed separately from the propulsion engine to supply the aircraft with pneumatic pressure, oil pressure and electricity.

| | |
|---|---|
| | Runway 34. |
| 16h45m19s | The Aircraft took off. |
| 16h46m30s | The Radar understood the report of engine failure of the Aircraft. |
| *16h48m19s | The Tower inquired to the commuter airplane, "There's some smoke in the grass area. It seems like a fire. Any problem to you?" (in Japanese language) The commuter airplane answered to the Tower "I think no problem to take off." (in Japanese language) |
| *16h48m40s | The Tower gave the commuter airplane a clearance to take off. Then the commuter airplane took off. |
| 16h52m00s | The Aircraft reported the Radar to return to Kagoshima Airport. At around this time, fire engines put out the fire in the grass area alongside the Runway. The Information Officer arrived at the fire area. |
| 16h53m04s | The Aircraft set the transponder to emergency code. |
| *16h58m53s | Another commuter airplane received a clearance to take off from Runway 34, then took off. There was no report from the two commuter airplanes about the fragments on the Runway. (Under the direction and velocity of the wind at the time of this serious incident, the ground-run distance necessary for the two commuter airplanes to take off is about 600 to 700 meters. The two took off commuter airplanes were inspected just after their flights, and no damages to the tires and the like, were confirmed.) |
| 17h04m01s | The Aircraft landed on Runway 34.<br>As a result of an unscheduled check of the runway conducted by Information Officers found that engine fragments were scattered along the right side of the Runway 34 between 800 and 1,400 meters from its south end. |
| About 17h22m | The Runway was closed to collect the fragments. |
| About 18h20m | Runway was re-opened. |

2.11.7 Similar Events to This Serious Incident

(1) Cases of HPT 2nd stage blade separation of the same type of engine

This serious incident is the first case in which HPT 2nd stage blade made of DSR142 material mentioned in 2.10.2 (1) was fractured during flight. SB72-0982R02 issued on July 5, 2002, recommended to replace HPT 2nd stage blades with those of DSR142 materials, principal operators in Japan completed the replacement.

Other cases, prior to this serious incident, in which HPT blades made of Rene 80 material and not of DSR142 material, are as follows:

① There were 18 cases in which the HPT 2nd stage blades were fractured at a length of 10 to 20% from the root of blade airfoil, where;
  1) In almost all cases, experienced high vibrations (maximum 5 units)
  2) IFSD: 5 times, turn back after takeoff: 3 times, takeoff abort : 3 times
  3) Puncture of LPT case; 8 cases
  4) Damage to aircraft: 3 cases

② There were 15 cases in which the HPT 2nd stage blades were fractured at a length of 60 to 85% from the root of blade airfoil where;

1) In several cases, high vibrations

2) In one case, puncture of LPT case and damages to engine cowl and aircraft

(2) Fuel leakage from the fuel supply tube

A similar case to this serious incident

In October 2003, three blades of HPT 1st stage of a Boeing 747-400 freighter were fractured below platform, which generated engine vibration, and caused fracture of two fuel supply tubes, then resulted in a fire inside the cowl. Cause of the fracture was hot corrosion in the HPT 1st stage blades, after ingestion of a large amount of sand into the engine.

2.11.8 Opinion of the Engine Manufacturer and the Aircraft Manufacturer to this Serious Incident

In this serious incident, the core cowl was overheated causing material distress and loss of some cowling skin. The fire detector response to this fire was not immediate, but was sufficient to control the spread of the fire. The engine manufacturer and the aircraft manufacturer regard that it was not a threat to the safety neither the fire duration nor the damage to the engine cowling was a threat to airplane safety. Also in the Federal Aviation Administration (FAA) of the United States of America opinion the quick response certification standard was not met, no FAA mandated corrective action is warranted.

# 3. ANALYSIS

## 3.1 Qualification of the Flight Crewmembers

The PIC and the co-pilot held adequate airman certificates and valid airman medical certificates.

## 3.2 Airworthiness of the Aircraft

The Aircraft had valid certificate of airworthiness and had been maintained and inspected in accordance with applicable regulations.

## 3.3 Contribution of Weather

When this serious incident occurred, as described in 2.7, because there issued no NOTAM or information on volcanic ashes which could damage the engines of the Aircraft, nor did the crewmembers report any odor of volcanic fumes, it is estimated that a large quantity of volcanic ashes or yellow sands for example, were not ingested into the engines in a short time. It is considered that the grayish white deposits on the HPC blades and so on were caused by a long period of flying in the air, which contained a very small amount of volcanic ashes or yellow sands. Those deposits contained sulfur and sodium, as mentioned in 3.4 (4).

## 3.4 Fractured Blades

(1) First separated blade

As described in 2.10.1 (3)④, the three separated HPT 2nd stage blades were #58, #57, and #56, in the order from the least remaining portion. As described in 2.10.2 (1) ②, because Blade #58 exhibited characteristics consistent with low-cycle fatigue, it is thought that Blade #58 was the first separated one.

(2) The time at which Bade #58 separated

Concerning the time at which the blade was separated, on the basis of the following, it is estimated that when the Aircraft was rotated and airborne, it was about 16h45m19s.

① As described in 2.1.1 (1), one second after the AIR/GND sensor of the Aircraft's main landing gears turned to AIR, the right engine EGT increased to about 926℃ while N1 and N2 dropped remarkably.

② As described in 2.1.2 (1), the PIC stated, "At the time of airborne during take off from RWY34, a high-amplitude low-frequency airframe vibration started."

③ As described in 2.1.2 (5) and (6), passenger 1 said, "At takeoff, I saw a burst of many thin flames spew from the engine for a moment." And passenger 2 said, "The Aircraft kept rattling at takeoff."

④ As described in 2.4, fragments of blades and the like, were scattered around the runway from which the Aircraft took off and the grass area on the east side of the runway (right side).

(3) Cause of fracture

As described in 2.11.7 (1), this serious incident was the first case in which the HPT 2nd stage blade made of DSR142 material was fractured. The fractured surface was perpendicular to the grain boundary of DSR142.

According to the result of the metallurgical analysis conducted by the engine manufacturer in the presence of the NTSB investigators, as described in 2.10.2 (1) ②, although it was not able to identify the cause of origin of Blade #58 fracture precisely, because the fracture surface exhibited characteristics consistent with low-cycle fatigue, it is considered possible that the following factors were combined to cause the fracture.

① TA Radius

The fracture origin of Blade #58 was found in its TA area. As described in 2.10.2 (1)⑥, because the TA radius of Blade #49, which was made by the same Blade manufacturer as the separated blade and had a crack in the passage, was smaller than the nominal model definition, it is estimated that the TA area had a high stress concentration. Meanwhile, because the TA radius of Blade #2, which was made by another Blade manufacturer and had no cracks, was larger than the nominal model definition of Blade manufacturer A, it is considered that it had relatively low stress concentration. These were verified by the computer simulation conducted by the engine manufacturer as described in 2.10.3(3).

Because it is estimated that Blade #58 had a TA radius smaller than the nominal model definition, it is considered possible that the crack was caused by the stress concentration similar to Blade #49, and resulted in the fracture.

② Coating thickness and presence of sulfur

Blade #58 was made by Blade manufacturer A and its coating was made by supplier F, whose coating had higher rate of cracks as described in 2.10.3(2). Moreover, the coating layer of the TA area exhibited a trace of hot corrosion blisters, which was analyzed with EDS and found to have contained sulfur.

Also, as described in 2.10.2 (1)③, HPT 2nd stage Blade #49 had a crack in the passage TA area propagating through the coating into the base metal, and sulfur was detected with EDS. The engine manufacturer assumes that the presence of sulfur in the crack indicates that the hot corrosion was not of type Ⅰ but of type Ⅱ, as described in 2.10.2(4).

Sulfur, even just a meager quantity, generates metal sulfide. Adhered sulfur can degrade the protective coating and underlying metal at elevated temperature. Once the sulfide corrosion starts, corrosion continue even additional sulfur is not supplied from outside.

The expert advisor for this serious incident expressed his opinion as follows.

"Generally speaking the tendency of fatigue-crack generation depends on the surface conditions. Under the same stress conditions, with or without a blister, it is considered that the chances are high where the number of repeated blade use differs before a crack is generated when a crack initiation site is given. Therefore the possibilities cannot be denied that the hot corrosion and its affiliated oxide expedited the generation of a crack." (See photo number 22.)

③ Thickness of shank wall

The shank wall of Blade #58 was, as described in 2.10.2 (1)②, thicker than the nominal model definition in its Mid passage but thinner than the nominal model definition in the LE passage on the CVX side, and when compared to other blades, thinner on the CVX side (0.010in) and thicker on the CCV side. The engine manufacturer contended from their experience that such thickness variation might exist. However, the thinner shank wall on the CVX side means thinner airfoil wall and thicker shank wall on the CCV side means thicker airfoil wall, which resulted with poor balance of the entire blade, causing stress concentration on the TA area of the LE passage on the CVX side, and it is considered possible that they eventually led Blade #58 to fracture.

(4) Cause of deposits of sulfur or sodium

As described in 2.10. 2(3), although the composition of deposits on the HPC blade surfaces and on the shanks of HPT 2nd stage blades were not exactly the same, both contained sulfur and sodium. The volcanic ash sample taken from Mt. Sakurajima did not contain sulfur. It is considered that the sample had lost sulfur due to vaporization or wash by precipitation.

Concerning the difference between deposits on the HPC blade surfaces and on the shanks of the HPT 2nd stage blades, it is considered possible that the deposits on the HPC surfaces had either passed through the hot area such as the combustor, or had been transformed by the heat from the combustor. It is estimated that the deposits were from such things as volcanic smoke, yellow sands or sea salt.

However, as described in 2.11.2 (2), the Company had no standards to wash engines based on time in service or degree of contamination, nor did they have washing facilities. Because the engine manufacturer had posted on CESM that it was up to the operators to decide when to wash the engines, as described in 2.11.2 (1), and also because the Company was outsourcing the engine overhaul including the washing, it is estimated that the Company did not regard the facilities as necessary.

(5) Change in fuel flow

In this serious incident, as described in 2.1.1 (1), the fuel flow of the right engine increased by 256 1b/h from 18,080 1b/h at 16h45m18s to 18,336 1b/h at 16h45m19s, the time at which the blade was estimated to be fractured as described in 3.4 (2), then dropped significantly immediately after that. As described in 2.11.1 (2), it is estimated that FADEC, upon detecting the decrease in PS3 caused by the fracture of the blades, increased the fuel flow temporarily in order to increase N2. However, because the N2 did not increase, then PS3 did not increase, it is estimated that FADEC controlled to reduce the fuel flow.

At 16h45m22s, the right and left TLAs increased, and the left engine fuel flow, which had been normal, increased. It is considered possible that the crewmembers operated to increase the thrust because the Aircraft had been using derated thrust (DERATE-2) for take off.

## 3.5 Damage to Engine Case

It is considered that the damages on the blades, stators, and engine case of LPT, described in 2.3 (1) and (2), were triggered by the fracture of HPT 2nd stage Blade #58, as described in 3.4, and caused, in a chain reaction, and some of those fragments made two small holes in the LPT case adjacent to the LPT 1st stage. However, because there were no traces of collision of fragments, on the inner surface of the core cowl covering the LPT case, it is estimated that these fragments did not penetrate the LPT case.

## 3.6 Fracture of Fuel Supply Tube

(1) The timing of fracture of fuel supply tube #6

According to the DFDR and the QAR, the engine vibration level stood at maximum 4.99 units just after the takeoff. From the statements of passenger 1 and 2 in the cabin, “The Aircraft kept rattling.” it is considered that the engine had heavy vibration. The engine manufacturer estimated that the fuel supply tube #6 was fractured after experienced of load caused by the engine vibration, as described in 2.10.2 (5). Because the vibration was caused by the imbalance of the HPT (N2) rotor, as described in 3.4 (2), it is considered possible that the fuel supply tube #6 was fractured since HPT 2nd stage Blade #58 had been fractured. Furthermore, as for the slight bend observed at the coupling of the fuel supply tube with the fuel manifold described in 2.3 (3), it is estimated that it was bent when it was dislodged from the shroud can due to the heavy vibration and fuel pressure (about 1,000 psi) applied on the fractured area of the fuel supply tube, as described in 2.11.1 (8).

(2) Factors of fracture of fuel supply tube

Regarding that the fracture of fuel supply tube #6 was originated in its bead area which had a thick tube wall as described in 2.10.2 (5), because the ductility of the bead and its surrounding area is said to be reduced by the heat when welded and their strength is said to become lower than the base metal, it is estimated that the fracture started from the thick-walled part due to low cycle fatigue.

Also, concerning the unequal thickness of the tube near the fracture surface of the fuel supply tube #6, it is considered possible the tube was deformed by necking due to high stress load of repetitive tension and compression in constant amplitude.

## 3.7 Condition of Inside Core Cowl

(1) Fuel leakage

It is estimated that the leaked fuel from the fractured fuel supply tube #6 accumulated in the shroud can, and caused the O-ring and retaining ring to dislodge when they became unable to withstand the fuel pressure in the shroud can, then spewed in the engine designated fire zone inside the core cowl. The timing is considered to be shortly after the Aircraft became airborne based on the statements of passenger described in 2.1.2 (5) and as described in 3.6 (1).

Because the fractured fuel supply tube was located at 2h30m position as described in 2.3 (3), the opening between the fractured fuel supply tube and the shroud can was shaped like a crescent the widest part of which was directed to 3h position (toward the core cowl), and leaked fuel was influenced by the gravity, it is considered that the leaked fuel was sprayed aft downward and also just around downward from the fuel supply tube #6 as well.

(2) Start and spread of flame

① Based on the statements of passenger described in 2.1.2 (5), it is considered possible that the leaked fuel from the fuel supply tube #6 fractured shortly after takeoff, was sprayed and caught fire when it passed by the hot CRF.

② It is considered that the engine case was covered with soot as described in 2.3.(4) ① and ② because the fuel flow was high just after takeoff when the fuel started to leak, as described in 2.1.1 (2), and insufficient oxygen, caused incomplete combustion.

③ As described in 2.3 (5), based on the facts that the core cowl was melted from 3h to 4h position, where the temperature rose to 1200°F (642℃).

④ It is considered that after a hole was opened on the core cowl, air entering the core compartment redirected fuel spray and flame. It is likely the cowl overheated and the weakened material yielded to fan air flow loads. This weakening resulted in the tearing of the cowling material around the edge of the hole.

⑤ On the basis of the data described in 2.1.1 (1) and (2), AIR/GND sensor of a main landing gear perceives AIR in 19 seconds at 16:45, and the vibration value of the right engine reached a measuring upper limit (4.99 units) in 23 seconds at 16:45. Also, as it described in 2.4, that the fragments of engine blade etc. were getting scattered in the runway and runway east side grass area, and also that a runway east side grass area was burning with respect to about 100 ㎡, the possibility that the flame occurred from the hit that the vibration value of the right engine reaches to the measuring upper limit is conceivable after AIR/GND sensor detected AIR.

⑥ On the basis of the data described in 2.1.1 (3), the EGT started to rise at 16h46m51s after showing a downward trend until then, and kept the rising of about 30℃ for 20 seconds from 16h46m53s to 16h47m13s. During the time, the fuel flow and the N2 had not varied. Therefore, the EGT increase can be attributed to a combination of the inability to extract energy out of the airflow and the onset of the aerodynamic stall on the HPT rotor system due to the damage to the HPT blades.

⑦ The forward and aft of the stiffener ring in the forward part of the core cowl were burnt in a round shape at the upper and lower ends, so it is considered that the flame around this area spread differently from the flame that melted the core cowl mentioned in ④. Moreover, not only from the discolored heat shield described in 2.3 (4) and the burned isolator of the lower aft fire detector but also from their positions, it is considered possible that the direction of the fuel spray was changed as the fuel manifold failed and the shroud separated and moved.

## 3.8 Fire Detection

(1) Timing of the start of flame

As it described to 3.7 (2) ⑤ about the flame occurrence opportunity at the time of the serious incident, from the fragments of many engine blade etc. were discovered on the runway and in the runway east side grass area, and from the runway east side grass area was burnt, so it was thought the possibility that ruptured it when the aircraft took off. The vibration value of the right engine was becoming a measuring upper limit in the record of DFDR from this point and the fuel supply tube ruptured and fuel flowed out, and the possibility of flame occurrence conceivable from around 23 seconds at 16:45.

Based on the statements given by eye witnesses, the timing of the start of flame is further examined as follows;

① Eyewitness on the ground

The eye witness on the ground cited in 2.1.2 (8) stated, “The Aircraft was making a right turn but I did not see anything like smoke from the Aircraft. After the Aircraft completed the right turn and flew over the airport, I could see a thin brownish line trailing from the right engine”. Because the Aircraft started to change from right turn to left turn at about 16h47m55s according to the record on DFDR, it is considered possible that the flame had already existed by that time.

② Eyewitnesses of passengers

Passenger 1 described in 2.1.2 (5) stated, “After takeoff, I saw thin white streams like cloud or smoke coming out of the engine.” In addition, passenger 2 described in 2.1.2 (6) stated, “After a while from the take off, white smoke was flowing out from the right engine.” Based on these statements, it is considered that the fuel started to leak or the leaked fuel had caught fire shortly after the

Aircraft took off.

Concerning the statement of passenger 2, “There were flames just like those from a burner emitted a few times, and then they were gone.”, it is considered possible that the right engine EGT rose by about 30℃ starting to melt the core cowl at about 16h46m53s, or the passenger might have witnessed the increased combustion as a result of TLA increase operation by the PIC, which is to be described in 3.9 (3).

From those factors, although the timing of the start of flame cannot be accurately determined, it is considered that “Flame in the engine designated fire zone” started inside the engine core cowl at least one and a half minute earlier than the fire warning which activated at 16h48m42s as described in 2.1.1 (1).

(2) Reason why the fire warning did not activate promptly

① Timing at the beginning of fuel leak

At the beginning of fuel leak, the ventilation flow was weak as N1 dropped about 50%, oxygen was not supplied sufficiently and high fuel flow caused incomplete combustion, therefore, it is considered that the flame temperature was still lower than the fire warning activation temperature.

② Timing of reaching to Max value of Right Engine Vibration

The fire warning did not activate against the fire considered to have possibly existed in the core cowl from 3h to 4h position from 16h45m23s to 16h47m13s as described in 3.7 (2) ⑤. As described in 2.11.1 (5)②, the reason is considered that fire detector is not installed on the engine case between 3h and 4h position, and the ambient temperature of the fire detector did not rise to 635℃ at which the fire warning activates due to the outside air coming in through the hole opened by melting, which gave wall effect and cooling effect.

③ Timing after TLA increasing operation by the PIC

It is considered that the flame took time to reach the fire detector, because as described in 2.1.1 (1), after the PIC increased TLA temporarily from 16h47m13s, which might spread the damage in the turbines, the engine was rotating only by wind milling, decreasing the right fuel flow to 672 1b/h at 16h47m50s, with a small quantity of leaked fuel keeping the flame in the core cowl small.

(3) Activation of fire warning

It is considered that the leaked fuel scattered from the hole in the core cowl and dropped just downward from the fuel supply tube #6, due to reasons such as influence of gravity etc., and accumulated in the bottom of the core cowl, and started complete combustion as the lack of fresh air was resolved as the core cowl was melted at about 16h48m42s, upon which the fire warning activated. As described in 2.3 (4)①, because the heat shield was discolored to blue, and the isolator in the lower aft fire detector was also burnt, it is estimated that the surrounding area of the fire detector exceeded 635℃, and the fire warning activated.

(4) Stop of fire warning

As described in 2.1.1 (1), for the fire warning that activated at 16h48m42s, the PIC directed the co-pilot “Fuel cut off, pull” at 16h48m52s, to which the co-pilot cut off the fuel control switch and pulled the fire switch to shut the fuel valve, and it is considered that the fire was diminished or put out, lowering the temperature around the sensor, thereby stopped the fire warning. As a result, the fire switch light, discrete fire warning light, EICAS "R ENGINE FIRE" (red) MSG, and fuel control switch warning light all went out.

(5) Numbers and locations of fire detectors

As described in 2.1.1 (1), because it was about 16h48m42s that the fire warning activated, it is considered that one reason why the fire warning did not activate for the fire (16h45m23s to 16h47m13s) described in 3.7 (2) ⑤ was the installation location of the fire detector as described in 3.8 (2) ②. Therefore, as described in 2.11. 4, the aircraft manufacturer need to examine the compliance with the standard which specifies that numbers and locations of fire detectors shall ensure the prompt detection of fire in the specified zones.

## 3.9 Response of Operating Crewmembers

(1) Immediately after takeoff

At 16h45m19s (takeoff), as described in 2.1.1 (1), abnormal vibration occurred to the Aircraft. At 16h45m23s, the PIC called, "Oh, I have control." and took over the control from the co-pilot. Because the rapid decrease of the right engine thrust, made yawing of the Aircraft by approximately 4° to the right and the airspeed was maintained, it is estimated that the crewmembers were conducting appropriate piloting operation in accordance with the AOM "ENGINE FAILURE AFTER V1 PROCEDURE" described in 2.11.5 (1).

(2) Takeoff climb

As described in 2.1.1 (1), the right TLA remained unchanged until approximately 1,600 ft above airport level at 16h46m02s (43 seconds after takeoff), and then began to decrease at 16h46m03s. It is considered that this was because the operating crewmembers stabilized the flying attitude, executed gear up operation, set the flight route, and made communication with the ATC facilities.

(3) TLA increase by the PIC

From the DFDR, CVR records, and the statement of the PIC described in 2.1.1 (1) and 2.1.2 (1), although it is considered that the PIC increased the right TLA at 16h47m13s (one minute and 54 seconds after takeoff) based on the checklist of ENGINE LIMIT OR SURGE OR STALL described in 2.11.5 (4) in order to check the engine response, the right PS3 increased up to 138 psi at 16h47m27s but then dropped to 71 psi at 16h47m29s and to 19 psi at 16h47m35s. It is considered that this was because the TLA increasing operation expanded the damages of turbines, led all right engine vibration level to 0 unit at 16h47m43s, further the right PS3 and P49 to become almost the same value as P0, thereupon lost the engine function. Based on these, it is considered that the PIC had better to judge from the right engine vibration levels and N1 condition, not to conduct the TLA increase operation.

The aircraft manufacturer had the opinion that this operation by PIC was not in error.

(4) Takeoff route

The flight route of the Aircraft after takeoff was in compliance with the reference procedure at engine failure at Kagoshima Airport specified by the Company, as described in 2.11.5(5). Although in Visual Meteorological Condition, the flight was continued on the specified flight route. Though it was not the shortest flight route to make an emergency landing at Kagoshima Airport, it is considered that they made appropriate response to ensure fire extinguishing operation.

## 3.10 Measures to Alleviate Damage and to Prevent Recurrence

(1) Fire detector of the same type engine

In this serious incident, it is considered that the fire warning system could not detect the flame promptly, as described in 3.8. As described in 2.11.8, the engine manufacturer and the aircraft manufacturer regard that the duration of the fire and the damage to core cowl was not a safety threat. The on site investigation revealed the flame did not extend beyond the fire containment feature of the nacelle. However, the aircraft manufacturer needs to examine the compliance with the standard described in 2.11.4 because the fire was not detected immediately in the engine designated fire zone.

(2) Improvement of blade manufacturing

① TA area

As the TA radius is smaller in the blades made by Blade manufacturer A as described in 3.4 (3)①, it is considered possible that stresses are concentrated on the TA area, and a crack originated and resulted in the fracture due to fatigue.

Furthermore, as described in 2.10.3 (3), the computer simulation conducted by the engine manufacturer proved that the larger TA radius was more resilient against stresses.

Based on these facts, it is considered that Blade manufacturer A of the fractured blade should make blades with larger TA radii in order to avoid stress concentration on the TA area.

② Wall thickness of CVX shank LE passage

As described in 3.4(3)③, it is considered possible that the thin shank wall on the CVX side could also mean a thinner blade airfoil on the CVX side, resulting in overall imbalance, which further caused the stress concentration on the CVX LE passage, separating Blade #58 from the TA area, therefore it is considered that for thorough quality control, blades need to be inspected correctly after manufacturing in order to minimize variation of the shank wall thickness on the CVX side.

③ Coating

As described in 2.10.2 (1) ②, the coating thickness of the TA area of the LE passage of Blade #58 on the CVX side was thinner than other areas, about half of the coating of Blade #49, but the hot corrosion was limited within the coating layer.

Moreover, as described in 2.10.3 (2) ①, the rate of crack differs from one coating supplier to another, and coating supplier G had a smaller rate of crack than coating supplier F. The blade fractured in this serious incident was processed by coating supplier F. Based on these, it is considered that measures need to be considered for thorough quality control in order to reduce cracks in the coating.

(3) Engine washing

Operators usually wash engines to secure the EGT margin. Although the monthly average EGT during cruising cannot be simply compared due to the differences in conditions including temperature, altitude and engine thrust setting, the monthly average EGT for the four-month period prior to this serious incident were a little lower than those of the same period of the previous year, as shown in 2.6.6. Therefore, it is estimated that the occurrence of this serious incident could not have been predicted from monitoring EGT trend.

As described in 3.4 (3) ②, because it is considered possible that the fractures of the blade was caused by type II hot corrosion in this serious incident, it is desirable that the operators carry out the engine washing program recommended in CESM of the manufacturer as described in 2.11.2 (1), in order to remove deposits containing such as sulfur, in the fans and compressors.

## 3.11 Rescue and Firefighting

The Fire Department of Kagoshima Airport had three fire engines set out along the runway immediately after receiving the emergency report of the Aircraft, and the fire engines promptly extinguished the fire which was considered to be caused by the high-temperature engine fragments, scattered by the Aircraft taking off in the grass area by the runway. It is considered that noticing those fragments was difficult because the firefighters engaged in the firefighting stated they did not notice the fragments such as turbine blades scattered around the runway in the dusk, and considering the fragments were small and black.

Then, after the Aircraft landed, the firefighters followed the Aircraft by fire engines prepared to start firefighting activities at any moment, and it is considered that their response was good. (See photos 4 and 5.)

## 3.12 Emergency Evacuation

It is estimated that the PIC judged there was no need for an emergency evacuation because the right engine fire warning went off, the event engine had already been shut down and there was no sign of fire. It is considered that such response by the PIC was appropriate under these conditions.

## 3.13 Response of Kagoshima Airport Office (JCAB)

As described in 2.11.6, after the Aircraft took off at 16h45m19s, two commuter airplanes took off from the same runway, then, the Aircraft made a landing. During this period, the fragments of the right engine of the Aircraft had been scattered on and near the runway and there was a fire alongside the runway.

It is estimated that the Tower Controller, as described in the statement in 2.1.2 (7), could not presume that the smoke in the grass area was caused by the fragments scattered from the failed right engine of the Aircraft, due to the following reasons;

①-The right engine of the Aircraft could not be seen from the Tower as the engine was in the shadow of the fuselage of the Aircraft during take off roll;

②-One minute and 11 seconds passed after takeoff when the engine failure was reported from the Aircraft to the Radar Controller, and then, the report was forwarded to the Tower Controller; and,

③-The engine fire and relevant information were not reported from the Aircraft to the Radar Controller.

Therefore, it is estimated that the Tower Controller gave clearance to one commuter airplane for take off at 16h48m40s as the commuter airplane reported to him that the smoke would not be an obstacle for takeoff.

At 16h52m00s, the Aircraft reported to the Radar that it would return to Kagoshima Airport due to engine failure, but it is estimated that the Information Officer, who was out at the scene of fire in the grass area, could not find engine fragments in the dusk, and the fragments scattered around the runway were black and small. Just as the

Controller would not relate the smoke in the grass area with the engine failure, it is estimated that the Information Officer could not possibly imagine that the engine fragments were scattered around the runway from the report of the engine failure of the Aircraft and from the fact of the fire in the grass area.

However, because there is always no sign of fire in the grass area, it is considered possible that the fire could have been detected and a safer environment would have been prepared for an emergency landing if the Information Officer had conducted investigation into the cause of the fire.

Later, by the time the second commuter airplane reported that they were ready for takeoff, the Tower Controller noticed the emergency by Aircraft's transponder, but allowed the second commuter airplane to take off, because;

①- the fire alongside the runway had been put out

②- the first commuter airplane had taken off without any trouble, and

③- there was enough space between the Aircraft and the second commuter airplane.

However, taking off from or landing on the runway with scattered engine fragments could cause tires to burst or fragments to be sucked into engines, and it is possible to cause a serious aircraft situation.

Therefore, it is of great importance for managers of the aircraft moving area to correctly handle a case when an aircraft report engine failure after takeoff, taking the possibility of scattered engine fragments on the runways into consideration.

# 4. PROBABLE CAUSE

It is estimated that this serious incident was caused as follows: When the Aircraft became airborne, one of the 2nd stage high-pressure turbine blades of the right engine was fractured which had had considerable fatigue crack, leading other blades to be fractured which caused vibration to the right engine from unbalanced rotation, and the vibration further caused fracture of the fuel supply tube #6 inside the shroud can, moreover, the retaining ring at the end of the shroud can was dislodged by the pressure of fuel injected in the shroud can as well as by the engine vibration causing the fuel to leak, then the fuel was ignited when it contacted the hot section, and resulted in a flame in the engine designated fire zone.

Although the exact cause of the fatigue crack found in the turbine blade could not be determined precisely, it is considered possible that the following factors were combined and contributed:

①-because the TA (turnaround) radius of the cooling air passage inside the blades was smaller, the stress was prone to concentrate on that area and thereby it became prone to cause cracks, and later, the cracks progressed due to low-cycle fatigue,

②-because sulfur was detected from the rupture surface, the blade was in the condition that crack came by hot corrosion.

③-thinner LE passage wall of the shank created the high stresses on the entire blades.

The most probable cause for the fracture of the fuel supply tube was the excessive loads placed on the tube due to vibrations from the engine imbalance.

As for the fire warning which did not activate immediately until a significant area of the core cowl had been melted, it is estimated that it was because the flame was directed to the area where no fire detector element was routed.

# 5. SAFETY RECOMMENDATION

The Japan Transport Safety Board, after having reviewed this serious incident, recommends that the Federal Aviation Administration (FAA) of the United States of America examine the following item of all the General Electric CF6-80C2 series engines and take necessary measures.

"Realignment of the engine fire detector locations"

In this serious incident, after the aircraft took off, the fire warning of the right engine, where flame had started, did not activate even when a significant area of the right engine core cowl melted, and it took more than about one minute and 30 seconds before the fire warning activated.

The airworthiness standard (FAR25.1203(a)) specifies that numbers and locations of fire detectors ensure prompt detection of fire in each designated fire zone.

However, in the case of this serious incident, judging from the extent of the damage to the core cowl, it is difficult to say that fire was "detected promptly" as set forth in the standard.

Therefore, the designers and manufacturers of the aircraft should examine numbers and locations of fire detectors to ensure prompt detection of a fire in an engine designated fire zone.

# 6. SAFETY OPINIONS

## 6.1 Implementation of Engine Washing

The exact cause of the fracture of turbine blade in this serious incident could not be determined. However, it is considered possible that the deposits containing sulfur which existed on the coating of internal cooling air passage of HPT 2nd stage blades caused the depletion of coating, the strength of the base metal decreased after reacting with oxygen, leading to a crack, then the fracture by low-cycle fatigue.

Since Japan is surrounded by seas, yellow sands containing sulfides blown in the air come to Japan, and there are many active volcanoes in Japan, it is considered that Japanese operational environment is harsh in which there are many occasions to have deposits containing such as sulfur that could adversely affect the internal parts of engine.

Although engine designers and manufacturers recommend washing mainly for the purpose of improving fuel efficiency, the washing is also expected to be effective in removing deposits, therefore operators that operate aircraft in Japan should implement engine washing as necessary taking into consideration of such factors as operating conditions of their engines.

## 6.2 Quality Control of Blade Coating

The metallurgical analysis in the investigation of this serious incident revealed that the rate of crack was remarkably different by supplier of blade coating. The coating suppliers should maintain thorough quality control to reduce crack occurrence thereby to prevent blade fracture.

# 7. REFERENCES

The following measures were taken by concerned organizations in order to prevent recurrence of such serious incident.

## 7.1 Japan Civil Aviation Bureau of the Ministry of Land, Infrastructures, Transport and Tourism

In the wake of this serious incident, Japan Civil Aviation Bureau of the Ministry of Land, Infrastructures and Transport took the following measures.

(1) On December 3, 2005, Japan Civil Aviation Bureau directed airline companies in Japan operating aircrafts equipped with the same type of engines as the event engine (General Electric CF6-80C2B6F) to inspect the HPT blades.

(2) On January 31, 2006, "make an effort to actively collect information to examine the information from diverse point of views and to comprehensively determine the necessity of on-site check should a fire start in the limited zones" is framed and added to the list of "Safety inspection and Flight Limit in the Limited Zones, etc." of the Aviation Safety Services Procedure Rules.

(3) On June 22, 2007, TCD7129-2007 was issued based on AD2007-11-20. (Replace retaining rings with snap rings)

## 7.2 Federal Aviation Administration (FAA) of the United States of America

As of July 10, 2007, the implementation of SB73-0337 was mandated for the relevant engines based on AD (airworthiness improvement order) 2007-11-20. (Replace retaining rings with snap rings)

## 7.3 The Engine Manufacturer

(1) After this incident, the Engine manufacturer added the requirements of the appropriate coating thickness to prevent cracks, and has made changes to the drawings to reduce thickness of coating in the internal shank.

(2) On February 12, 2008, the Engine manufacturer issued the Service Bulletin of CF6-80C2 (SB) 72-1283 " High Pressure Turbine Rotor – New Stage 2 Blade" on the cooling air passage inside the turbine blades. For the purpose of reducing the likelihood fracturing, the Engine manufacturer redesigned the blades to reduce the stress concentration on TA radii.

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| Figure 2 | Boeing 767-300 |
| Figure 3 | DFDR data |
| Figure 4 | CF6-80C2 Engine Overview |
| Figure 5 | Sooty Condition of Engine Outside |
| Figure 6 | Fuel Supply Tube |
| Figure 7 | Engine Fire Detector |
| Figure 8 | Airflow in Engine Case |
| Figure 9 | Hole of LPT Case |
| Figure 10 | Combustion Chamber & HPT Blade Cooling Air Passageway |
| Figure 11 | HPT Stage 2 Blade Cooling Air Passageway |
| Figure 12 | Retaining Ring |
| Figure 13 | Calculated Stress – TA Radius |
| Figure 14 | HPT Stage 2 Blade |
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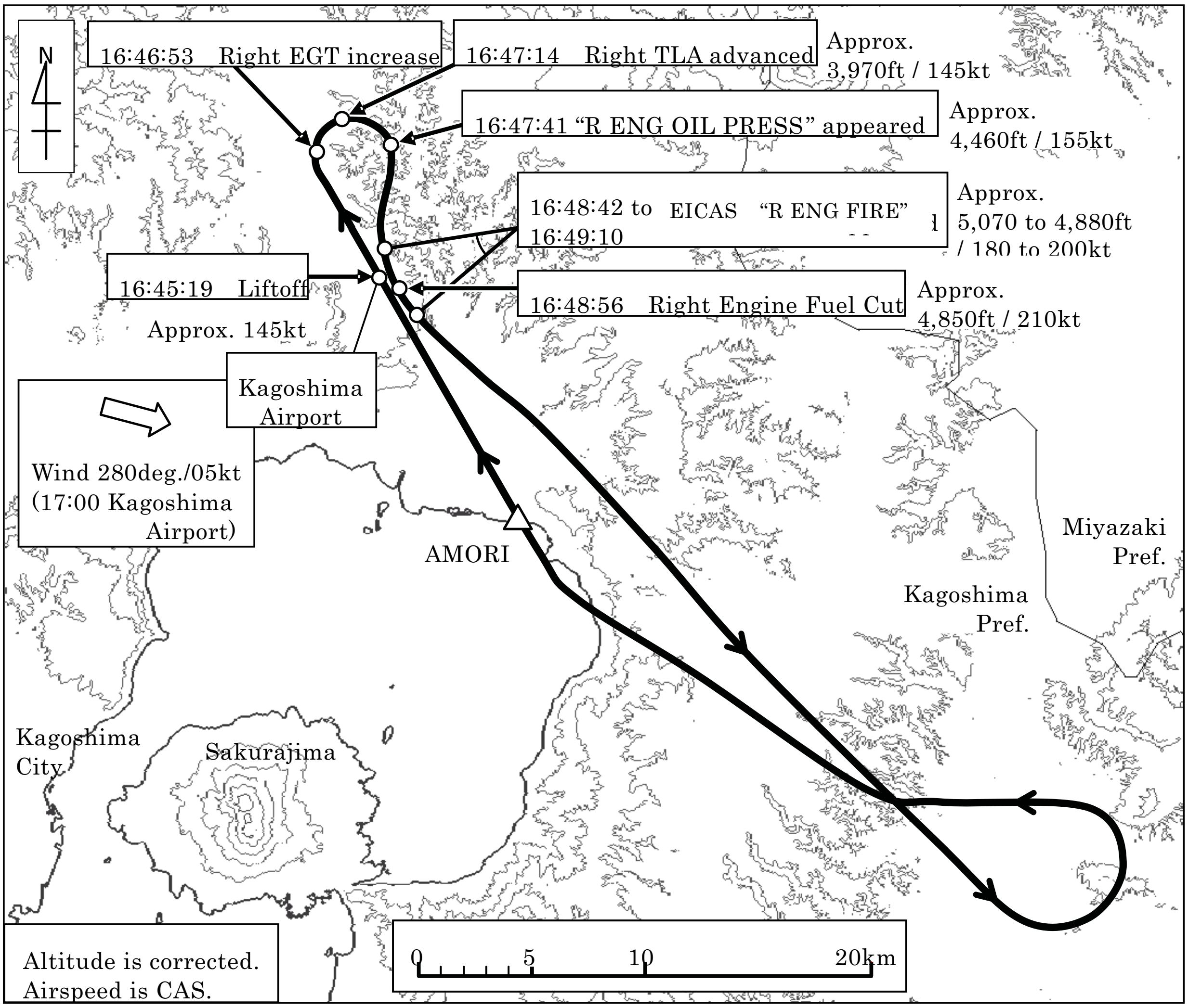
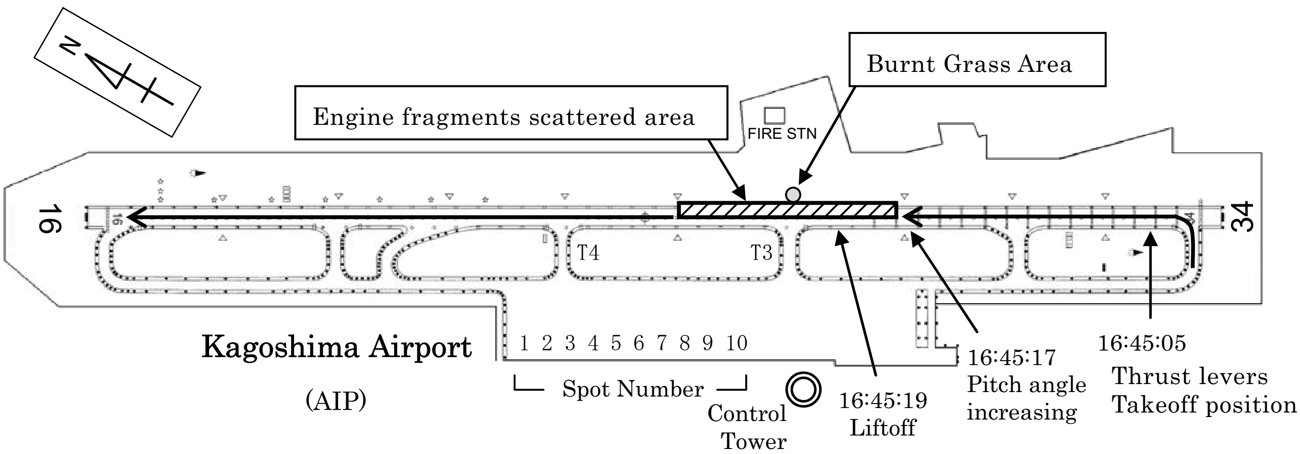
# Figure 1 Estimated Flight Route




Digital Map 50m Grid (Elevation), Geographical Survey Institute

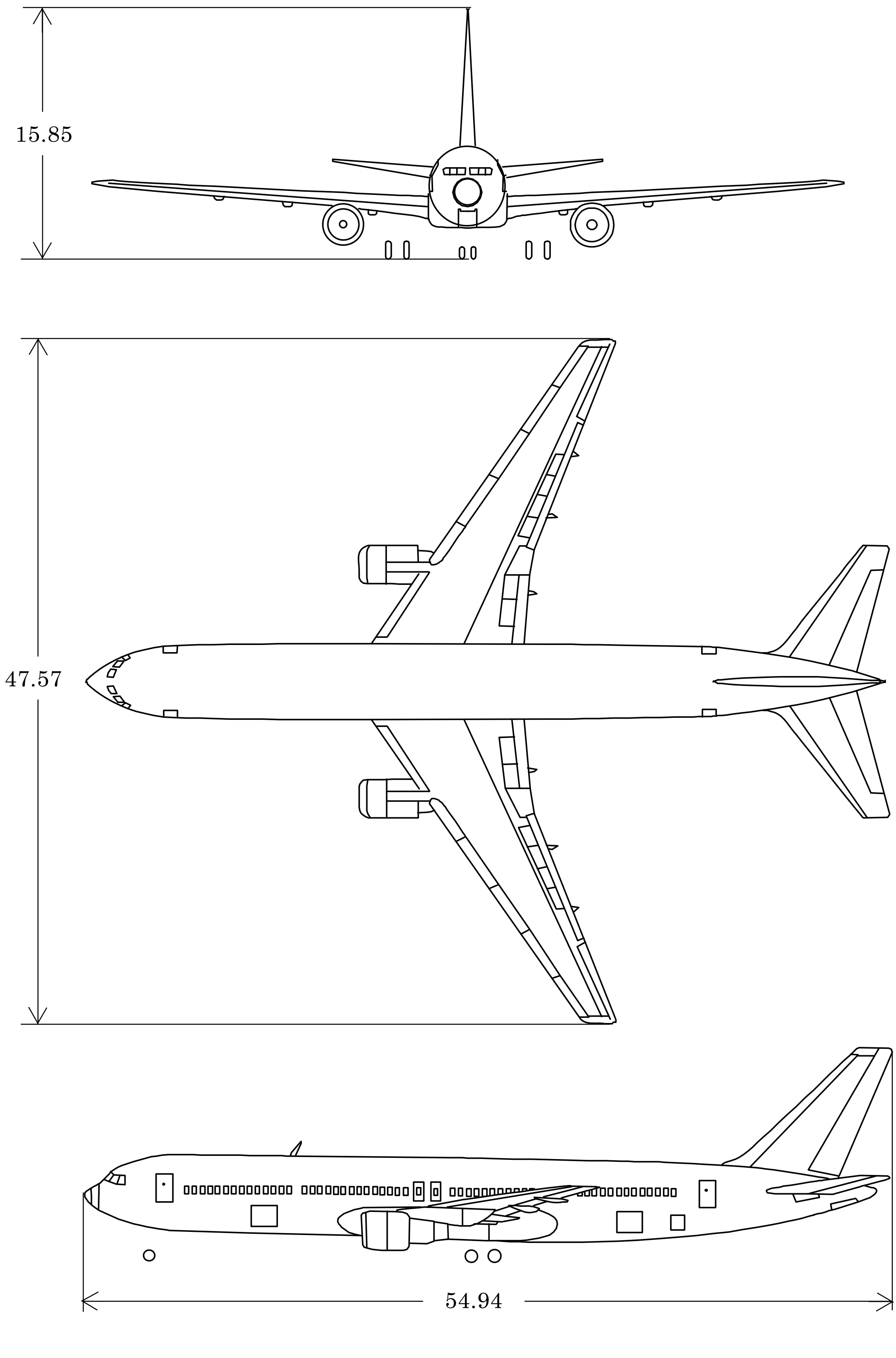



Kagoshima Airport

(AIP)

Figure 2　Boeing767-300

Unit：m



## Figure 3 DFDR data

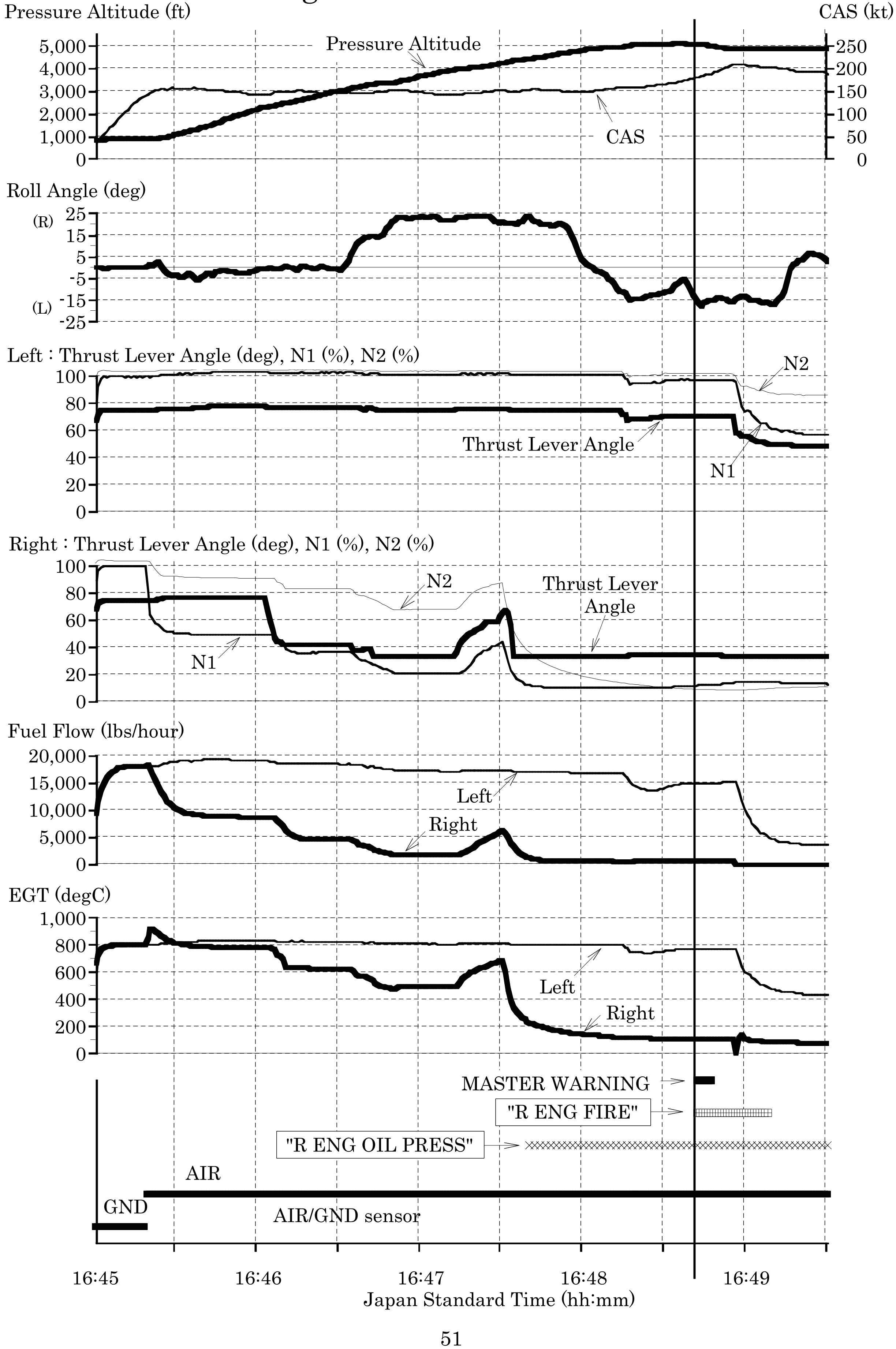
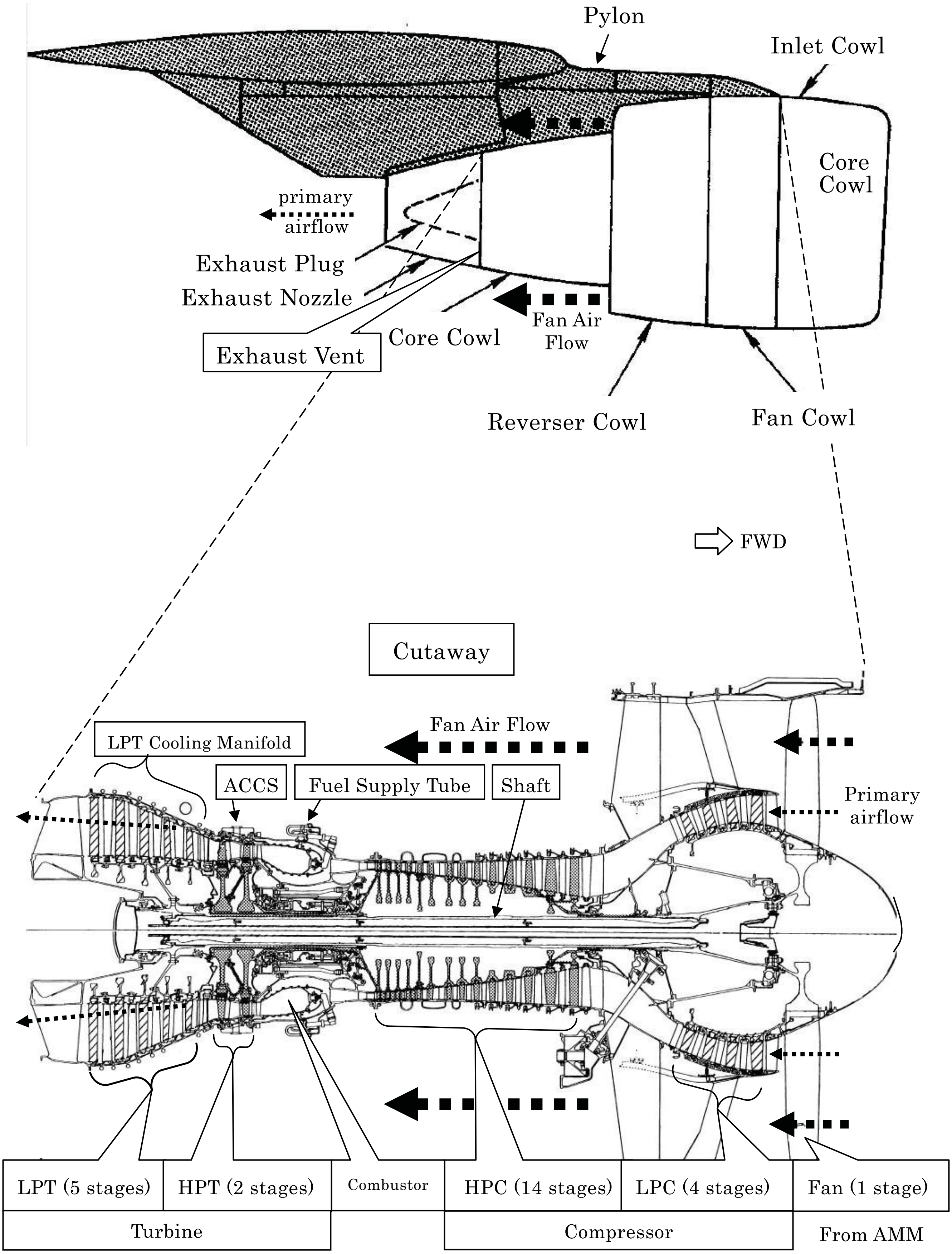
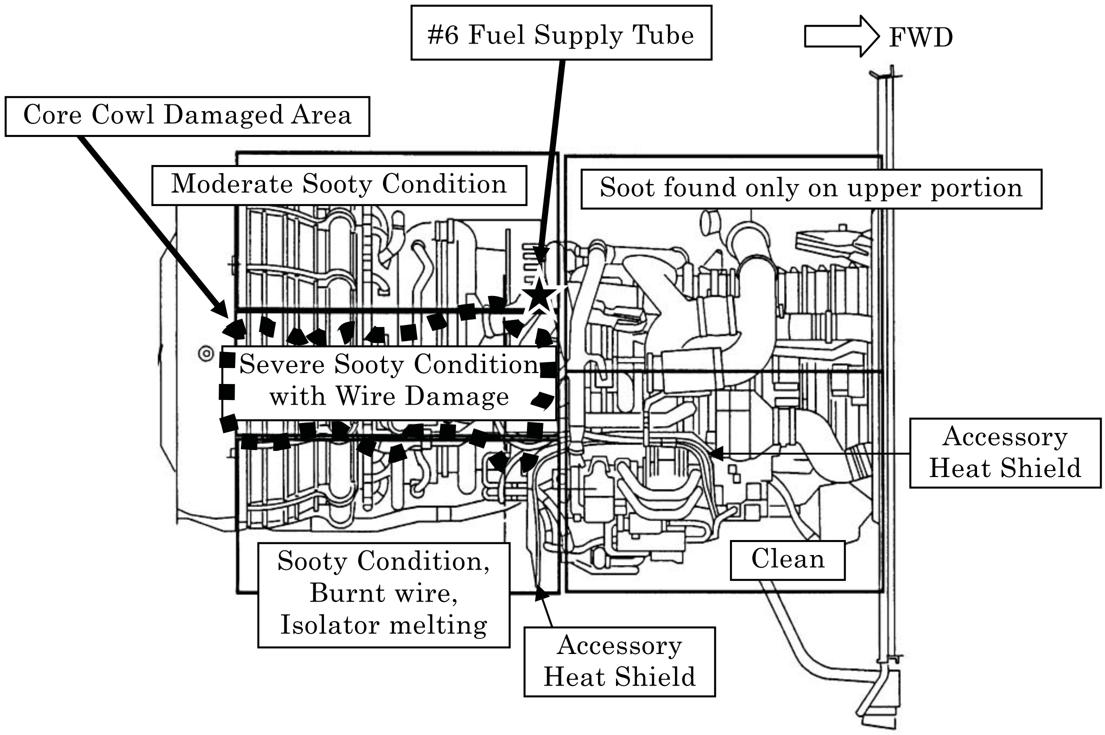


Figure 4　CF6-80C2 Engine Overview

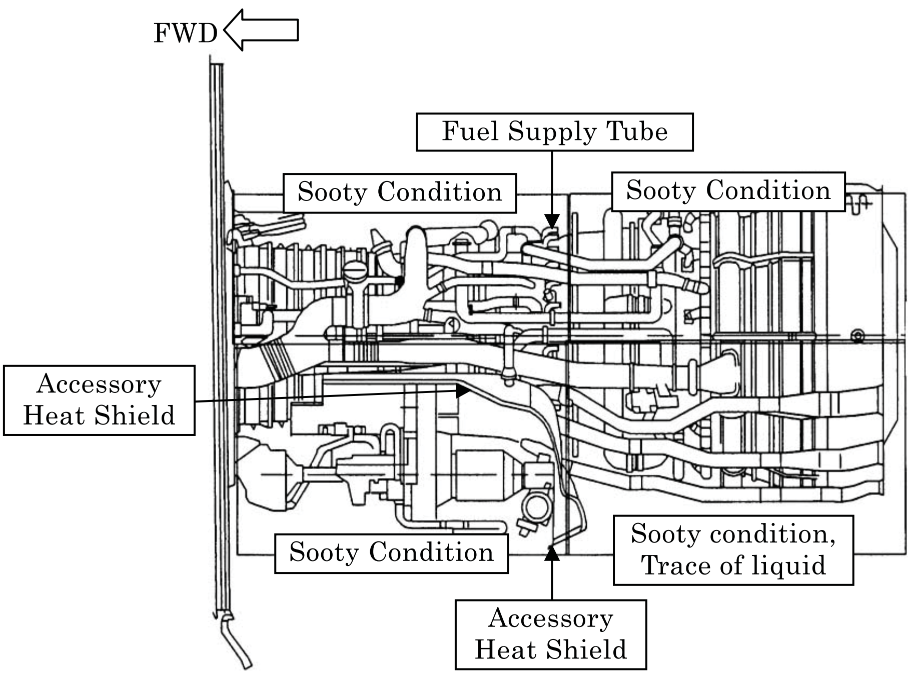


# Figure 5　Sooty Condition of Engine Outside

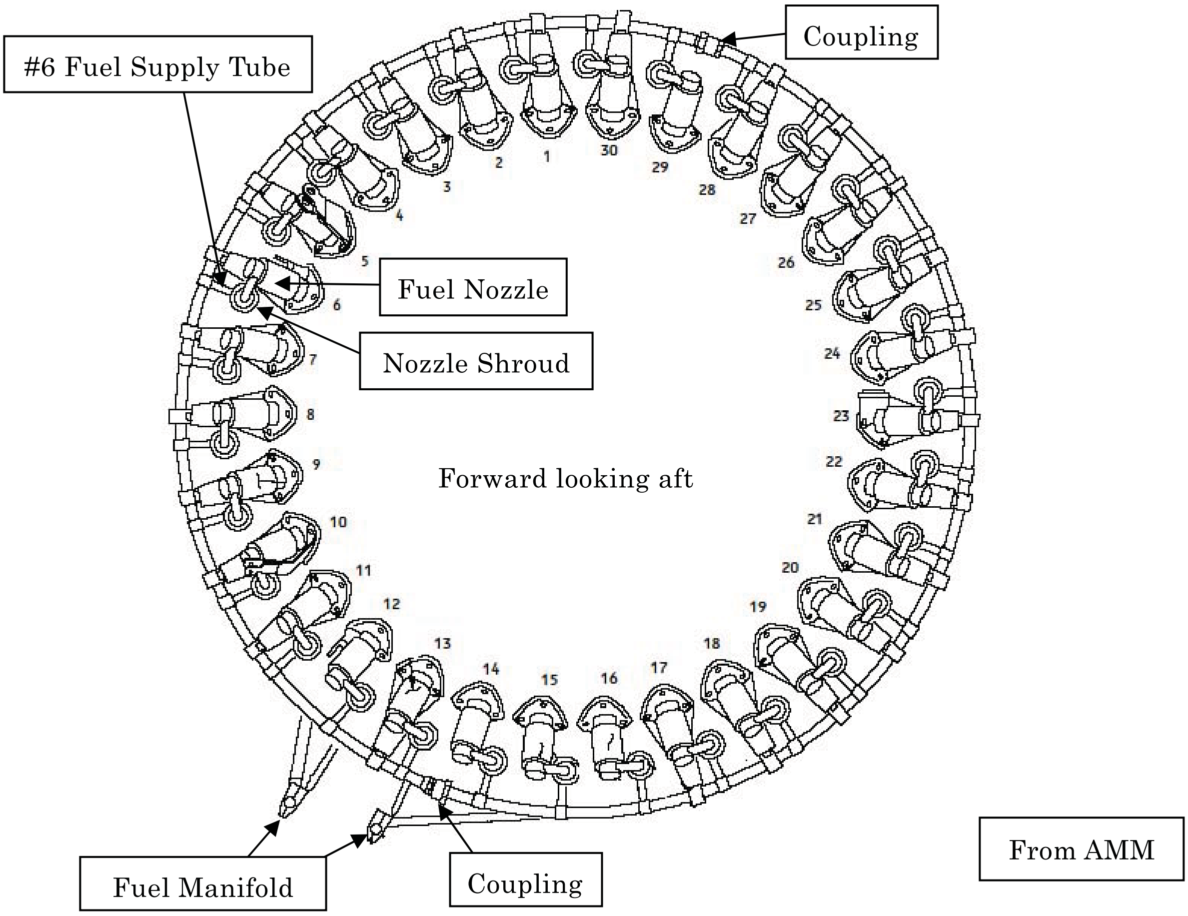
RH Side of Engine (see photo 6)



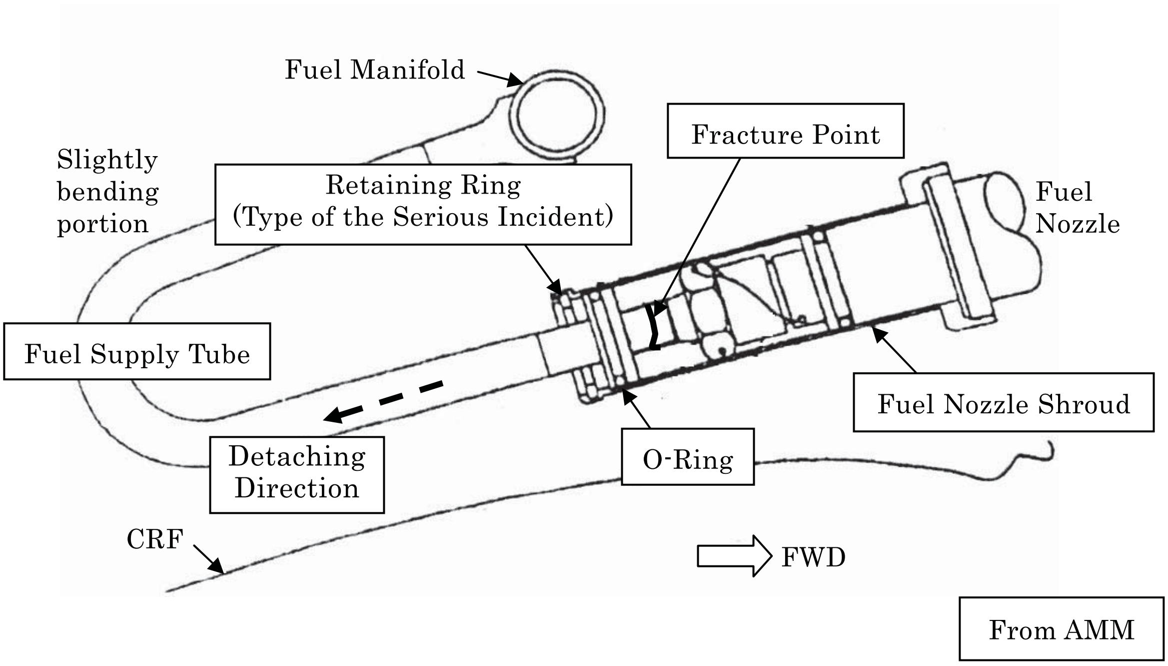

LH Side of Engine



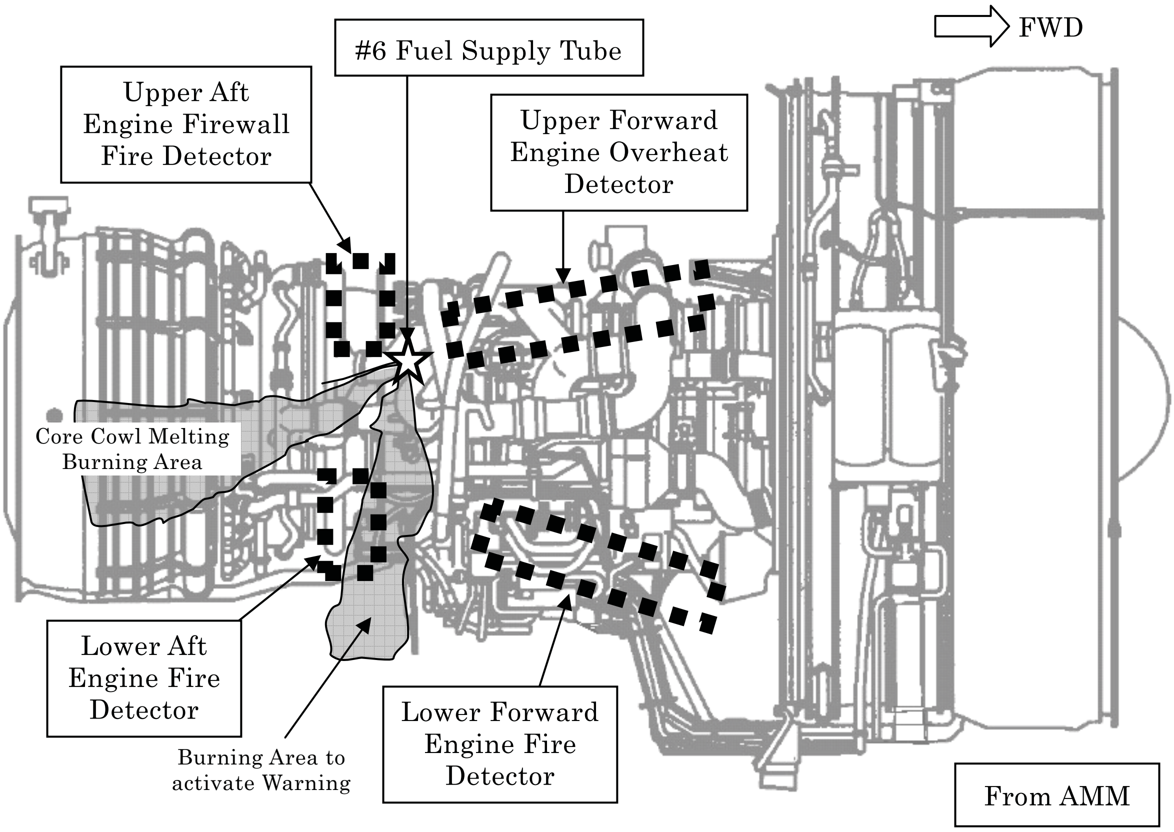
# Figure 6　Fuel Supply Tube



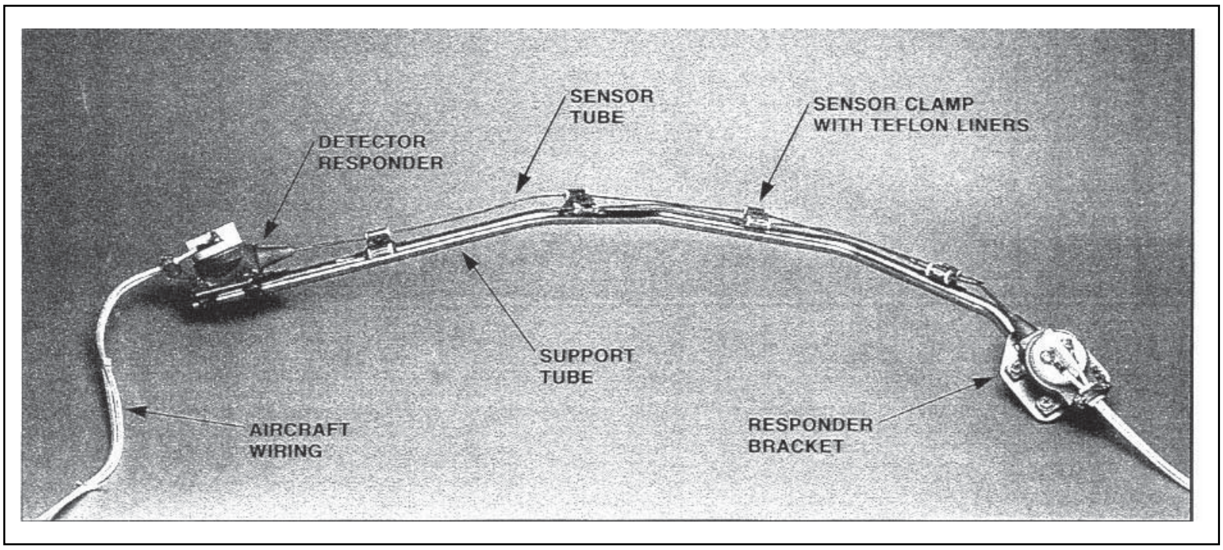

## Connection with #6 Fuel Supply Tube and Nozzle Shroud



# Figure 7 Engine Fire Detector

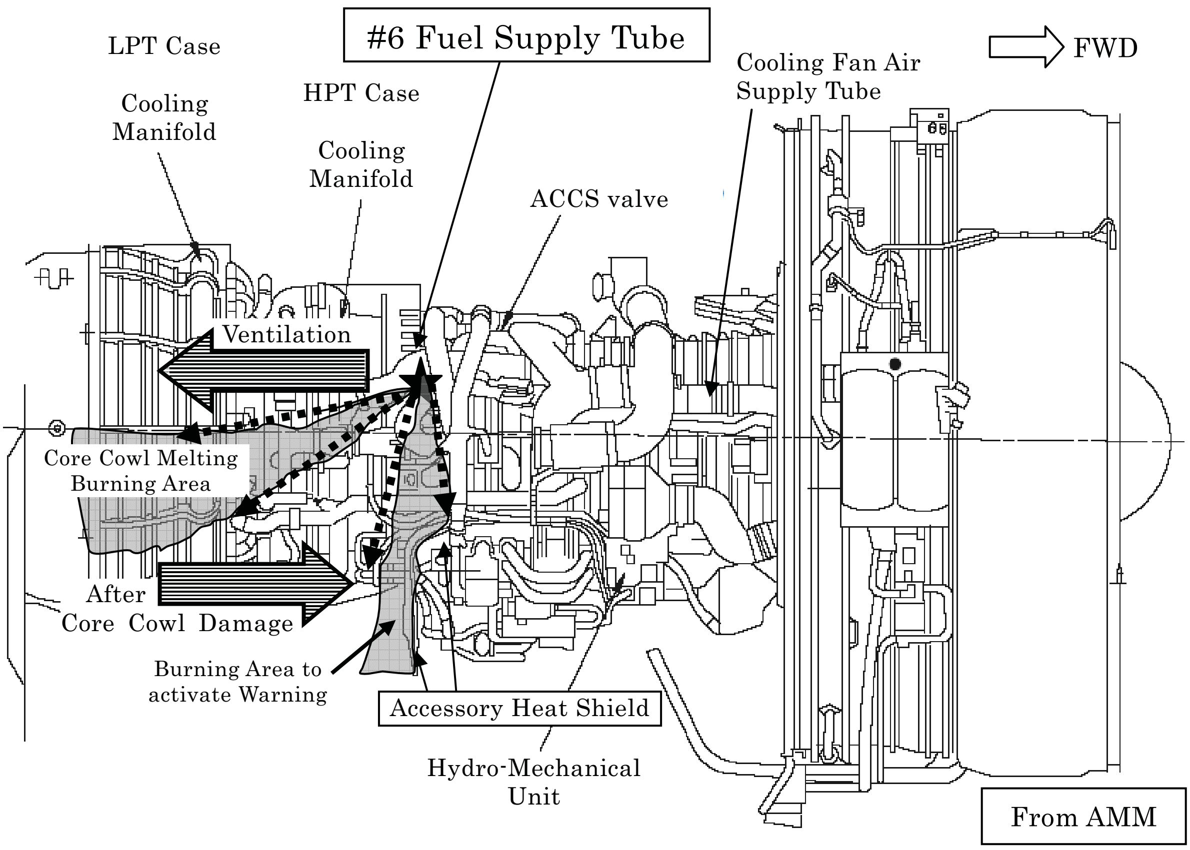



# Fire Detector




From Aircraft Manufacturer's Document

# Figure 8　Airflow in Engine Case




## Engine Case Cooling Airflow

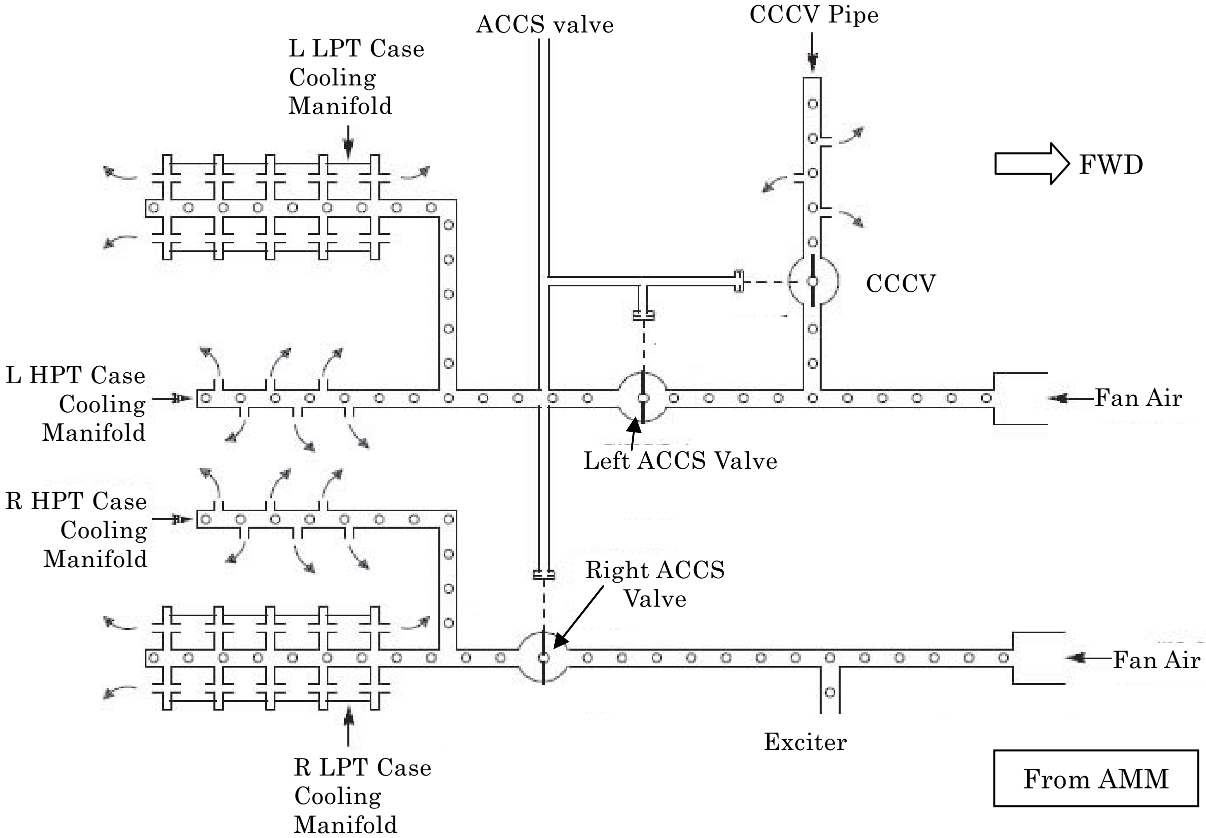
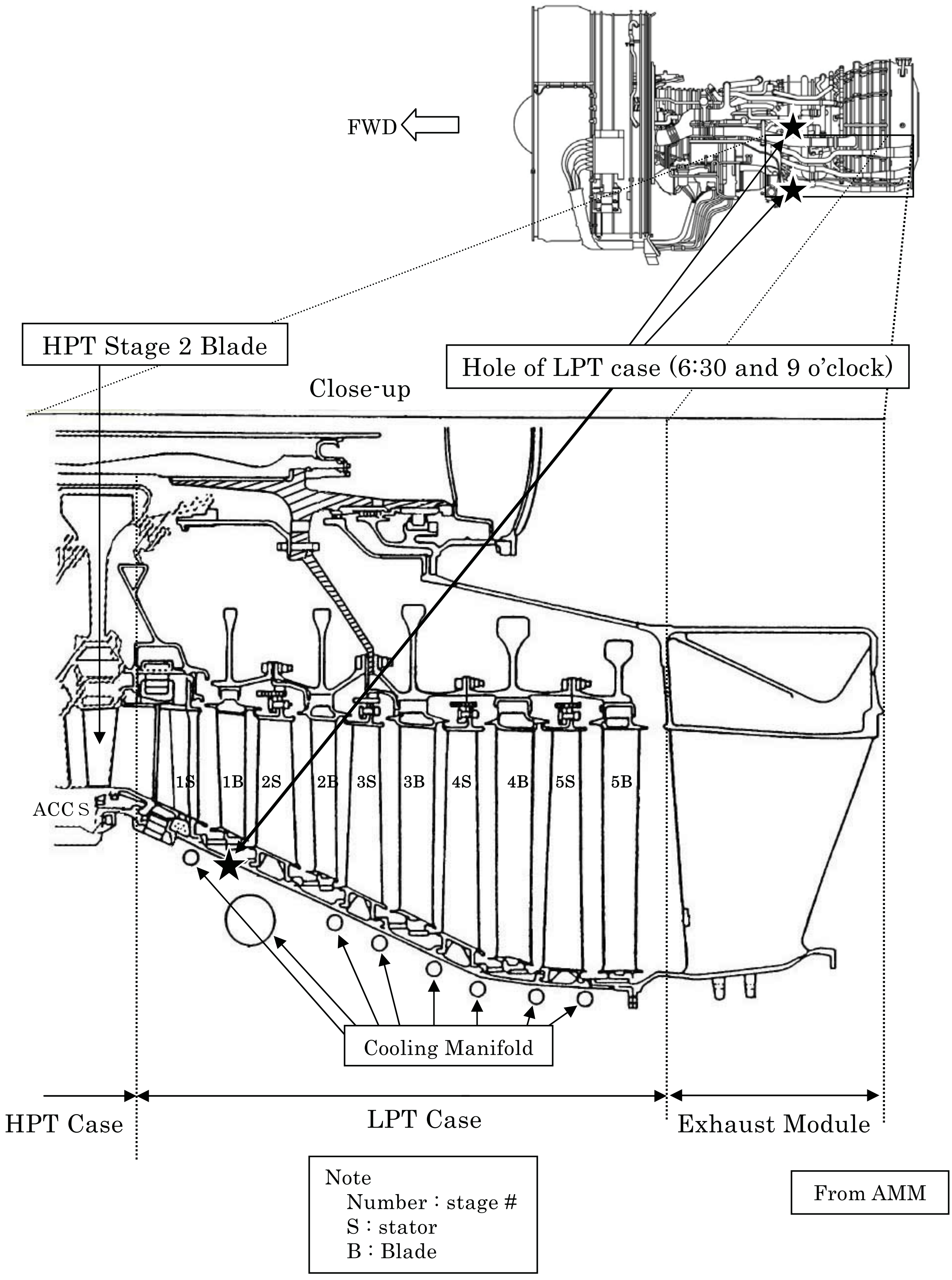
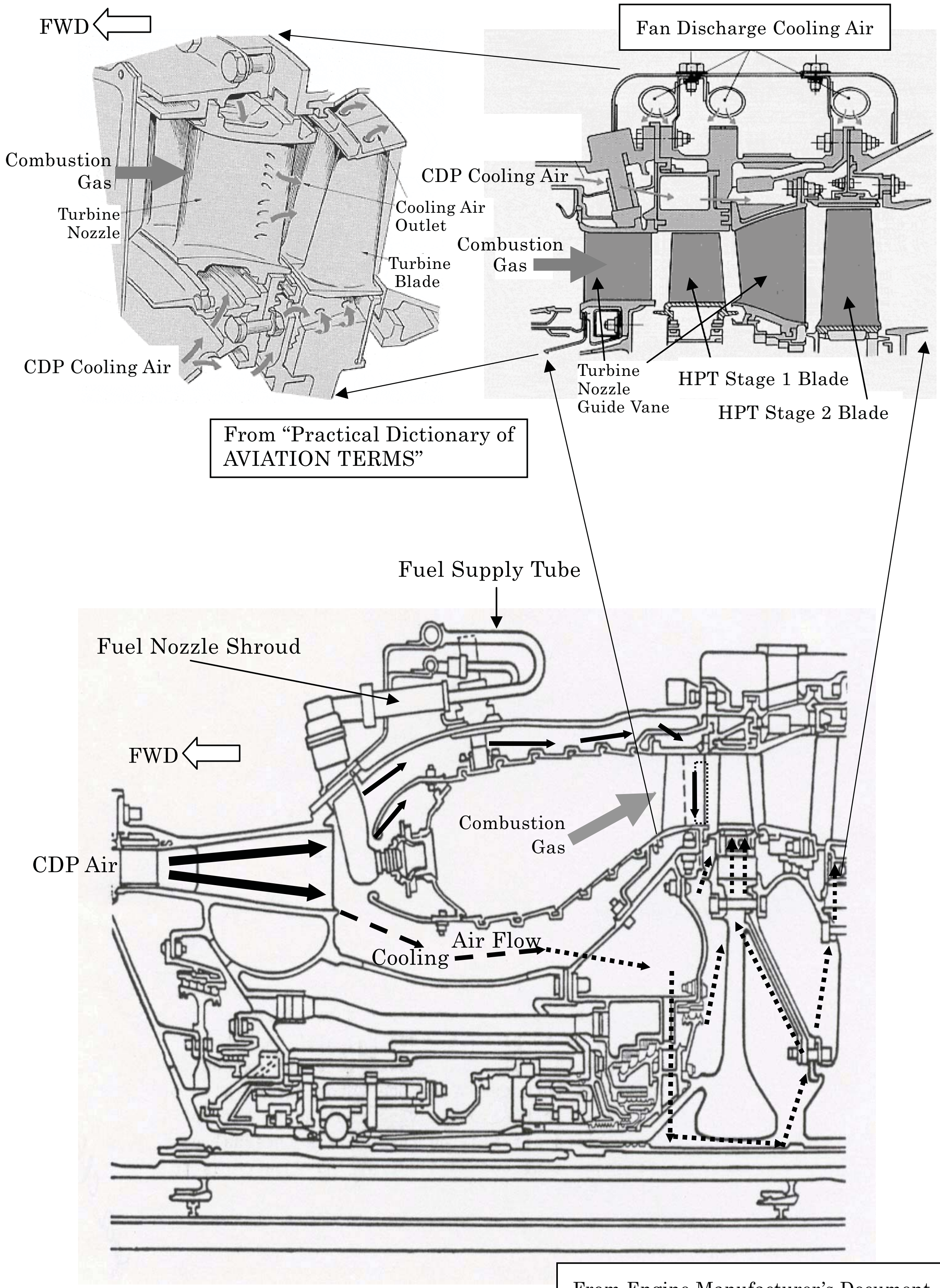


Figure 9　Hole of LPT Case

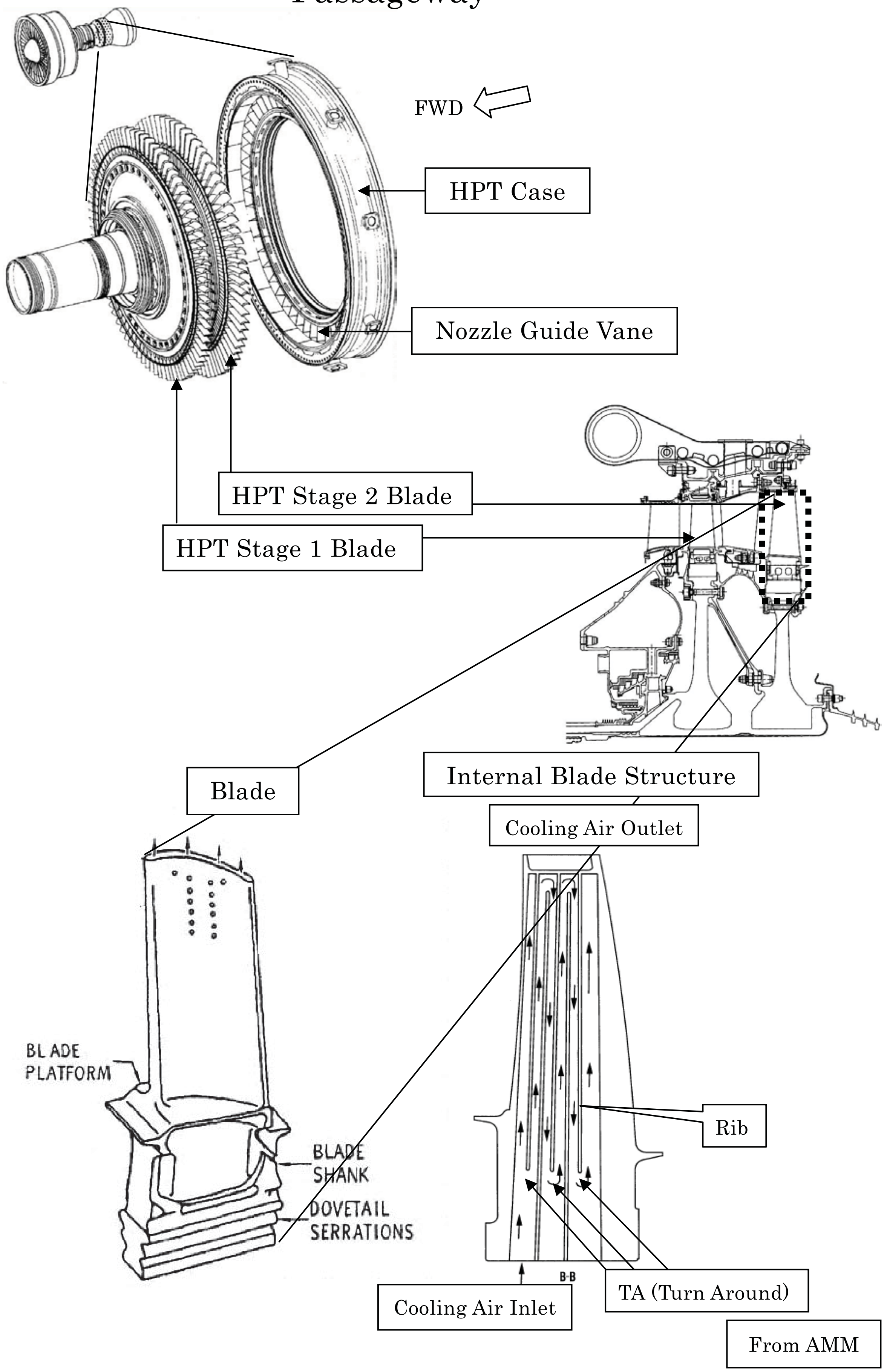


# Figure 10 Combustion Chamber & HPT Blade Cooling Air Passageway

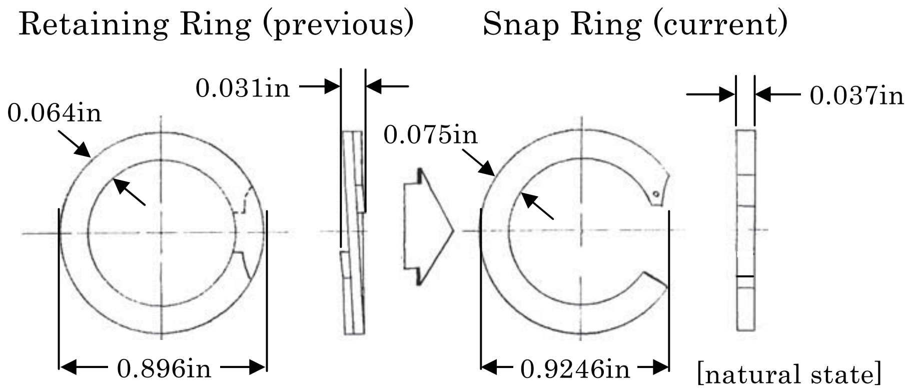


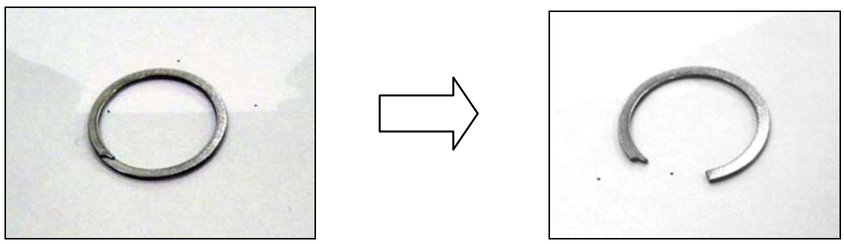

From Engine Manufacturer's Document

## Figure 11 HPT Stage 2 Blade Cooling Air Passageway

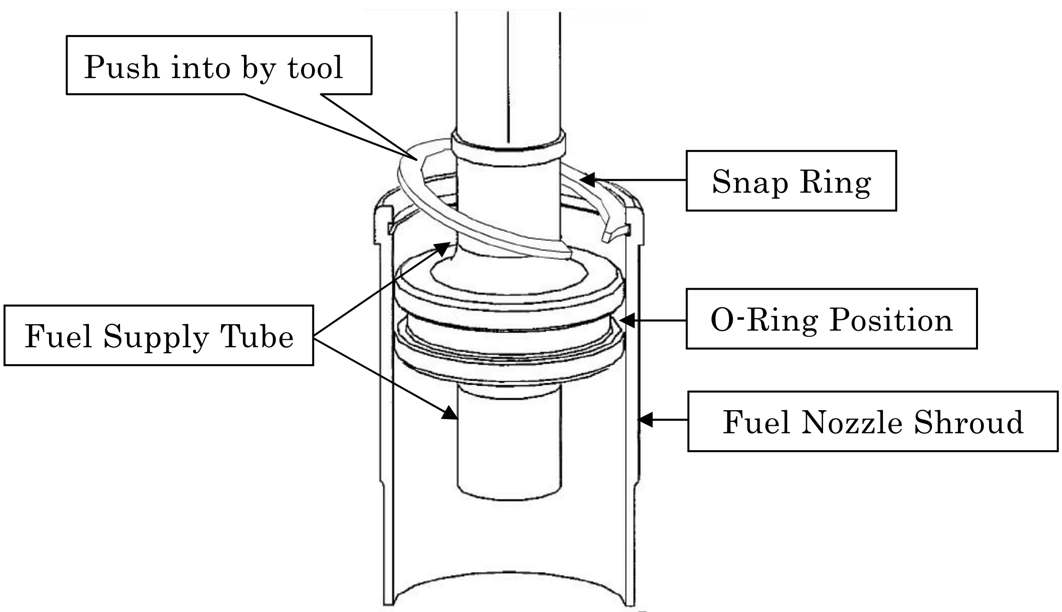


# Figure 12　Retaining Ring



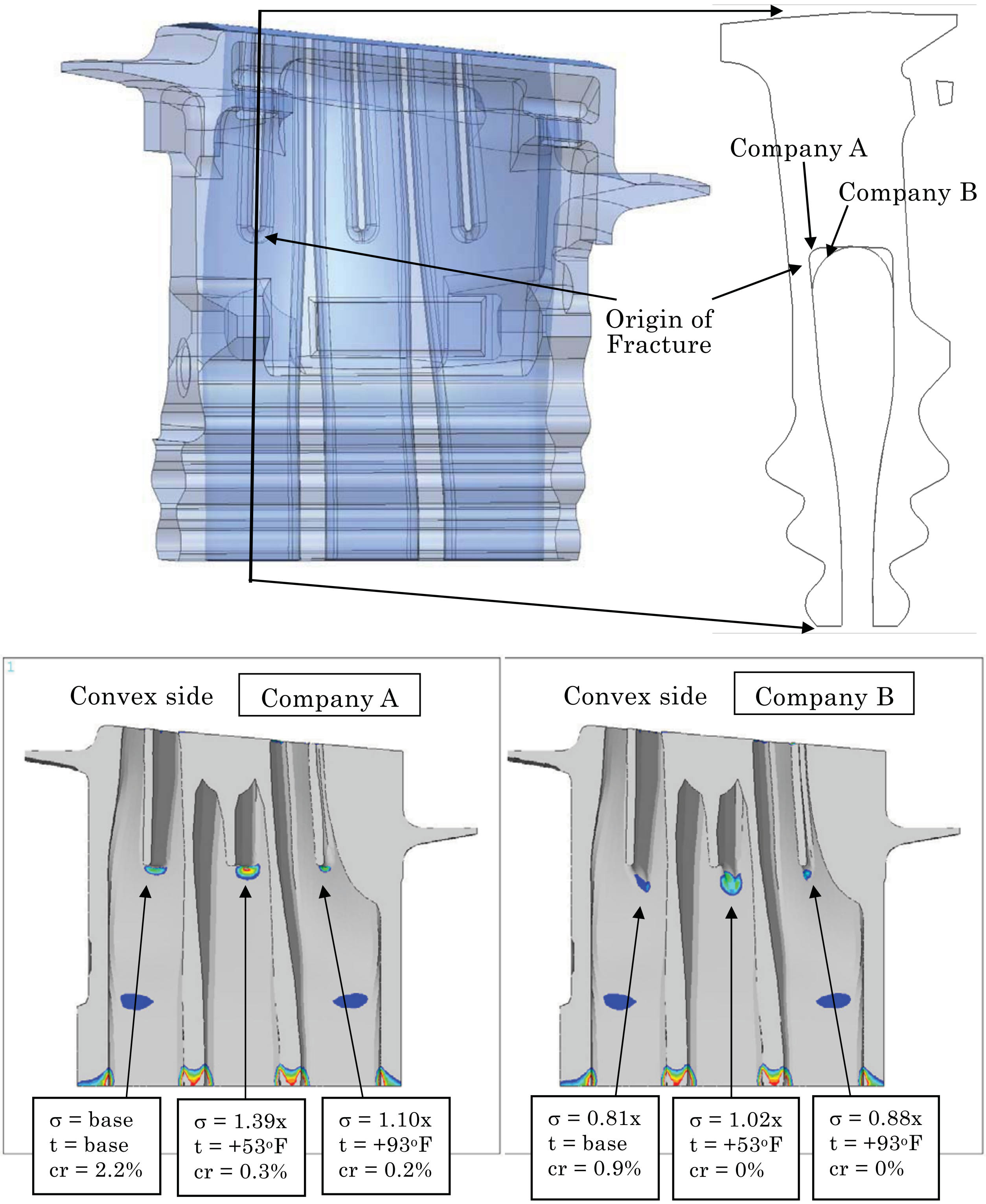



## Snap Ring Installing




From SB73-0337

## Figure 13　Calculated Stress - TA Radius

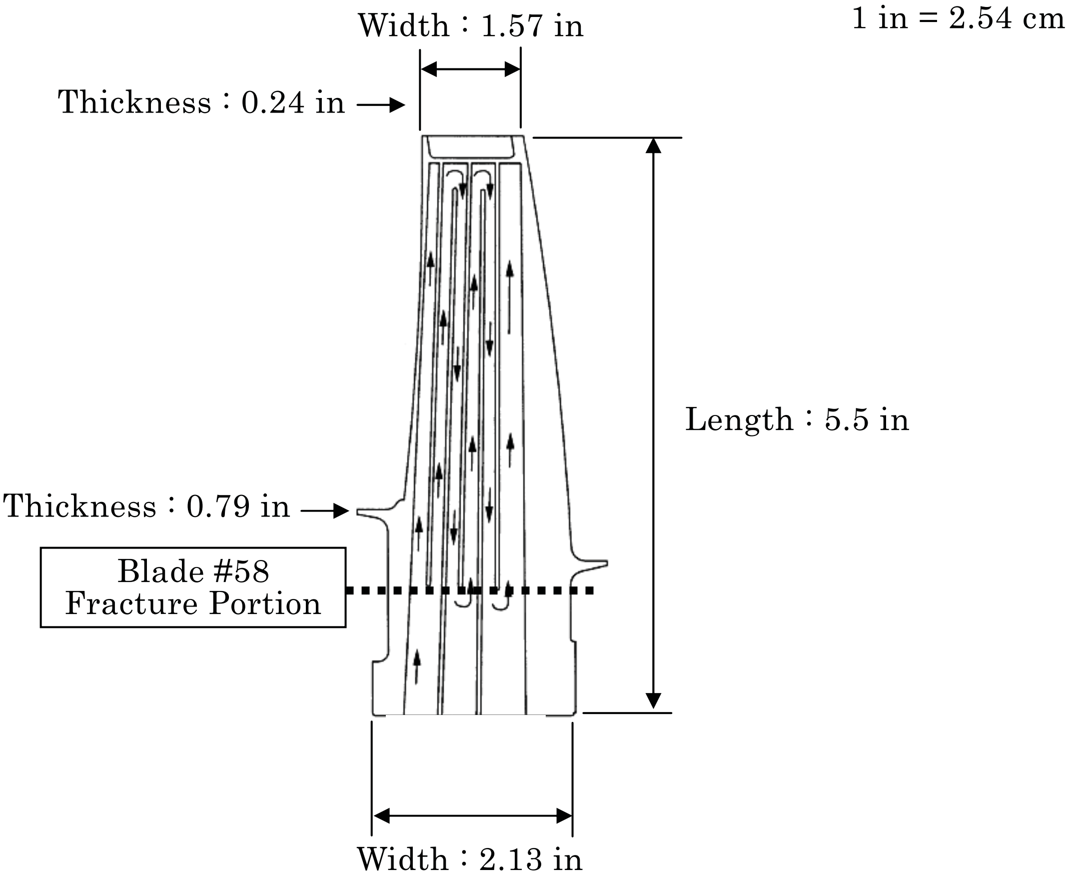



Legend :
σ = relative stress
t = relative temperature
cr = percentage of inspected parts with cracks

From engine manufacturer

# Figure 14 HPT Stage 2 Blade

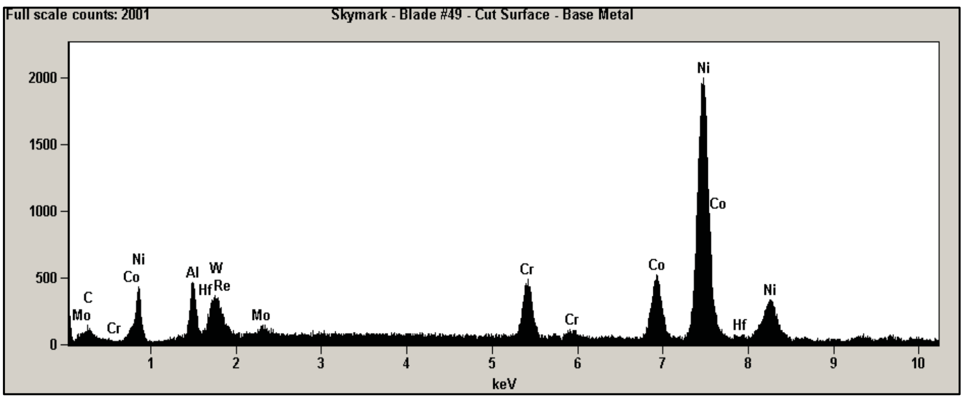
Part Number : 1881M52G05




Base Metal Spectrum − DSR142

Materials :

- Nickel (Ni)
- Cobalt (Co)
- Chrome (Cr)
- Hafnium (Hf)
- Aluminum (Al)
- Tungsten (W)
- Rhenium (Re)




From Engine Manufacturer’s Document

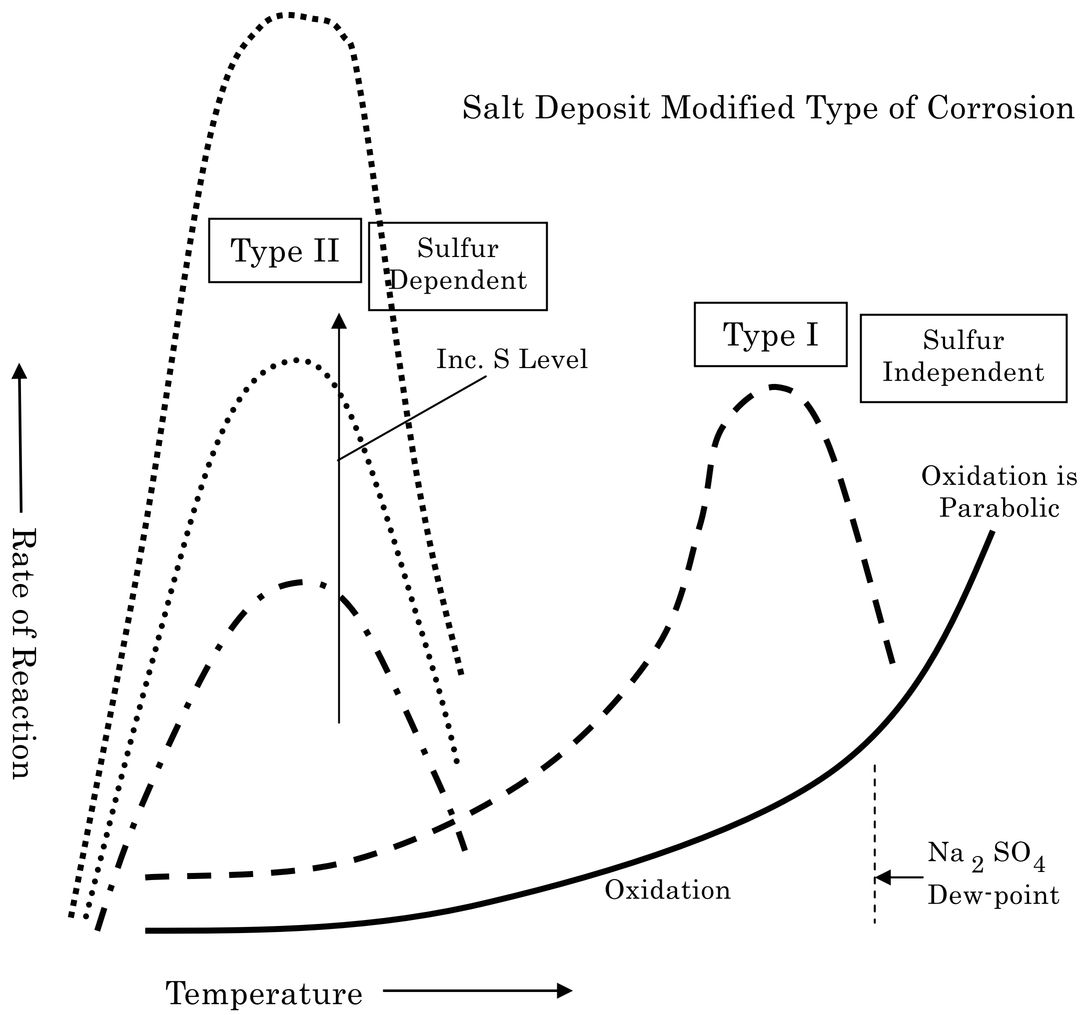
## Figure 15 Hot Corrosion and Sulfidation

Type I Hot Corrosion

- Occurs Over ~ 1550-1850°F
- Usually occurs by Basic Fluxing of Salt and Protective Oxide Scale. Ex: Sea Salt ($NaCl$) in Air Forms with Combustion Gases ($SO_3$) to Form $Na_2SO_4$
- $2Na_2SO_4 + Al_2O_3 + 1/2\ O_2 \rightarrow 2Na_2AlO_4 + 2SO_2$
- Molten $Na_2SO_4$ Dissolves the Protective Oxides of $Al_2O_3$ and $Cr_2O_3$.
- Corrosion Rate Increases with Temperature, Especially Above > 1500°F ($Na_2SO_4$ Melts).

Type II Hot Corrosion

- Occurs Over ~ 1100-1350°F
- Still Requires $Na_2SO_4$ and S.
- $SO_3$ Combustion Product forms $CoSO_4$ or $NiSO_4$ which Dissolve in $Na_2SO_4$ Forming Low Melting Eutectic Salt Causing Corrosion.
- $SO_3$ Partial Pressure Critical to Mechanism – Indicating Attach is S



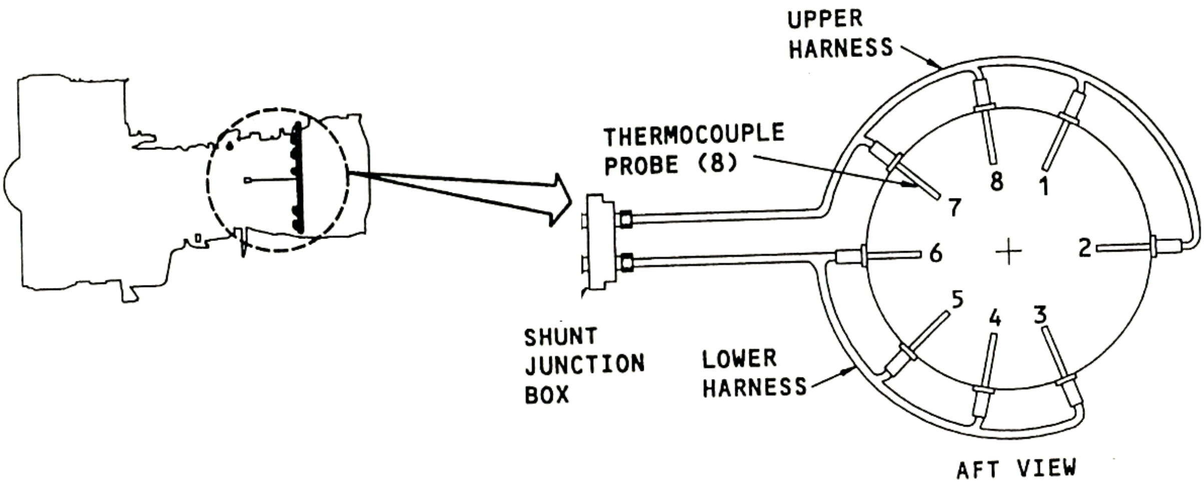

From Aircraft Manufacturer's Document

## Figure 16　Fuel Supply Tube Fracture Prevention Measure by Engine Manufacturer

Fuel Manifold Clamps


From Aircraft Manufacturer's Document

## Figure 17　EGT Probe




From Aircraft Manufacturer's Document

# Figure 18 Engine Failure Reference Procedure at Kagoshima Airport

**FLIGHT PLANNING DATA MANUAL**

*1-8-2 ENGINE FAILURE REFERENCE PROCEDURE*

## RJFK (RWY 16 / 34)

**RWY 16:** Climb via RWY track to 1300', turn RIGHT and proceed KGE, then via KGE R141 to the Pacific Ocean.

**RWY 34:** Climb via RWY track to KGE 2.7D, then turn RIGHT and intercept KGE R141 to the Pacific Ocean.

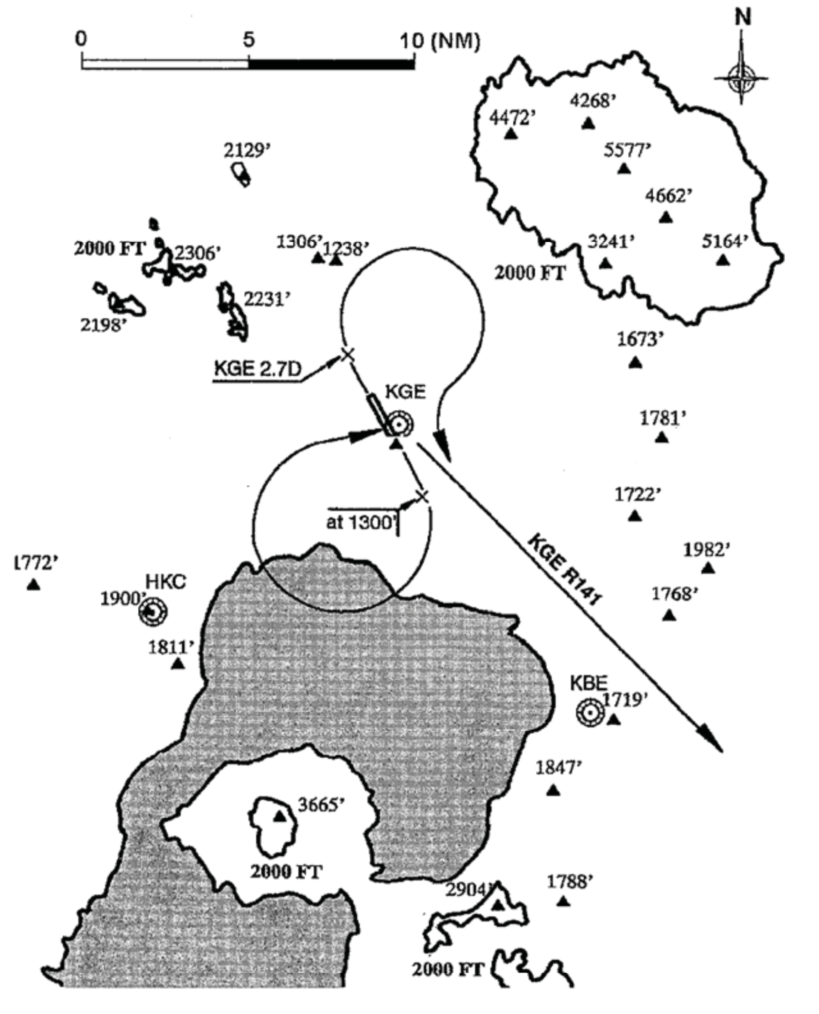
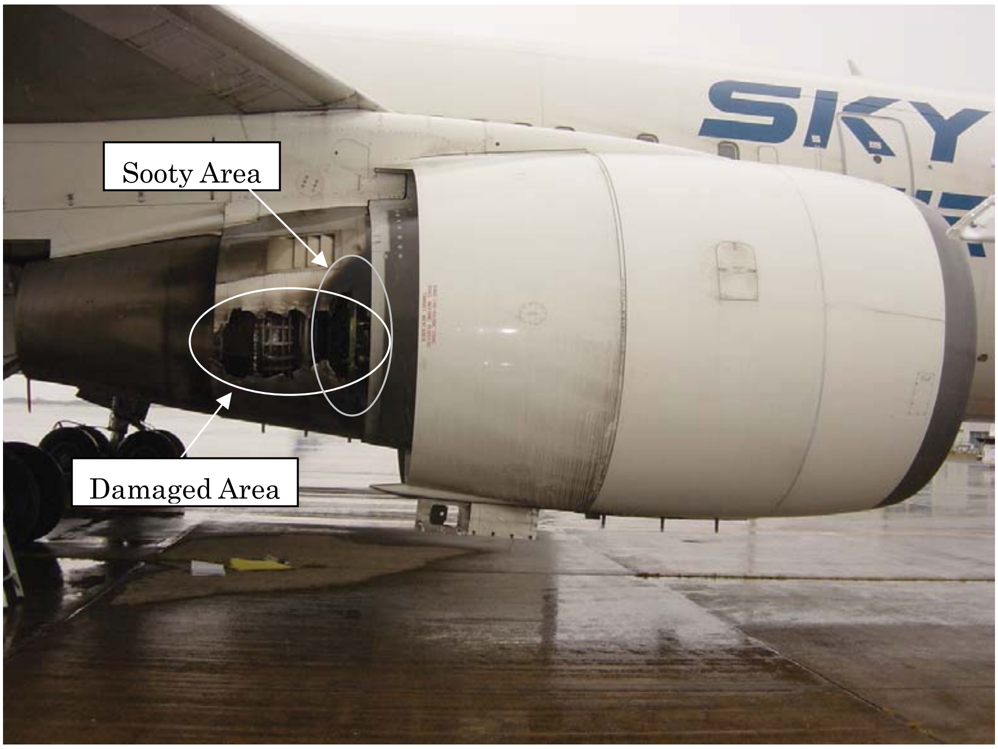
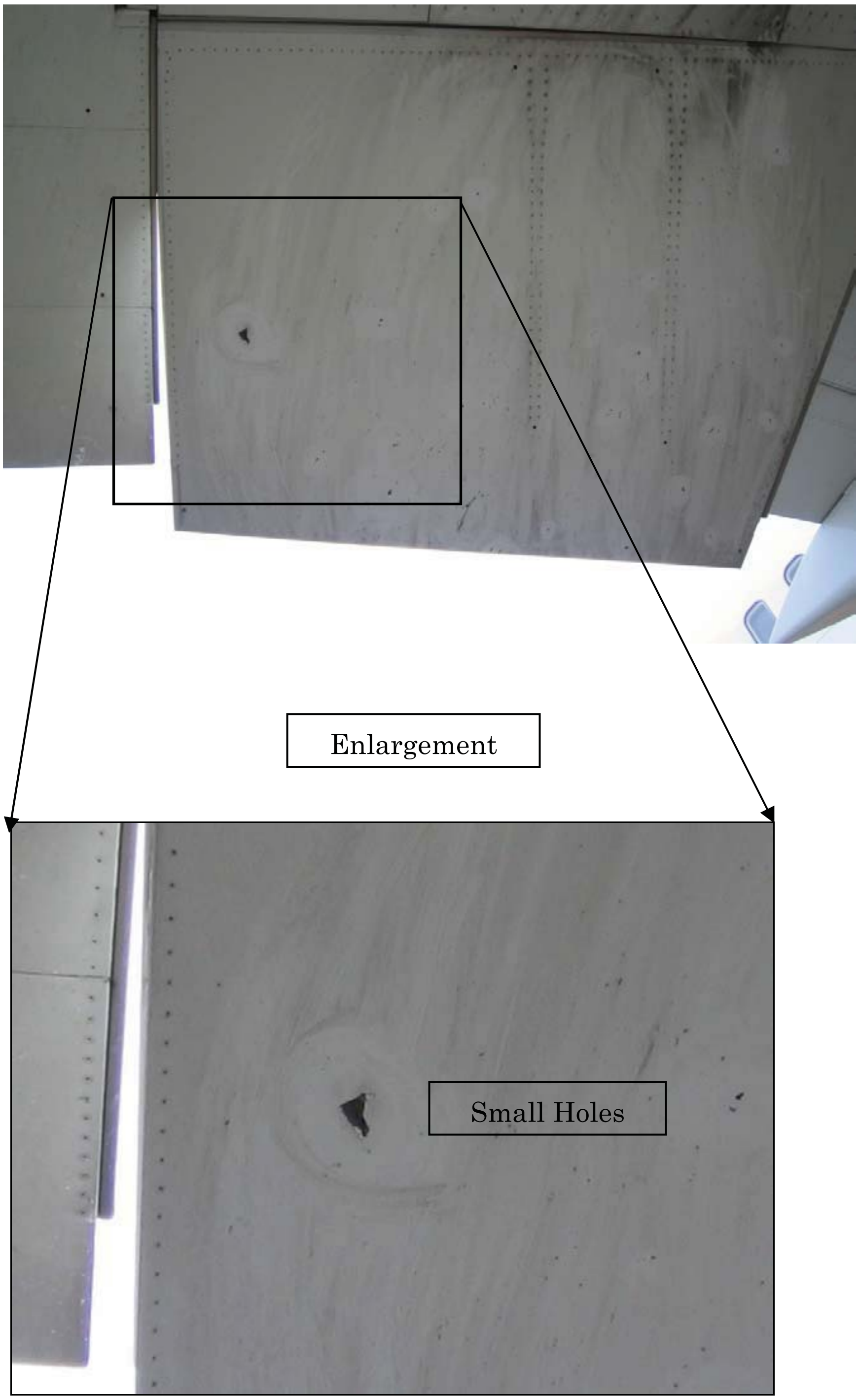


Photo 1　The Serious Incident Aircraft


Photo 2　Right Engine



# Photo 3　Damaged Inboard Aileron (Right Wing)



## Photo 4　Fire Kindled by Engine Fragments

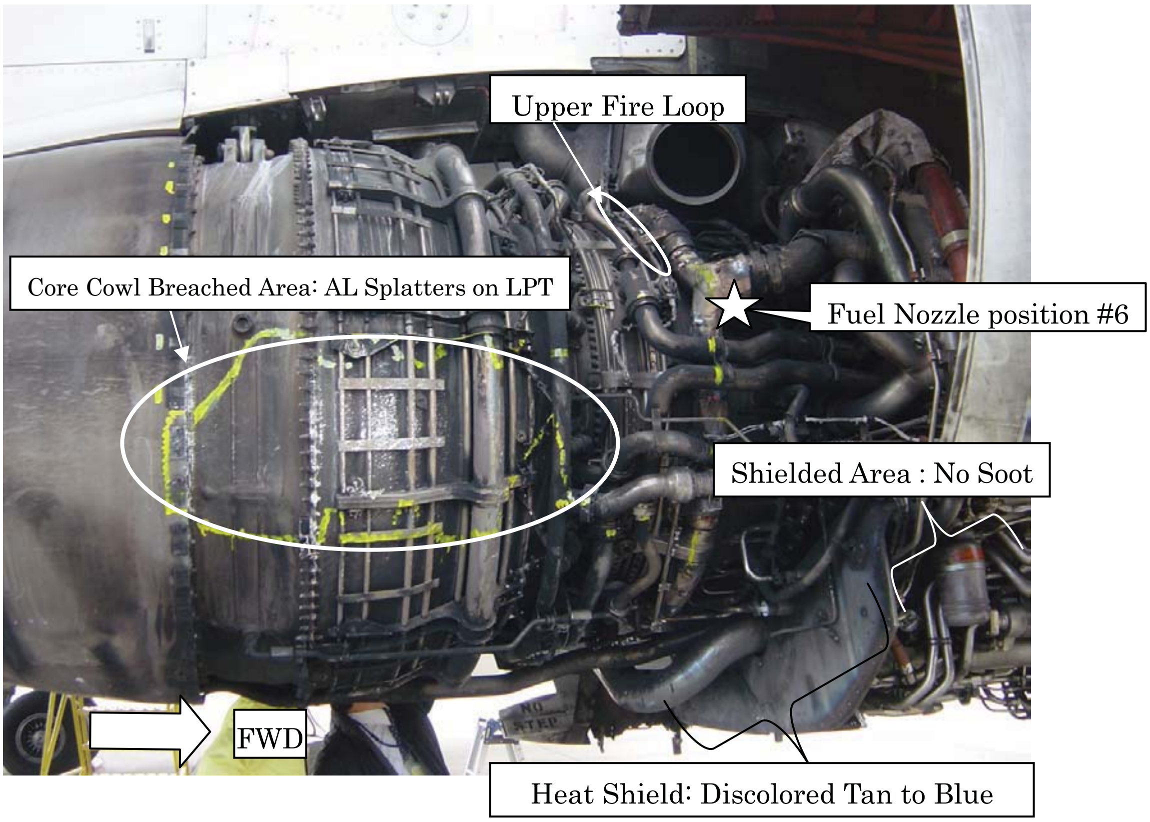

Provided by Kagoshima Airport Office

## A Fragment of the Engine Blade on the Runway


Provided by Kagoshima Airport Office

Photo 5 Burnt Grass Area

## Photo 6 Right side of the Engine




## Photo 7 Heat Shield and Lower Fire Loop

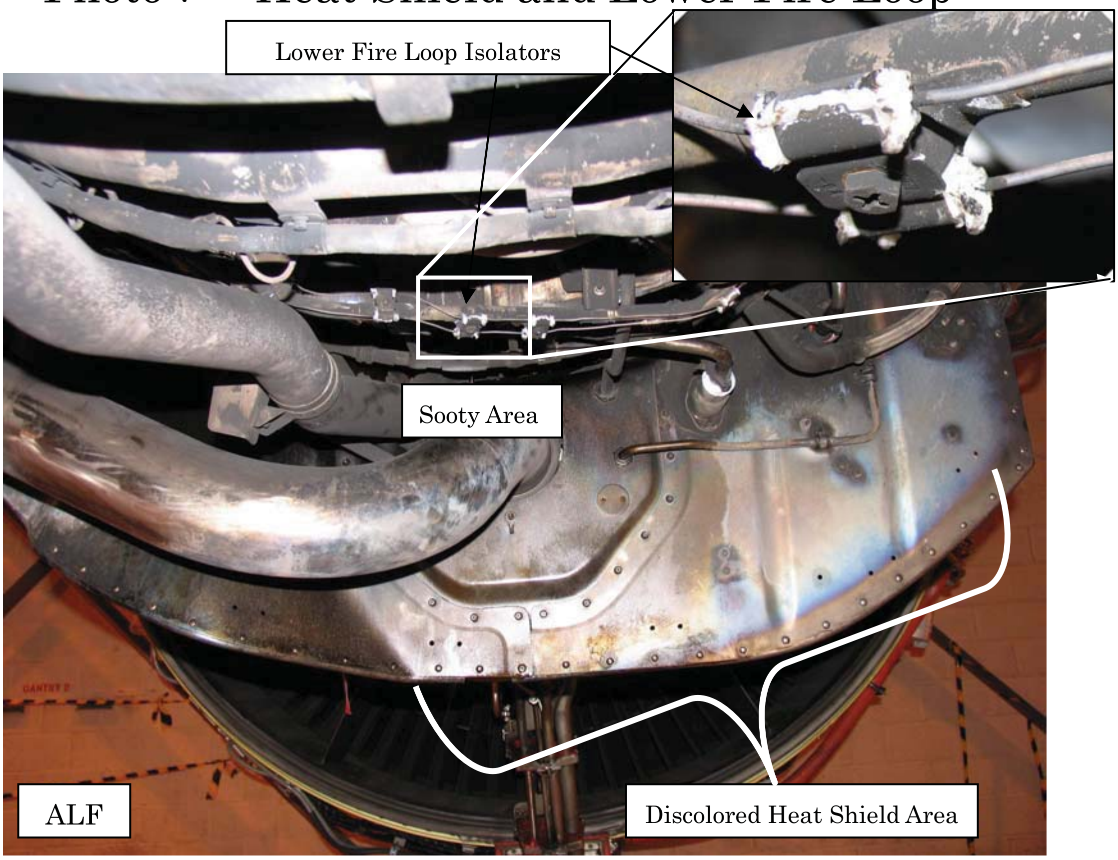


Photo 8 Right Blowout Door

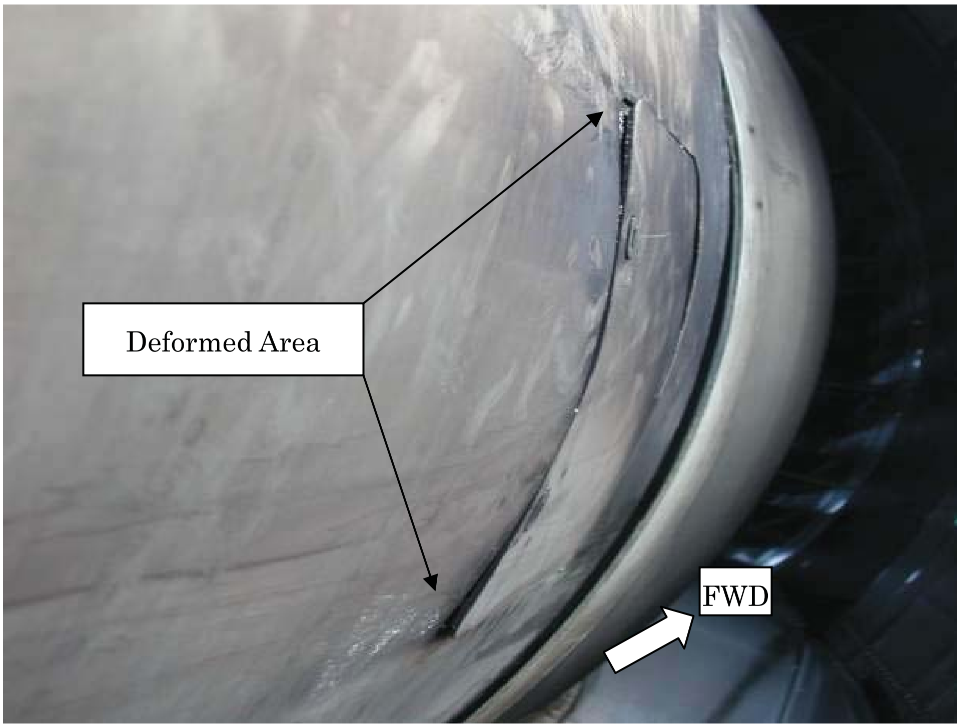



Photo 9 A Hole of LPT Case

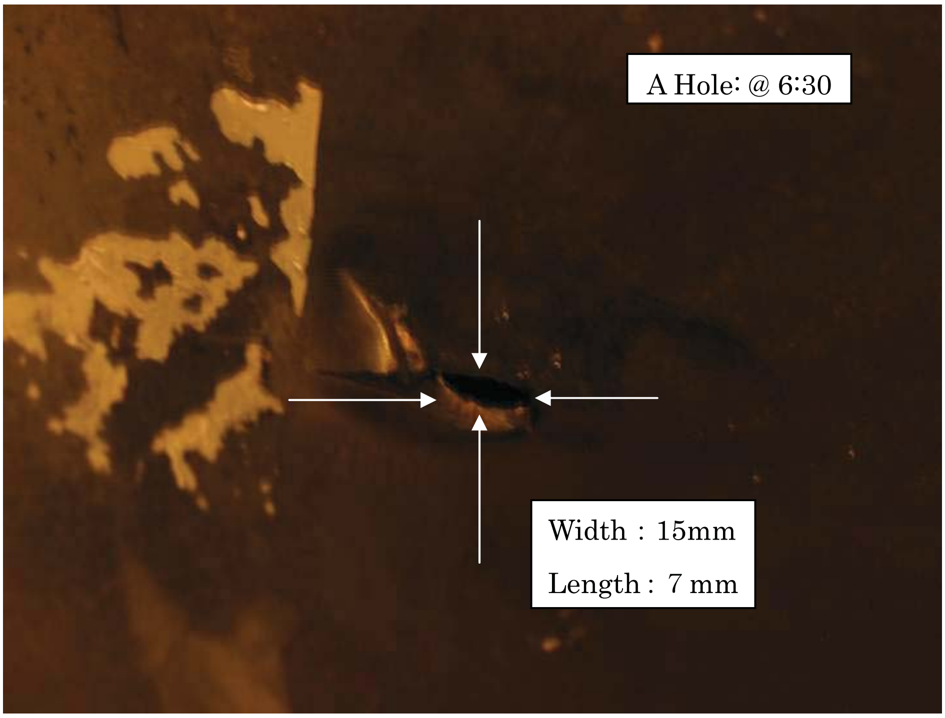


Photo 10　The Burnt Core Cowl

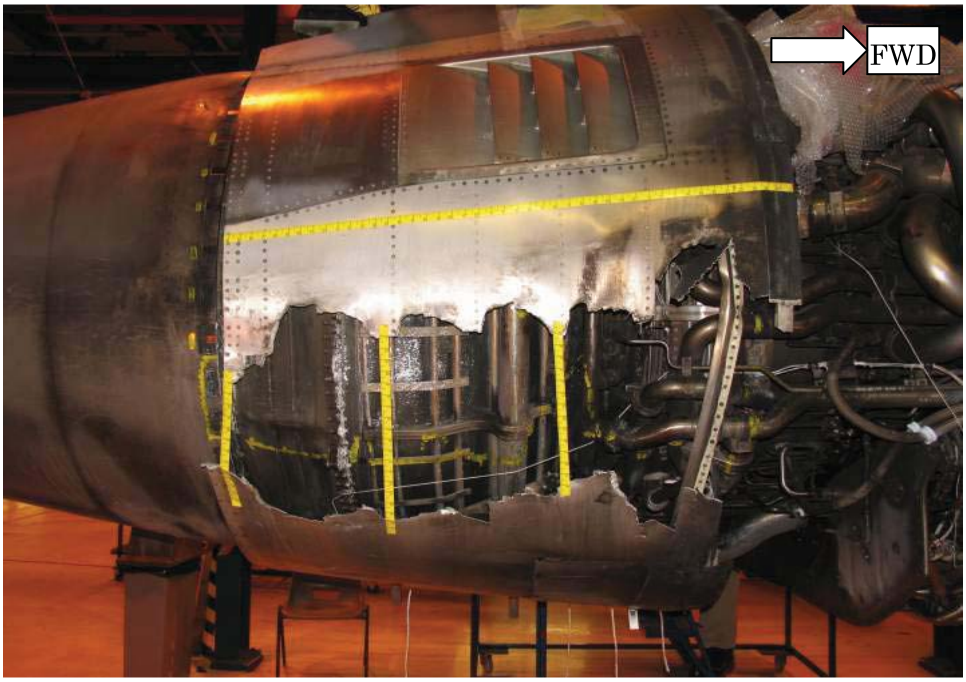



Photo 11　Burnt Area (took off the Core Cowl)

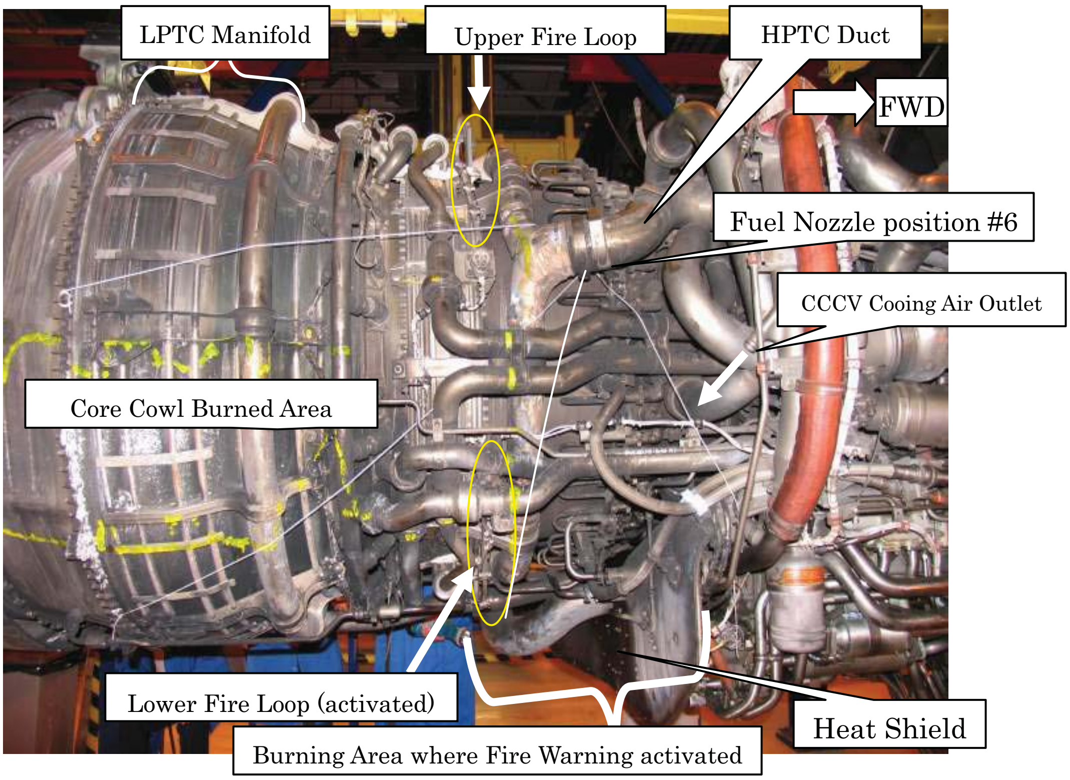


Photo 12　Fuel Manifold Position #6 Tube Separated and Dislodged from Shroud Can

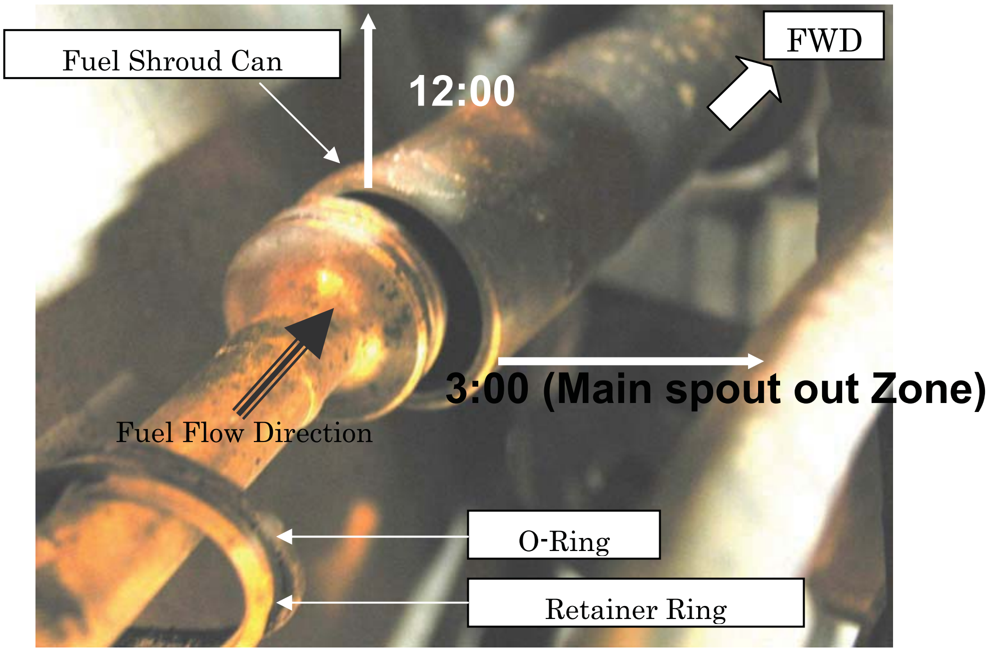
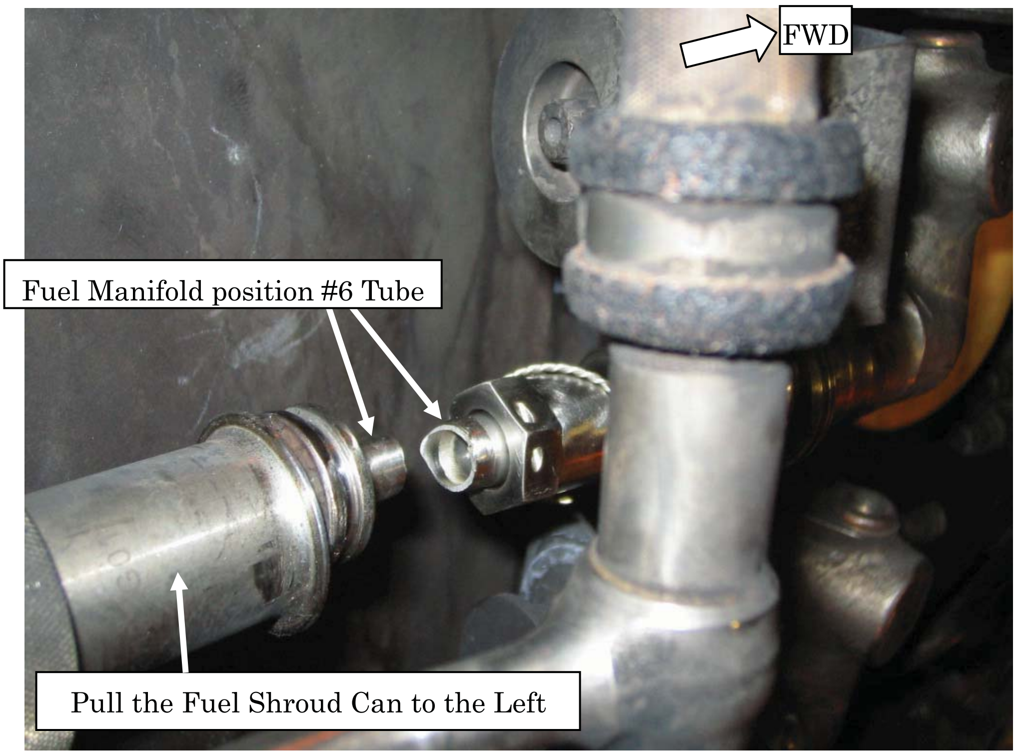
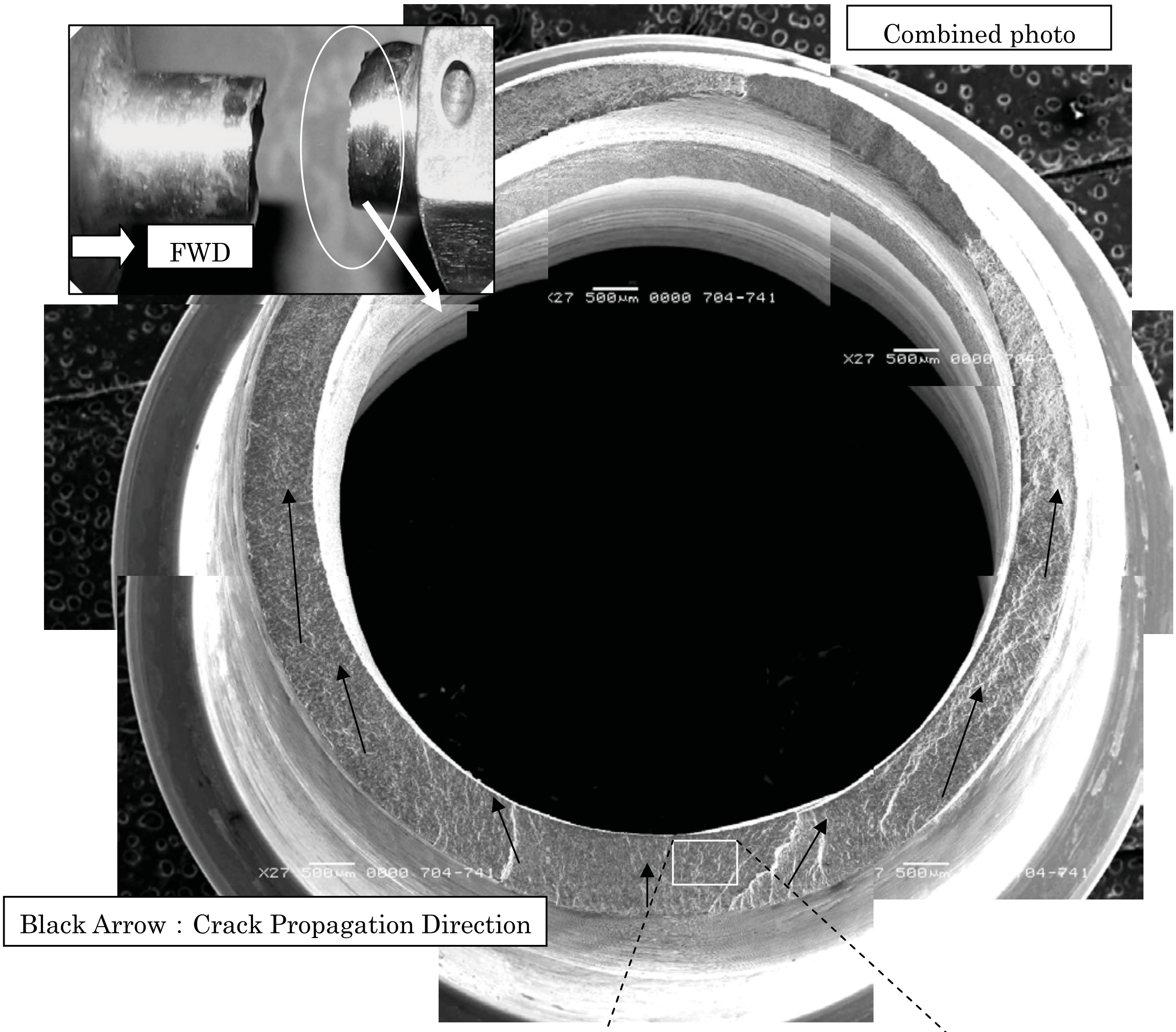



Photo 13　Separated Fuel Manifold Position #6 Tube

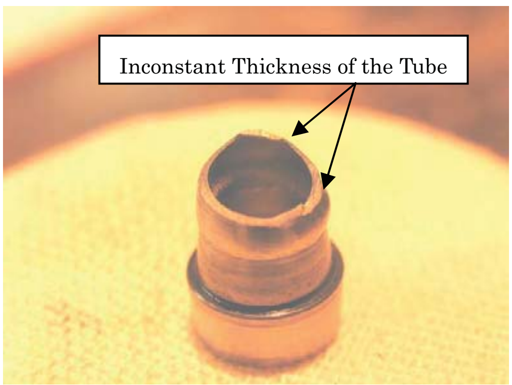


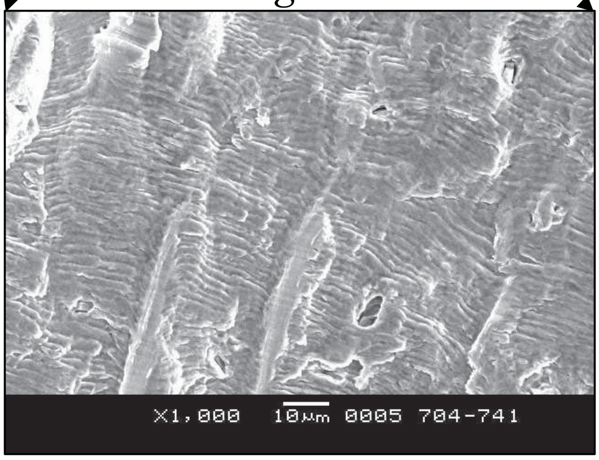
# Photo 14 Fractured Surface of Fuel Manifold Position #6 Tube



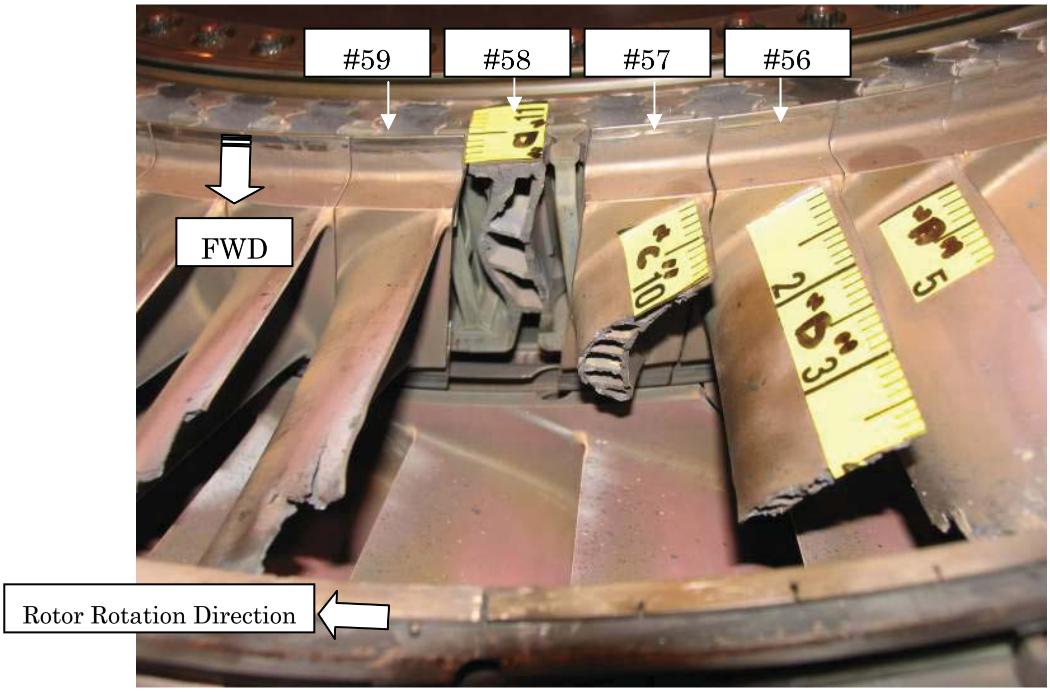

Enlargement

Traces of Fatigue Striations



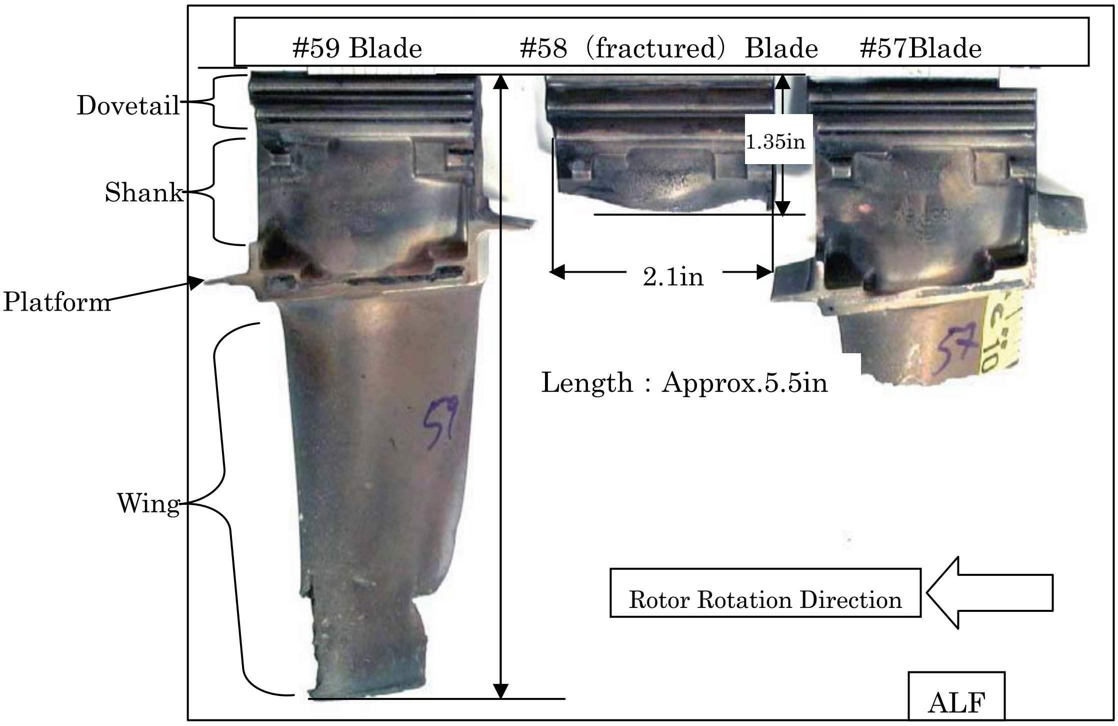



# Photo15　Stage 2 HPT Shank Separation and Adjacent Blade Damage

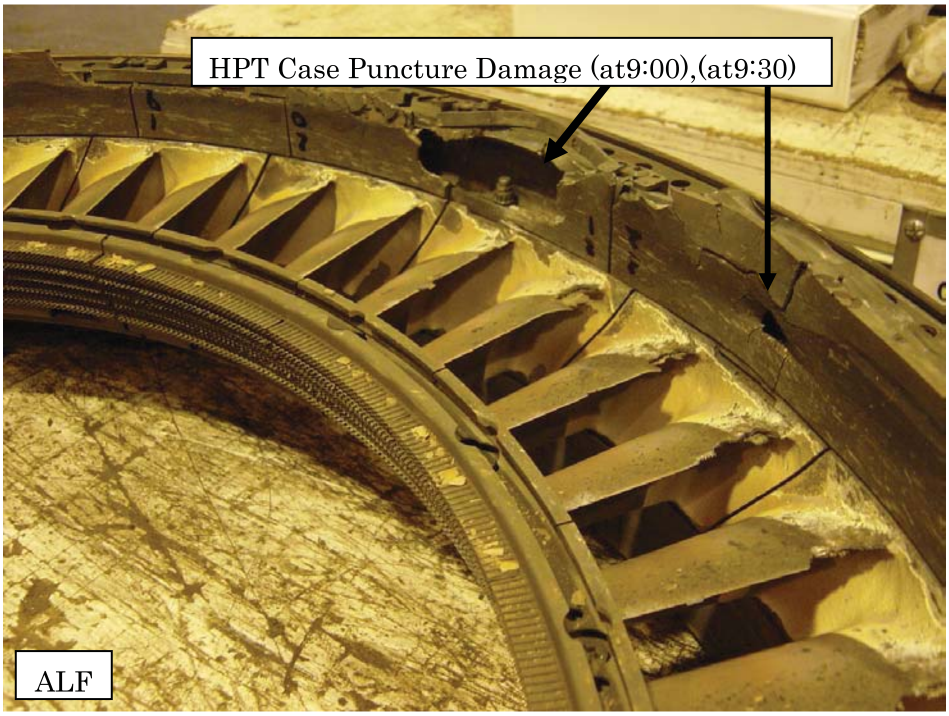



LAF

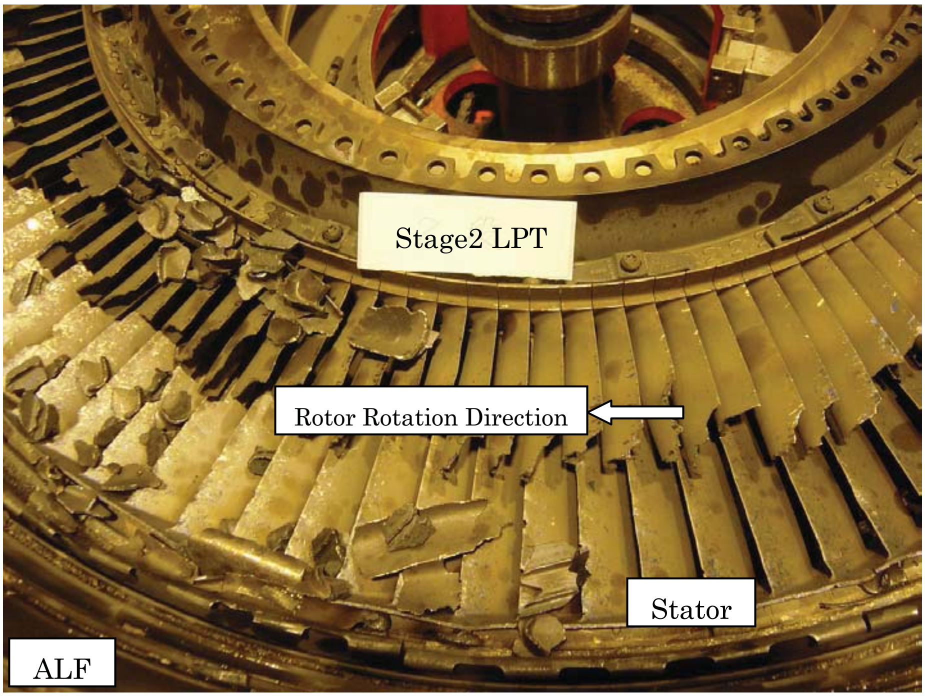
## Over View



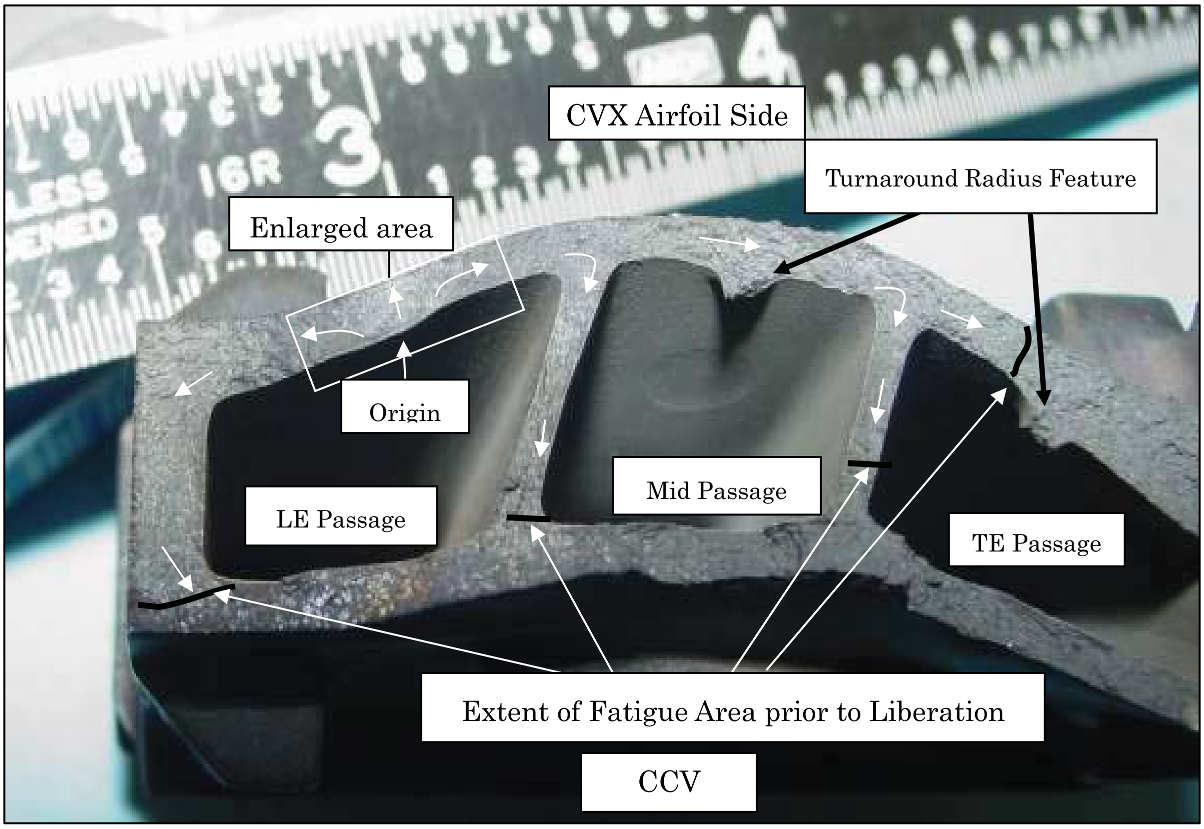
# Photo16　Damage of HPT Case



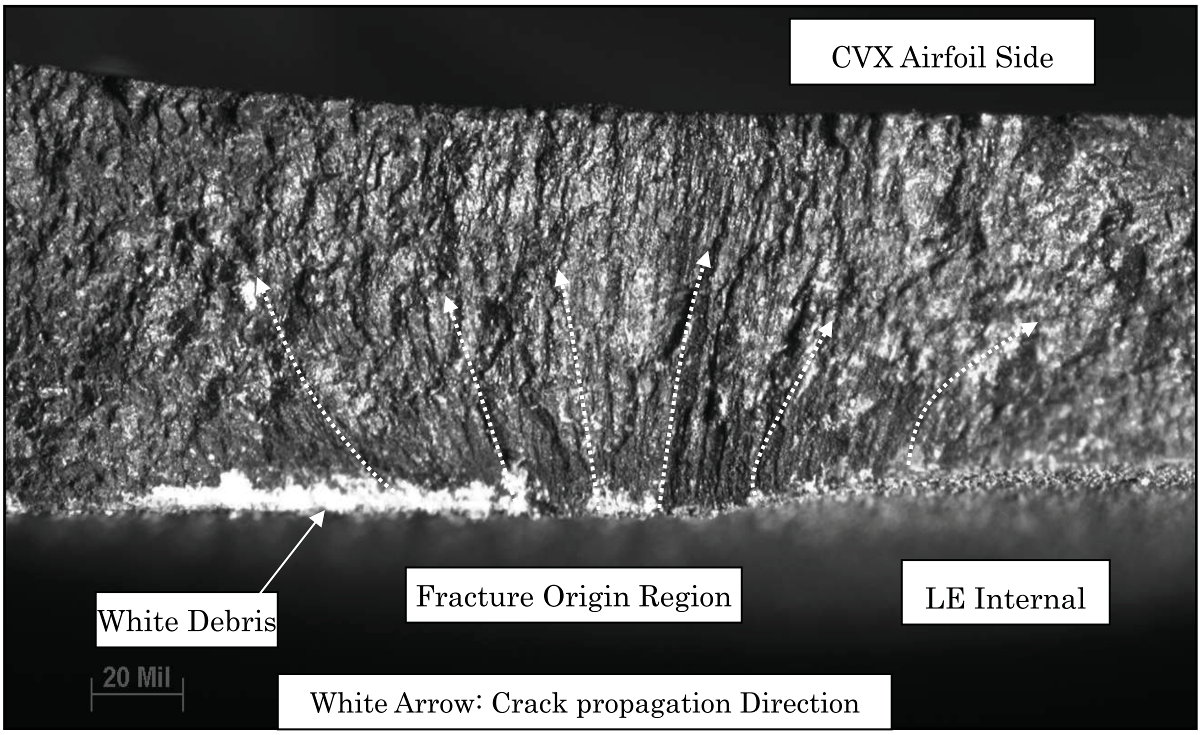

# Photo 17　Damage of LPT Blade and Stator



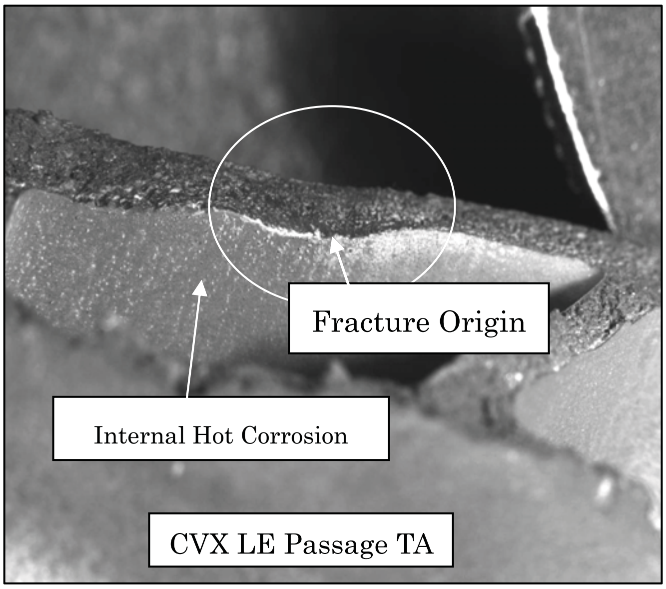
# Photo 18 Fracture Origin of #58 Blade and Internal Cooling Air Passage



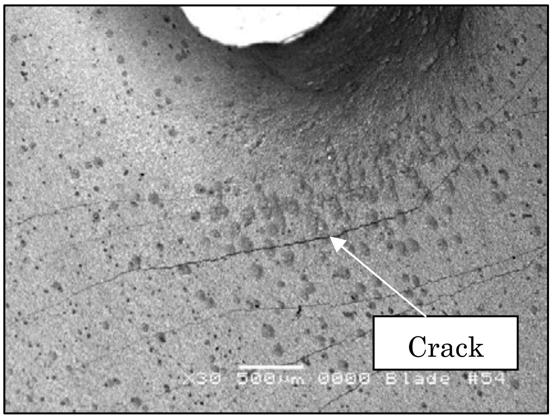

# Fracture Origin Region (Enlarged Area)

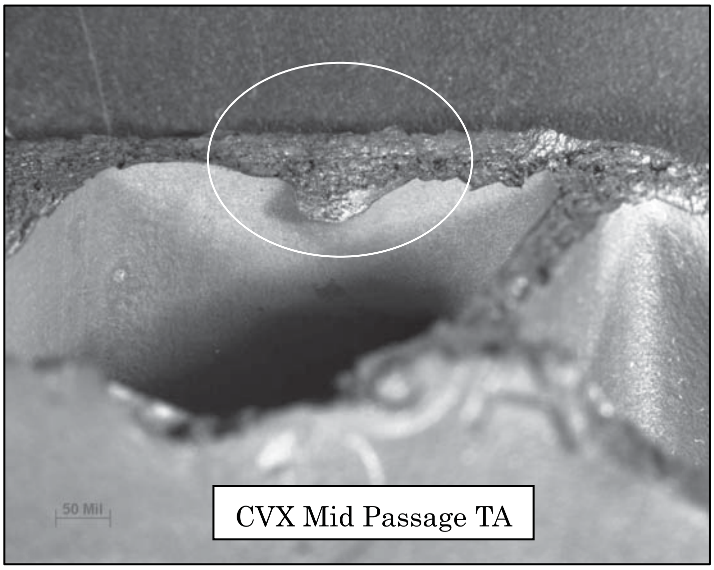


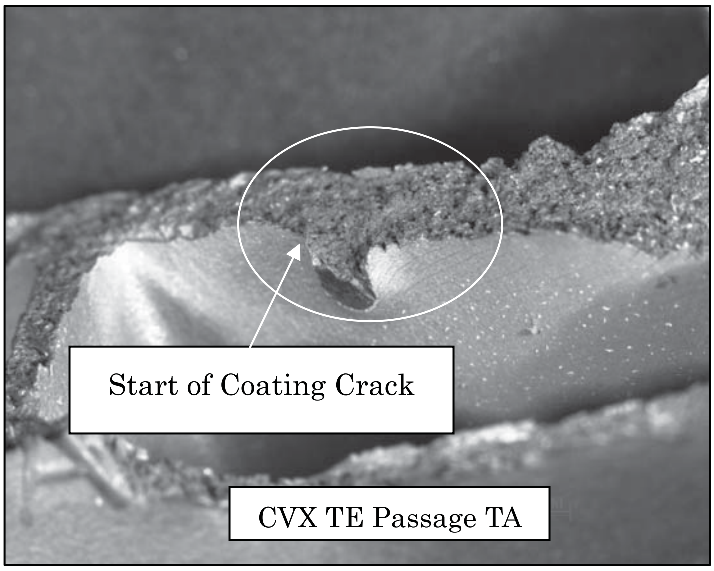
# Photo 19　TA Radius Feature of Blade #58



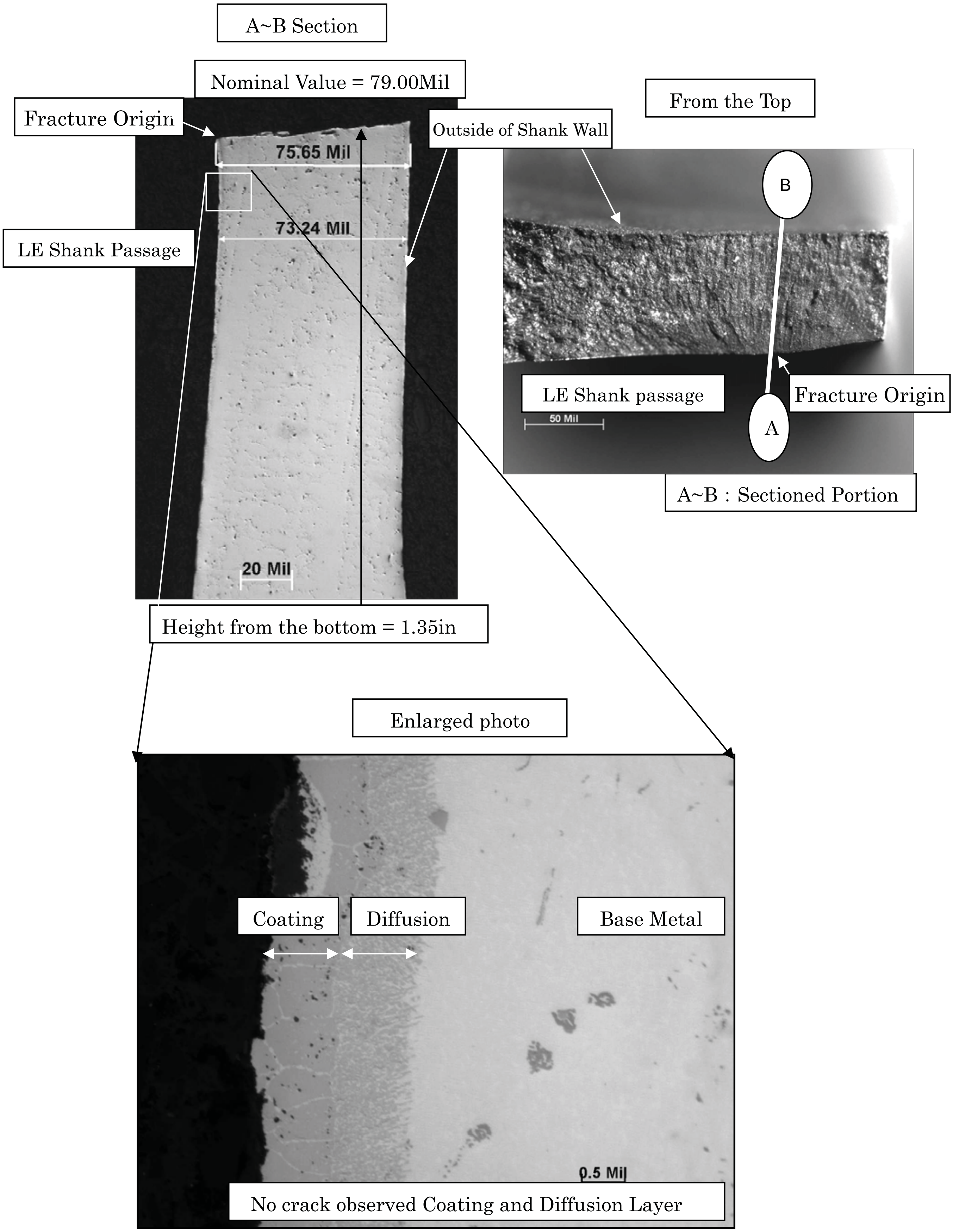

Ref : #54 Blade



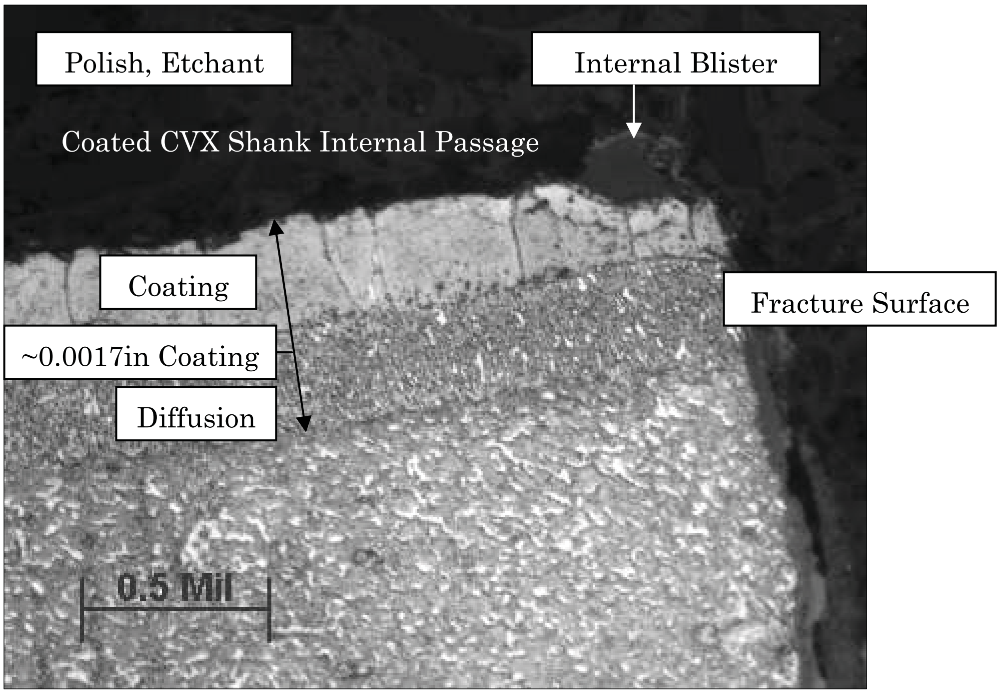



# Photo 20 CVX Shank wall Thickness of #58 Blade



# Photo 21 Blade #58 Primary Origin Region




# Blade #58 Fracture Surface with Alloy Depletion

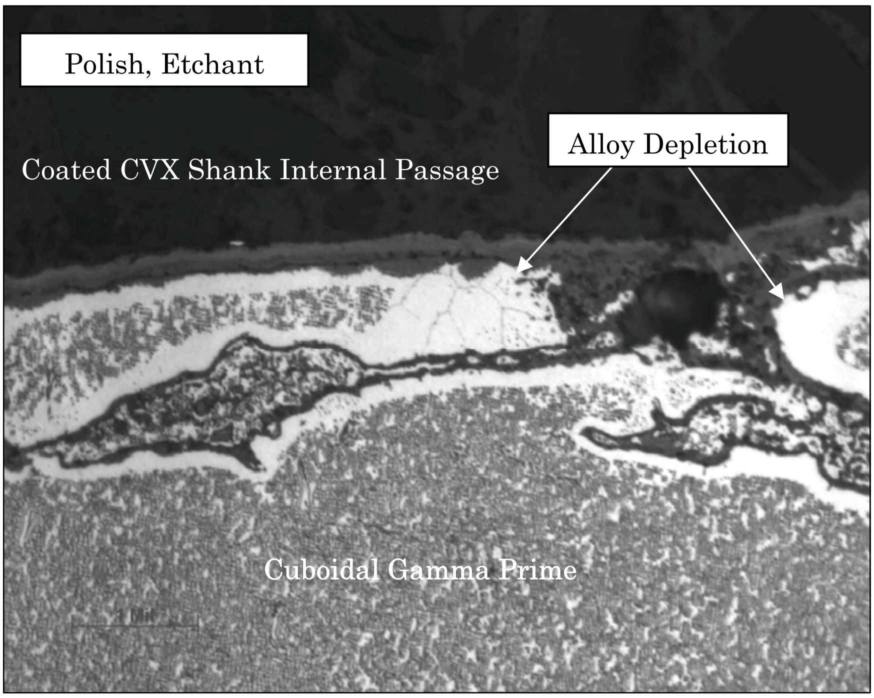
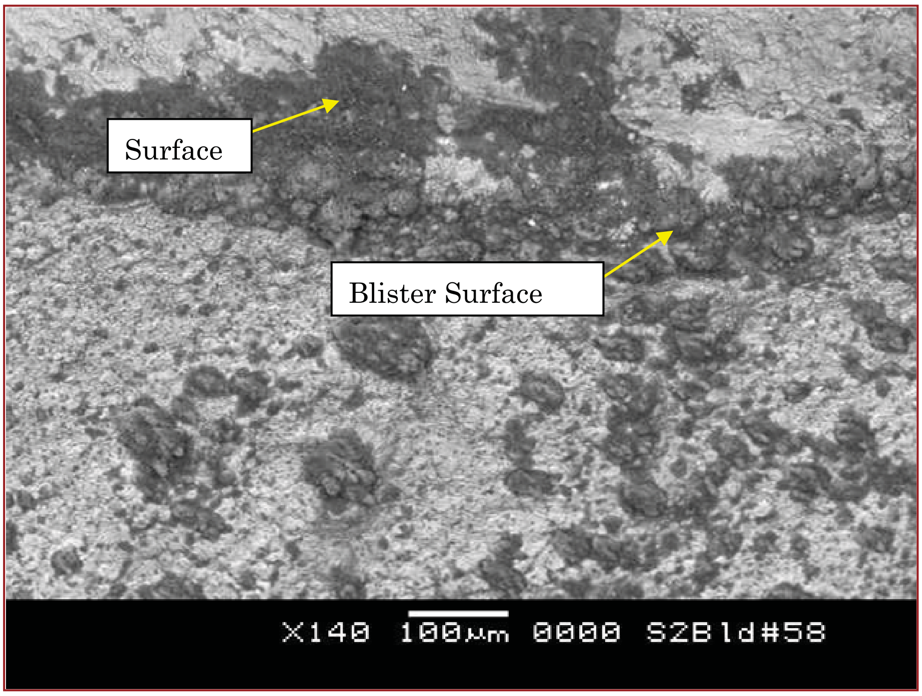
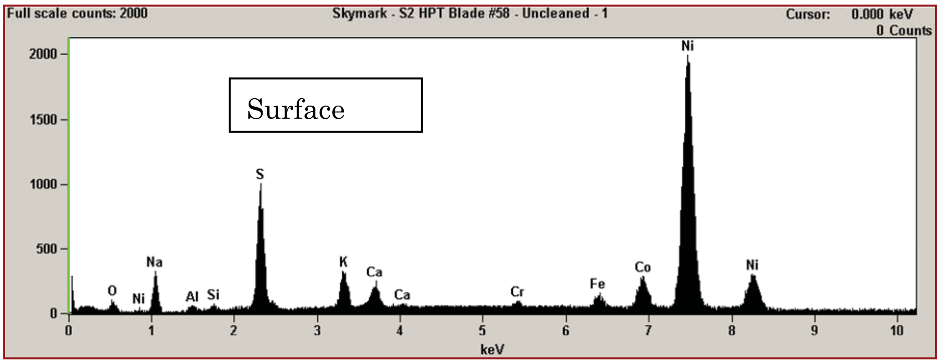
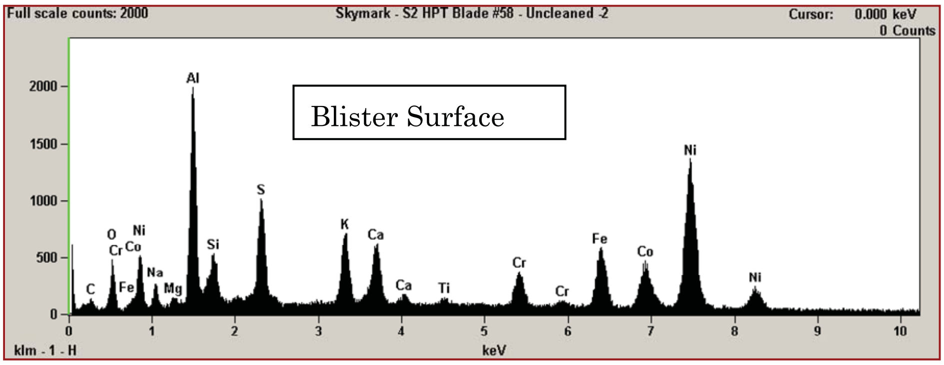


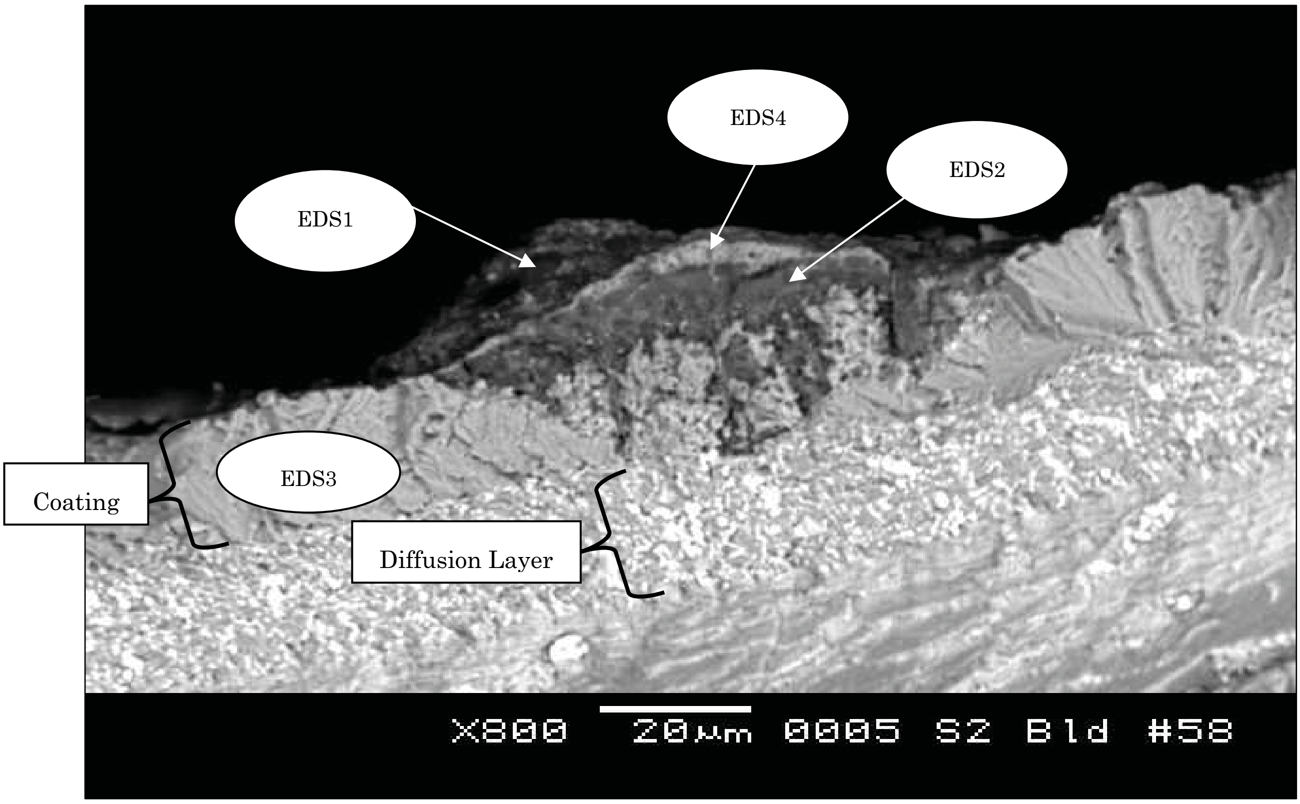
Photo22　Ingredient Analysis of Surfaces of Blades and Blisters of　Fracture Origin Area of #58 Blade



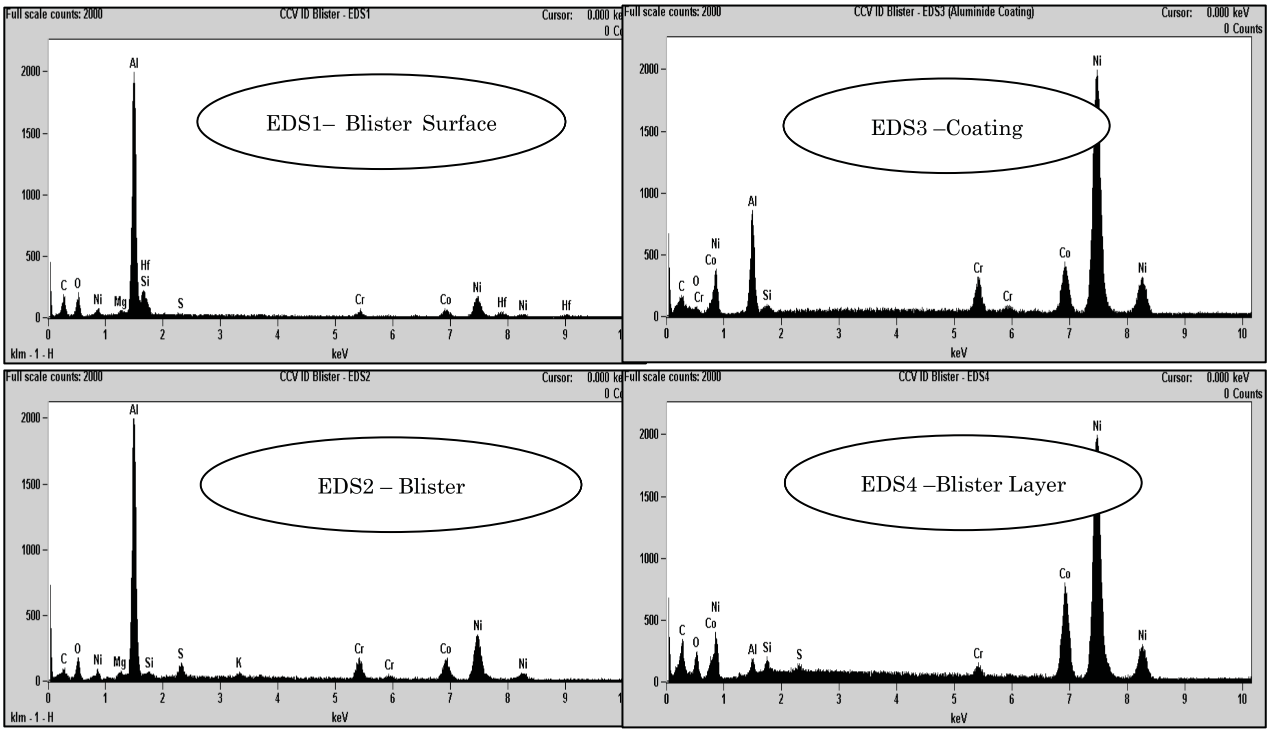



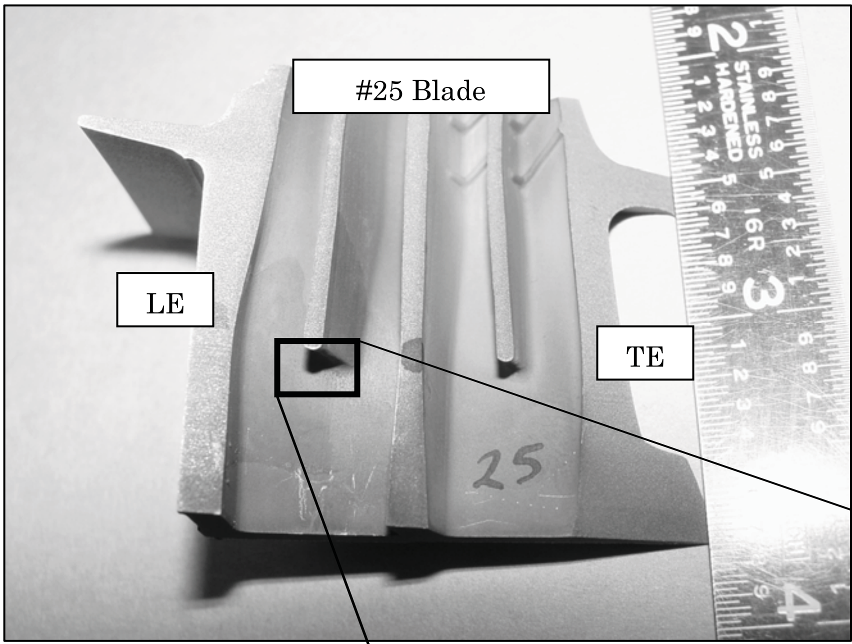
## Photo23　Blade #58 Hot Corrosion Blister on Internal Coating



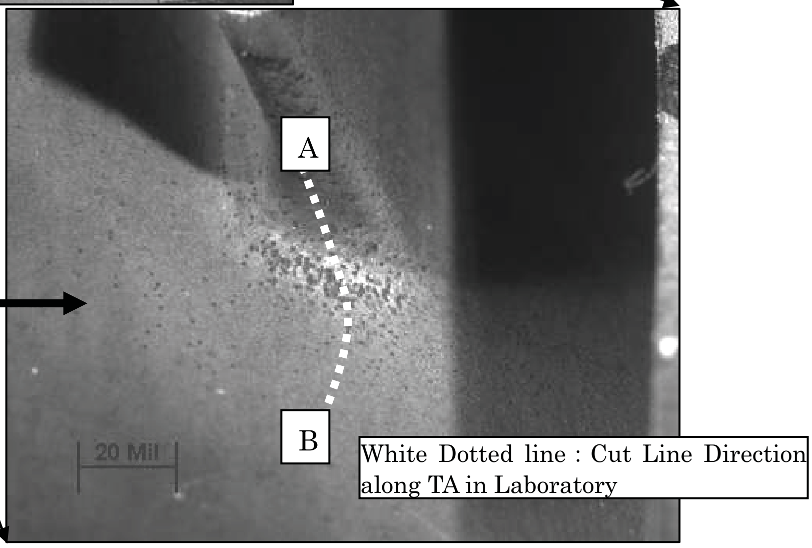

## Hot Corrosion Blister EDS Spectrum

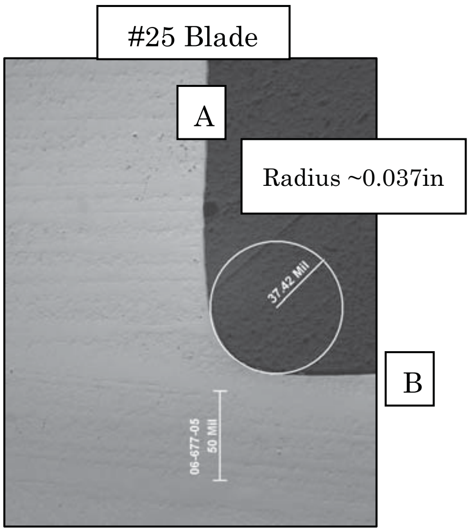


# Photo 24　Differences of Turnaround Radius Features

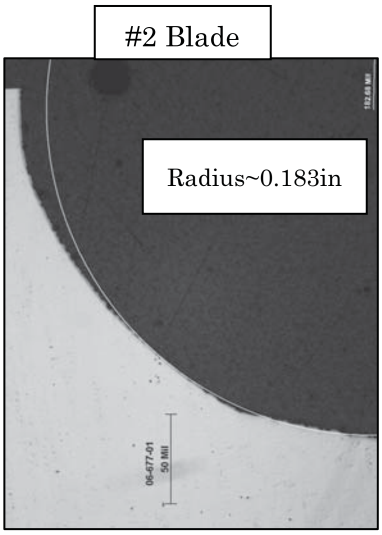



Viewing from this direction



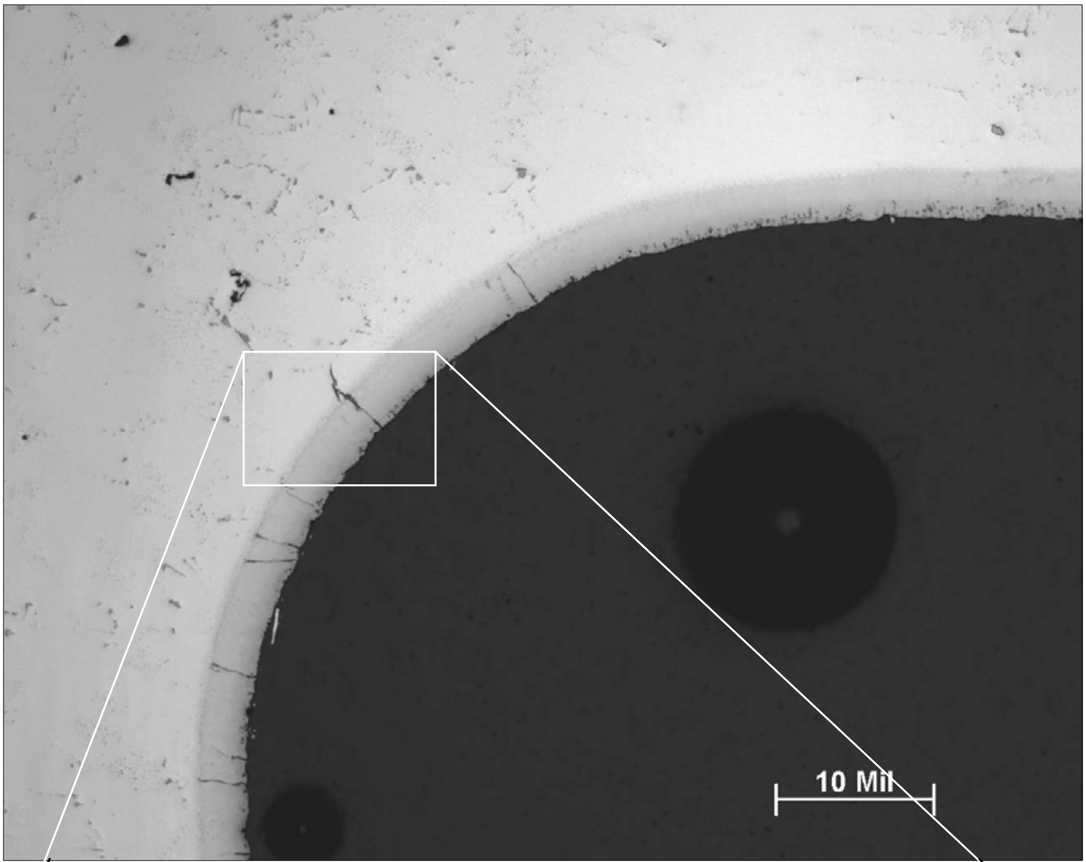




The Company A made




The Company B made

# Photo 25　A Crack of #20 Blade CVX Mid TA




Enlarged photo

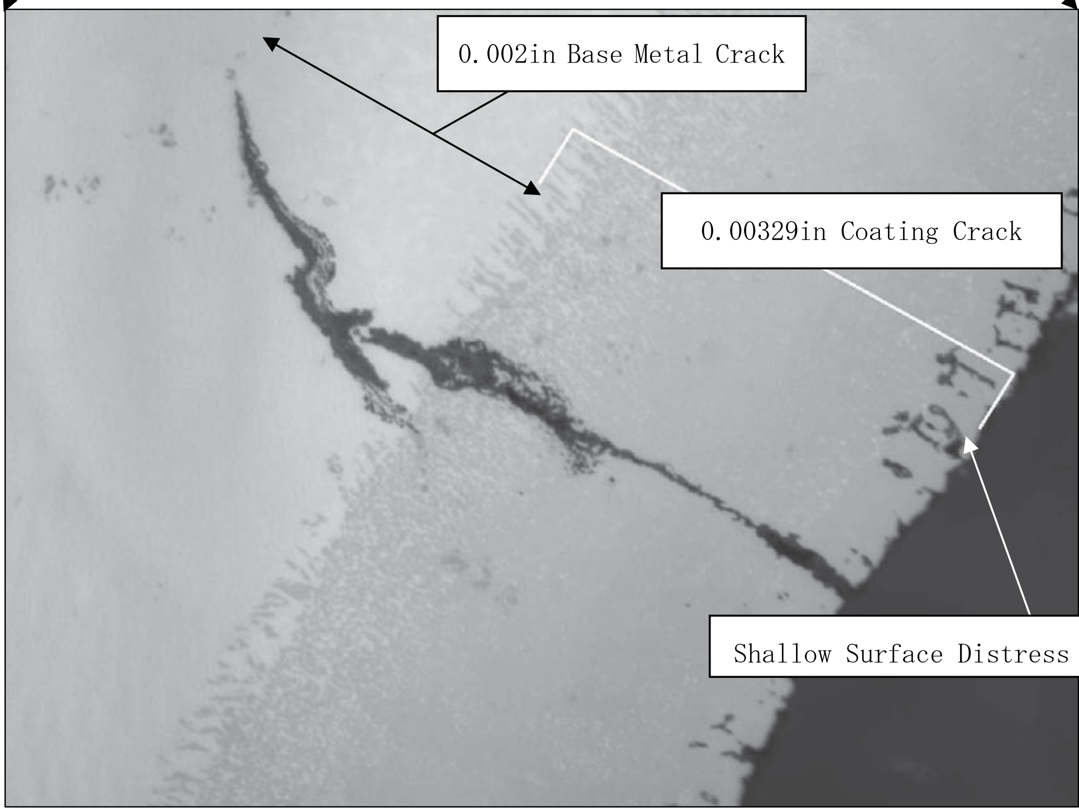
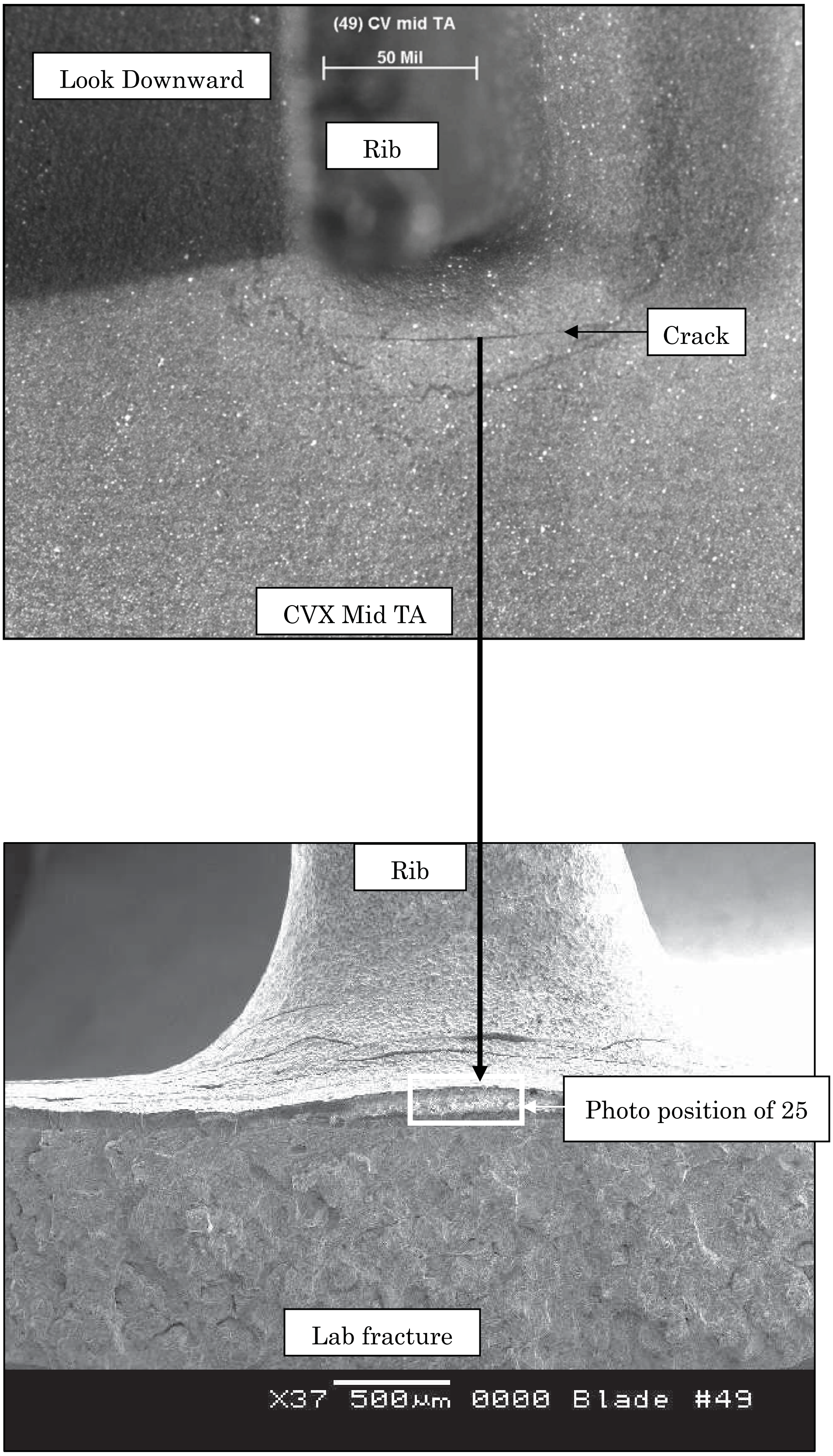
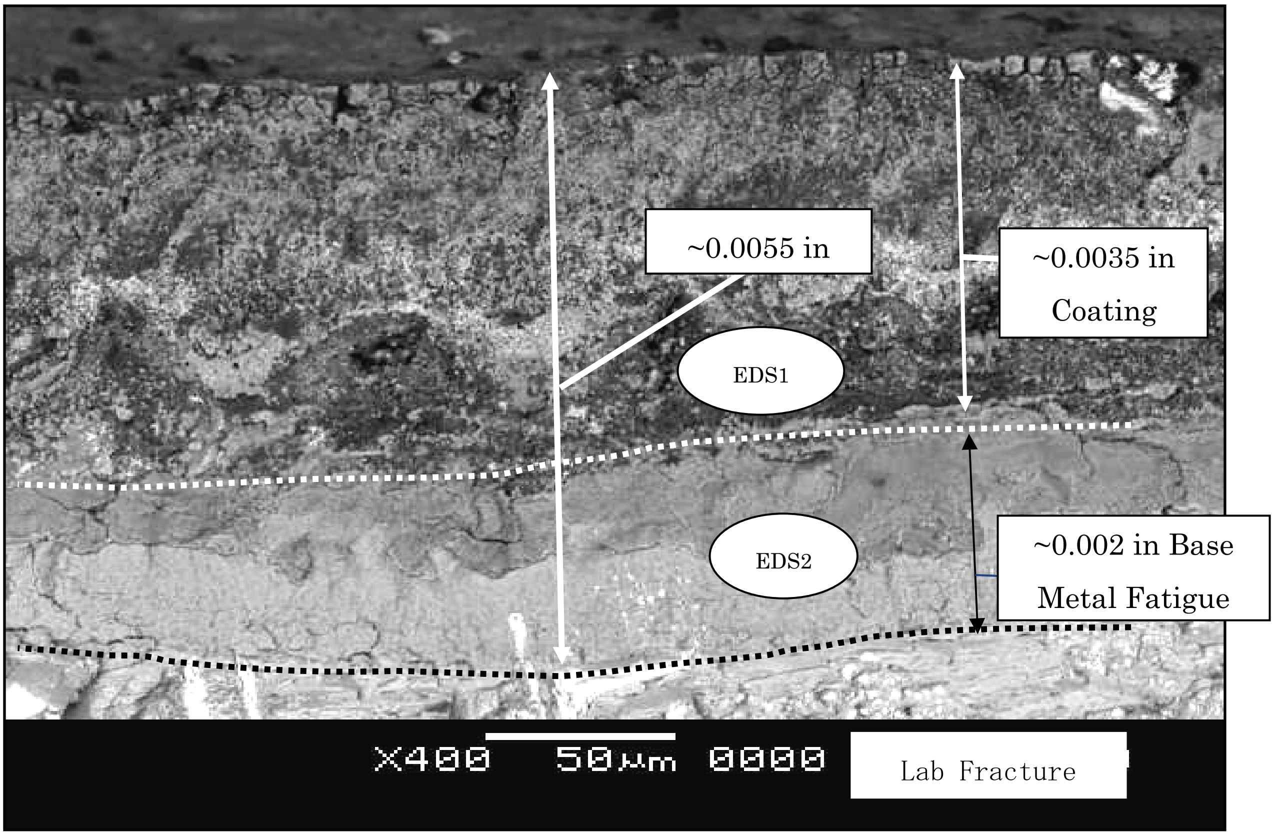
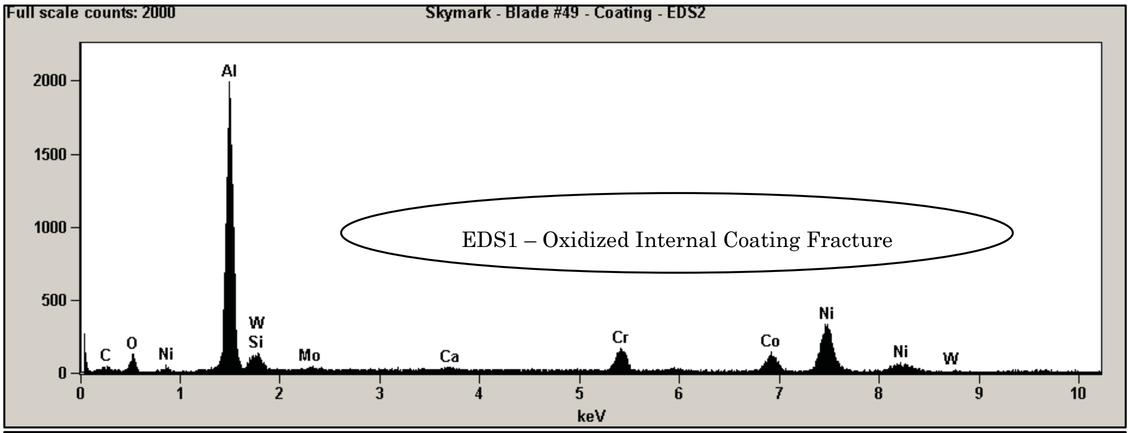


Photo26　A Crack of TA Radius Feature
#49 Blade CVX Mid Inner Passage



# Photo27　A Crack Surface of #49 Blade Base Metal



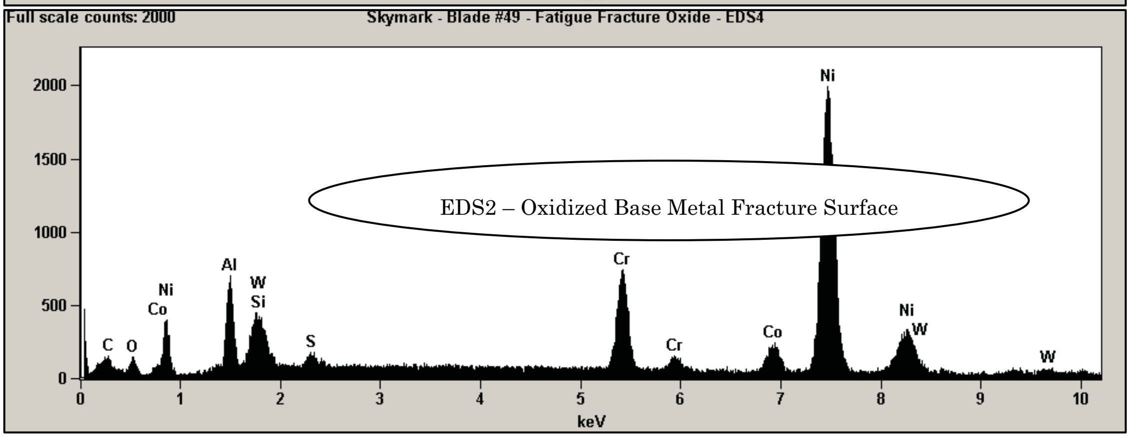



Photo 28　White Deposits on the HPT Blades

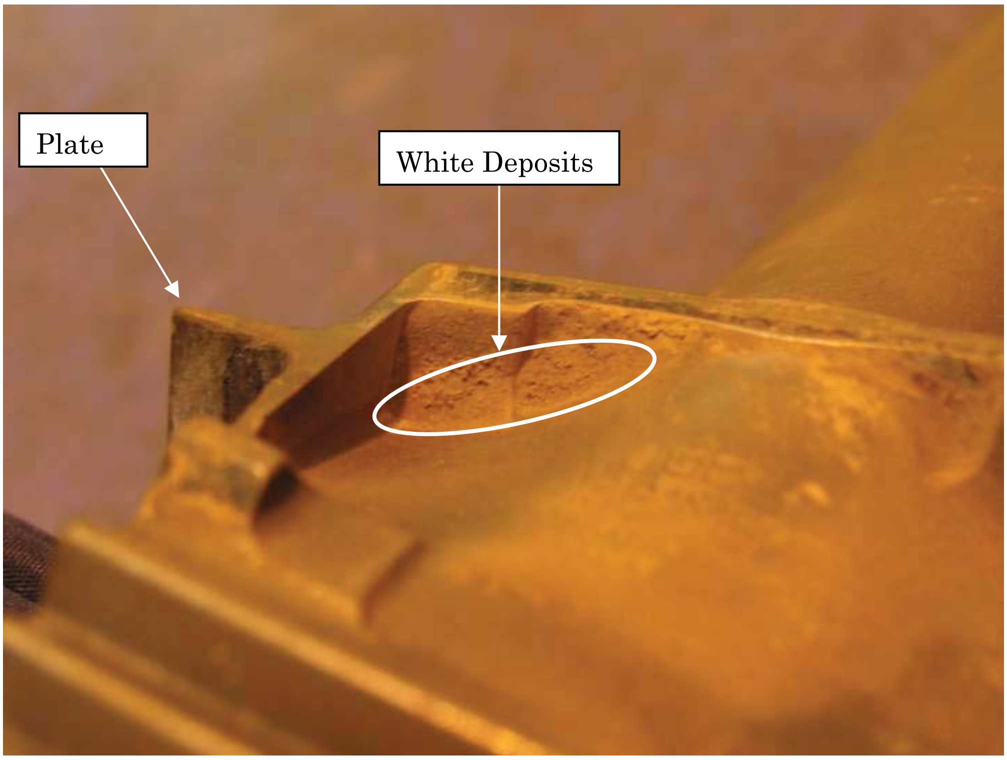



Photo 29　White Deposits on the HPC Vanes

Photo 30　White Deposits on the HPC Blades

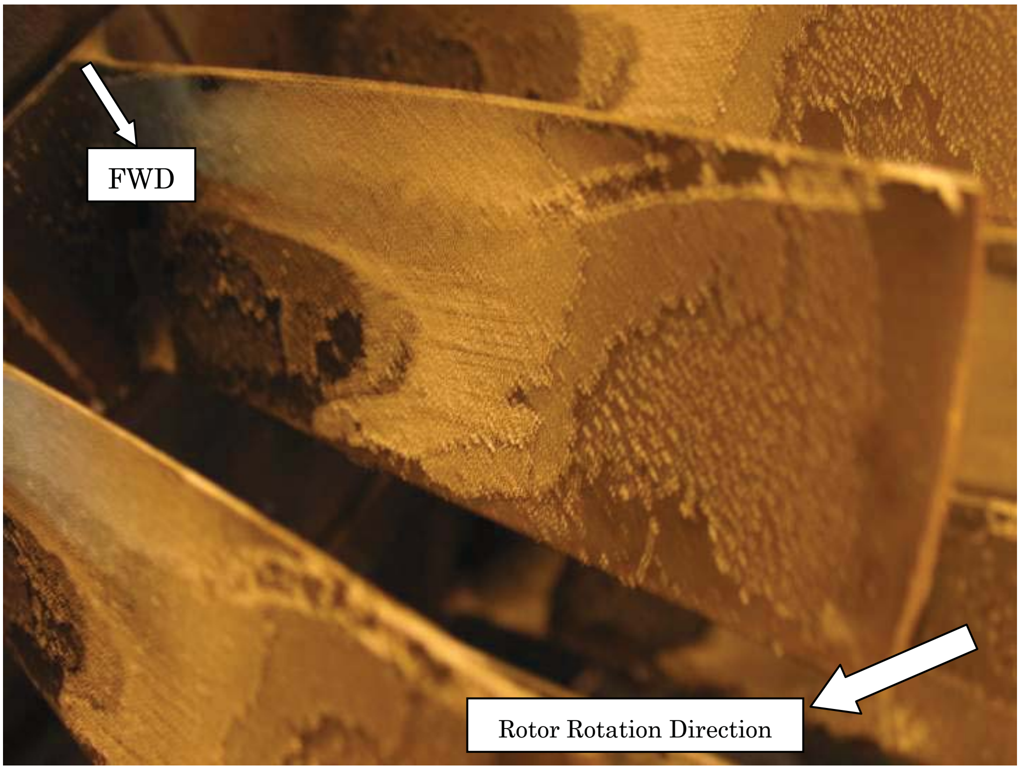
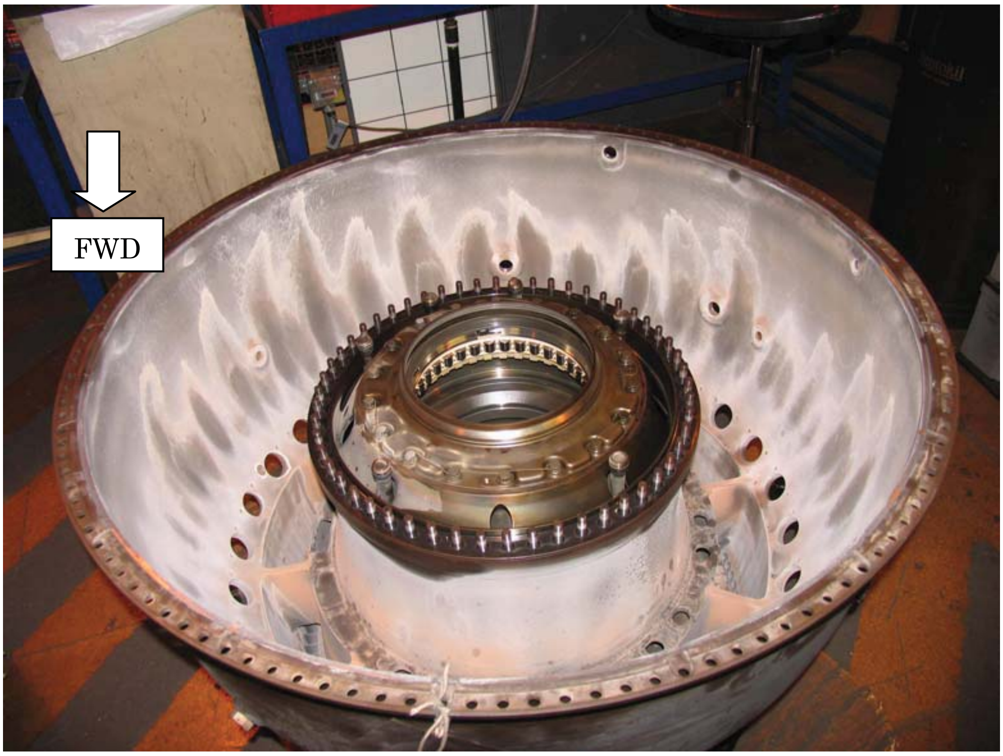
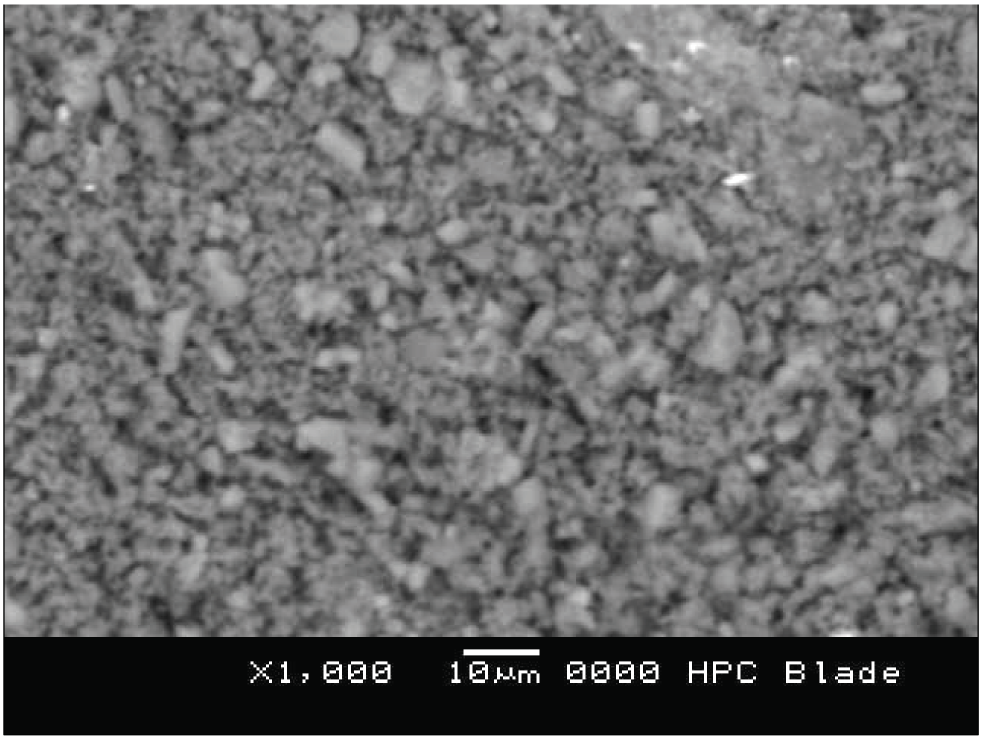
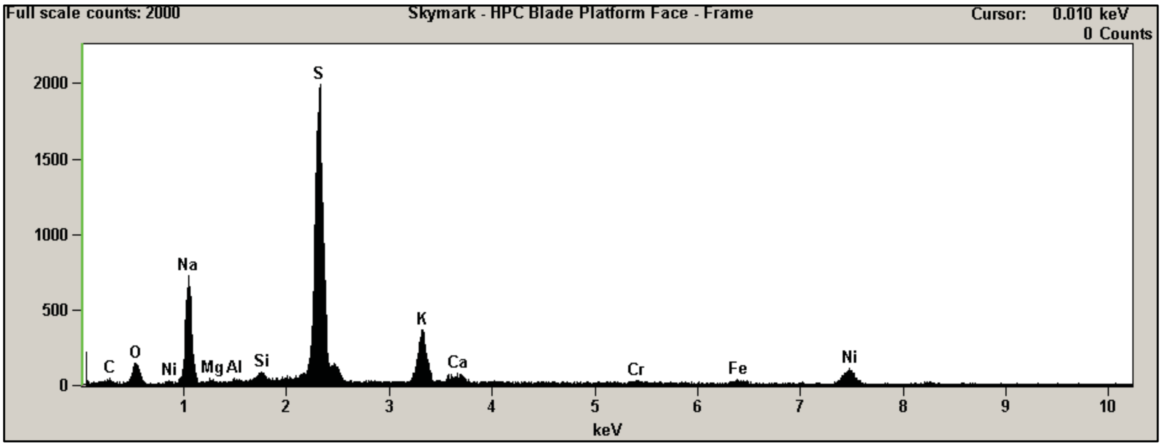



Photo 31　White Deposits on the CRF and Hub



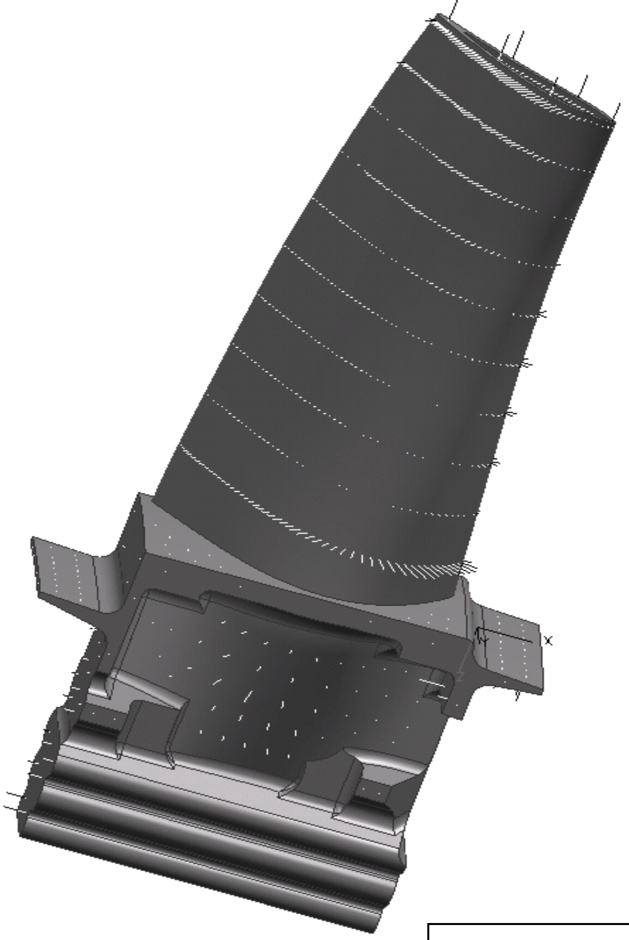
## Photo 32　Debris From Stage10 HPC Blades




White Debris Exhibited Elevated Sulfur(S), Sodium(Na) and Potassium(K) along with Oxygen(O)

Photo 33　Valisys for Evaluation of External Features




Provided by Engine Manufacturer