

NATIONAL TRANSPORTATION SAFETY BOARD

Office of Research and Engineering
Materials Laboratory Division
Washington, D.C. 20594



June 14, 2018

MATERIALS LABORATORY FACTUAL REPORT

Report No. 17-043

A. ACCIDENT INFORMATION

Place : Pensacola, Florida
Date : August 27, 2016
Vehicle : Boeing 737-700, N766SW
NTSB No. : DCA16FA217
Investigator : Pierre Scarfo, AS-40

B. COMPONENTS EXAMINED

Fan blades, fan blade shims, fan blade spacers, and fan blade platforms from the accident engine, 2 exemplar fan blades removed from service from other engines, and 10 blades returned from field inspections.

C. DETAILS OF THE EXAMINATION

On August 27, 2016, about 9:22 central daylight time, a Boeing 737-7H4, registration number N766SW, powered by two CFM International CFM56-7B24 turbofan engines, operating as Southwest Airlines flight number 3472, experienced a left (number 1) engine fan blade separation, loss of inlet cowling, and cabin depressurization while climbing through flight level 310. The flight crew declared an emergency and diverted to Pensacola International Airport (PNS), Pensacola, Florida. The regularly scheduled passenger flight was operating under 14 *Code of Federal Regulations* Part 121 from Louis Armstrong New Orleans International Airport (MSY), New Orleans, Louisiana, to Orlando International Airport (MCO), Orlando, Florida. None of the 99 passengers, 1 lap child, and 5 crewmembers onboard reported injuries.

An overall view of the number 1 engine after the emergency landing is shown in figure 1. The engine inlet was fractured, and the forward portion of the inlet was missing. One of the 24 fan blades was fractured at the root, and the remaining blades showed impact damage. During engine operation, the fan blades rotate clockwise as viewed looking forward. The fan blades were numbered sequentially also in the clockwise direction in accordance with blade numbering instructions provided in the Boeing Aircraft Maintenance Manual¹ where the number 1 blade is located immediately above an offset hole on the fan disk. The fractured blade was blade 23 as indicated in figure 1.

¹ Aircraft Maintenance Manual, Boeing 737-600/700/800/900, The Boeing Company.

The engine was a CFM56-7B engine manufactured by CFM International, a joint venture between GE Aviation and Safran Aircraft Engines. A schematic drawing from the Aircraft Maintenance Manual shown in figure 2 illustrates the assembly of fan blades, shims, spacers, and platforms in the engine fan assembly. To facilitate examination of the fan blades from the number 1 engine, the fan blades, fan blade shims, fan blade spacers, and fan blade platforms were shipped to the NTSB Materials Laboratory for examination.

1. Fan Blade Construction

The fan blades are manufactured of a titanium alloy with 6 percent aluminum and 4 percent vanadium (commonly known as Ti-6-4). All titanium alloy surfaces of the blades are shot peened. The contact faces of the blade dovetail² are grit blasted and then coated with a copper alloy containing nickel and indium (Cu-Ni-In) applied by plasma spray. A baked dry-film lubricant of molybdenum disulfide is then applied to the exterior of the Cu-Ni-In coating. Another coating of molybdenum disulfide lubricant is sprayed on the surface before the blade is installed in the shim.

The shims are manufactured of alloy 718, a nickel-chromium-iron alloy. The contact faces on the exterior of the shims are also coated with a baked dry-film lubricant with additional lubrication applied by spray and by brush before installation in the fan disk slot during first assembly of the engine.

The fan blade spacers are manufactured of Ti-6-4 with an elastomer coating. The fan blade platforms between the blades are manufactured of an aluminum alloy and have elastomer seals on the sides in contact with the blades.

2. Service History

At the time of the accident, the number 1 engine had accumulated 60,790 hours time since new (TSN) and 36,329 flight cycles since new (CSN). Since the last shop visit in June 2007, the engine had accumulated 30,237 hours and 17,956 flight cycles. All 24 fan blades from the number 1 engine experienced the same service history operating as a complete set with no blades within the set requiring replacement from the time of first installation on another engine in March 1998. In June 2001, the blades were reworked in accordance with CFM International CFM56-7B Service Bulletin 72-0253, which included machining of the dovetail profiles to prevent spalling and wear damage of the Cu-Ni-In coating at the forward and aft ends of the contact faces. Because of the rework, the part numbers of the blades changed from 340-001-022 to 340-001-027. At the time of the rework, the blades had 11,720 hours TSN and 7,754 CSN. The reworked blades were installed on the accident engine in July 2001. The total service time for the blades at the time of the accident was 62,851 hours TSN and 38,152 CSN.

When the accident fan blades were first installed in the other engine in 1998 and in the accident engine in 2001, the fan assembly design did not include fan blade shims.

² The dovetail is the root end of the blade where it is retained within a slot in the fan disk.

A new disk with a design requiring installation of fan blade shims between the blades and the disk was installed in the accident engine during the shop visit in June 2007.

The fan blades were last overhauled during the shop visit in June 2007. The blade overhaul process includes stripping the dovetail coating from the blades, fluorescent penetrant inspection (FPI), blend and repair of the airfoil leading edge as needed, shot peening of the entire blade, grit blasting the area of the dovetail to be coated, and coating the dovetail contact faces with the Cu-Ni-In coating followed by a cured dry-film lubricant.

After the last shop visit, engine records showed the fan blades received periodic lubrication. According to Aircraft Maintenance Manual procedures, the blades are removed from the disk and the components to be lubricated are cleaned. The blades are inspected for damage including nicks, dents, scratches, cracks, tears, and tip curl; and the dovetail is inspected for areas of missing coating from the contact face. Shims are also inspected for cracks, tears, punctures, scratches, nicks, dents, and distortion. Blades and shims with damage to the dry film lubricant are permitted to continue in service provided the fan blade set is lubricated before operating under revenue service. As part of the blade installation process, the fan blade spacers and shims, and the dovetail slots in the fan disk are all lubricated with Molykote D321 or Dow Corning 321 lubricant applied by spray or with Molykote G-n Plus or other lubricant meeting military standard MIL-L-8937 applied by brush. The thickness of the applied lubricant is specified to be 0.0003 inch to 0.0005 inch for the spray application or 0.004 inch to 0.008 inch for the brush application. The Aircraft Maintenance Manual states the blades and shims should be lubricated at intervals not to exceed 5,000 hours or 3,000 cycles, whichever occurs first.

On the accident engine, the fan blades and shims were lubricated 14 times since the June 2007 shop visit including the first installation after overhaul. The maintenance instructions provide the option to use spray or brush the lubricant onto the surface, and the operator chose to brush the lubricant on. The number of flight cycles between the last 6 fan blade lubrication applications (earliest to latest) were 1,199 cycles, 1,295 cycles, 1,334 cycles, 1,219 cycles, and 1,293 cycles. At the time of the accident, the fan blades had 1,396 cycles since the last lubrication. Early in the blade life, relubrication of the fan blades was not a required maintenance item. The first relubrication for the accident blades occurred in May 2001 after the blades had accumulated 11,242 hours TSN and 7,452 CSN.

During the lubrication that occurred 2,689 cycles prior to the accident, 2 fan blade shims were replaced. At the last lubrication prior to the accident, 4 fan blade shims were replaced. The blade numbers associated with the replaced shims and the cause for replacement were not noted in the maintenance record in either case.

3. Examination Process

Participants in the laboratory examination led by the NTSB included representatives from Southwest Airlines (airplane operator), GE Aviation (engine manufacturer), Safran Aircraft Engines (engine manufacturer), The Boeing Company

(airplane manufacturer), the Federal Aviation Administration (regulator), and the Bureau d'Enquêtes et d'Analyses pour la sécurité de l'aviation civile (French organization for safety investigations). Initially, the laboratory examination began at the NTSB Materials Laboratory in Washington, DC. In addition to work completed at the NTSB Materials Laboratory, work during subsequent phases of the examination also took place at the GE Engineering Materials Systems in Evendale, Ohio,³ and at the Safran Aircraft Engines Failure Analysis Laboratory in Moissy-Cramayel, France.

The components submitted for examination were the fan blades, fan blade spacers, fan blade shims, and fan blade platforms from the number 1 engine. Two exemplar blades referenced as blade A and B in this report were also obtained for examination during the course of the investigation. Exemplar blade A was the same part number and was from the same forging lot as fractured blade 23. Exemplar blade B was part number 340-001-026, indicating that it had been manufactured after the dovetail had been redesigned to prevent spalling and wear damage of the Cu-Ni-In coating at the forward and aft ends of the contact faces. Exemplar blades A and B had 30,676 CSN and 34,566 CSN, respectively. Eight additional fan blades with findings from field inspections were received later in the examination process, and results of the examination of those blades are described in the last section of this report.

The components were initially photographed and examined visually and under optical magnification. An initial examination of the fracture surface on blade 23 was also completed. After the initial examinations, the blade 23 piece was cleaned using an ultrasonic cleaner while submerged in a bath of acetone, and the intact blades were cleaned using a soft-bristle brush combined with a soap and water solution followed by an acetone rinse. Next, a detailed fractographic examination of the fracture surface from blade 23 was conducted including an examination using a scanning electron microscope (SEM). Metallographic sections were taken on blades 23 and 24. Blades 9, 17, 19, and 21 were selected for 3D measurement using a Faro Laser ScanArm fitted to a Faro Quantum FaroArm⁴ coordinate measurement device followed by local stripping of the Cu-Ni-In coating to facilitate a non-destructive inspection for cracks in the blade dovetail.

In the next stage of the examination, all blades from the accident engine and exemplar blade A were shipped to GE Aviation for additional photography, 3D measurement with a Nikon LC15DX laser scanner fitted to a Wenzel LH87 coordinate measurement machine, and ultrasonic testing (UT) inspections. The Cu-Ni-In coating was then chemically stripped from 16 of the blades (including blades 9, 17, 19, and 21 that had already been stripped in a local area and the exemplar blade A) by soaking the dovetail in a bath of concentrated nitric acid, and the laser scans and UT inspections were repeated. Additionally, fluorescent penetrant inspection (FPI) and eddy current

³ Metallurgical Investigation Report Log No. FA2016-17800, *Metallurgical Evaluation of CFM56-7 Fan Blades from a Liberation Event on ESN 874-112 Operated by Southwest on August 27, 2016*, GE Aviation (January 23, 2017).

Metallurgical Investigation Report Log No. FAL2017-18067, *Metallurgical Evaluation of a CFM56-7 Fan Blade from ESN 892-439 Operated by Southwest Airlines*, GE Aviation (September 5, 2017).

⁴ FARO Technologies, Inc., Lake Mary, Florida.

inspections (ECI) were also conducted. Selected blades both with and without crack indications from the nondestructive inspections (NDI) were further examined with metallographic sections through the dovetail in the area of crack indications. As needed, the additional examination was facilitated by etching the surface with Kroll's reagent modified to a higher acid concentration and sectioning to view the surfaces using an SEM. Blades with crack indications were then sectioned to open the cracks for fractographic inspection. Fractographic examination of crack features occurred in laboratories at the NTSB, GE Aviation, and Safran Aircraft Engines.

Five event blades and exemplar blade B were sent to the Safran Aircraft Engines laboratory for examination including photography, coordinate measurement and UT inspections before and after stripping the coating, and FPI and ECI after stripping the coating. Surfaces of the blade dovetails were examined by SEM. Surface roughness was also measured in the grit-blast area under the Cu-Ni-In coating after stripping on one of the blades.

Residual stress was measured on selected blades from the accident engine and the two exemplar blades. Tests were completed at Lambda Technologies Group (Cincinnati, Ohio) and at Safran Aircraft Engines.

Metallographic samples from 2 blades were further examined using electron backscatter diffraction (EBSD) at the GE Global Research Center in Niskayuna, New York, to map grain orientations in the selected blades.

4. Optical Examination

Overall views of the submitted components from the accident engine as received are shown in figures 3 and 4. The fan blade spacers, fan blade shims, and airfoil platforms are each shown below their respective blade as numbered at the bottom of each image. The shim for blade 23 remained installed on the dovetail, and the shims from the rest of the blades had been removed from their respective dovetails as shown in figures 3 and 4. The airfoil platforms are positioned between blades, and the platform number corresponds to the blade facing the convex side of the platform. Thus as numbered, the blade 23 platform was located between blade 23 and blade 24 and preceded blade 23 as the fan rotated. As indicated in figure 3, blade 24 preceded blade 23 in fan rotation, and blade 22 trailed blade 23.

a. Fan Blades

Blade 23 was fractured at the root, and most of the remainder of the blade was missing. The airfoils for the other 23 blades showed deformation, tearing, and fractures consistent with hard-body impacts such as from blade fragments. No evidence of feathers or blood was observed on the blades.

An overall view of the fracture surface for blade 23 is shown in figure 5. A portion of the fracture surface was relatively smoother with curving crack arrest lines, features

consistent with fatigue. Fracture features emanated from an origin area centered approximately 2.1 inches from the leading-edge face on the convex side of the blade as indicated with a bracket in figure 5. Ratchet marks⁵ were observed in the origin area, consistent with multiple fatigue origins.

To facilitate a further examination of the fracture surface, the blade 23 fan blade shim was disassembled from the blade 23 dovetail by sliding the shim forward off the blade piece. The shim was simultaneously pried open slightly to prevent sliding contact marks on the mating surfaces. An overall view of the blade 23 piece and shim after disassembly is shown in figure 6, and a closer view of the convex surface of the blade piece in the area of the fatigue origin is shown in figure 7. The fracture initiated at the contact face just inboard of the outboard edge⁶ of the Cu-Ni-In coating (see figure 7).

A closer view of the fatigue region on the blade 23 piece is shown in figure 8. A dashed line in figure 8 represents a boundary of the fatigue region where well-defined curving crack arrest marks were observed. This region was 1.143 inches long axially at the blade surface and extended up to 0.217 inch deep. Beyond that region, fracture features were in a relatively flat plane with a curving boundary indicated with unlabeled arrows in figure 8. The flat-plane region defined by unlabeled arrows extended up to 2.277 inches axially and nearly through the thickness (0.695 inch deep). Beyond the boundary defined by unlabeled arrows, the fracture surface had a lighter gray appearance with prominent shear lips, features consistent with ductile overstress fracture. Following the optical examination of the fracture surface, a detailed SEM examination of the surface was conducted as described later in this report.

Areas of the Cu-Ni-In coating on the blade 23 dovetail were disbonded and missing from the dovetail contact face in an area adjacent to the fracture and in a larger area at the inboard edge of the contact face directly inboard of the origin area as indicated in figure 7. The larger area of missing coating at the inboard edge of the contact face was 0.87 inch long and extended up to 35 percent of the width of the contact face. Gray lubricant material was observed on the blade surface in the inboard disbonded area where the coating was missing. Linear features corresponding to variations in lubricant thickness were also observed in the inboard disbonded area as indicated in figure 7. The lines of varying thickness were oriented slightly angled relative to the blade longitudinal axis.

The dovetails for the remaining blades were examined visually and using an optical stereomicroscope. No evidence of cracks in other blades were detected during the optical examination. However, many of the blades showed areas of disbonded and missing Cu-Ni-In coating in the contact faces of the dovetails. In addition to blade 23 shown in figure 7, Blades 2, 3, 11, 17, 18, 19, and 24 had areas of missing coating on the convex side of the blade near the forward end at the inboard and/or outboard edge of the contact

⁵ A ratchet mark is a small step in the fracture surface formed when two adjacent fatigue cracks originate on slightly offset planes.

⁶ Inboard and outboard reference positions relative to the fan disk's rotational axis where outboard is further from the axis in the radial direction.

face. Sizes and locations of the areas of missing coating are listed in table 1. Blade 17 had the largest area of missing coating with up to 85 percent of the contact face width missing (see figure 9).

Some blades had Cu-Ni-In coating missing near the aft end of the contact face on the concave side of the dovetail. Missing coating in this area was noted in blades 1, 2, 8, 9, 10, 13, 15, 17, 20, 22, and 24. An example of coating damage on the concave face near the aft end on blade 9 is shown in figure 10. The missing coating was also frequently associated with areas where lubricant and coating was flaked away from the surface inboard of the contact face. Missing coating material inboard of the contact face was observed on blades 3, 14, and 18. The area of missing coating from the concave side near the aft end was generally less than 10 percent of the width of the contact face.

Table 1. Blades with Missing Coating at the Convex Side

Blade Number	Length (inch)	Furthest Extent from Forward Face (inch)	Width Missing (percent of contact face width)	Edge with missing material
2	1.25	2.52	35	Inboard
3	0.75	2.40	10	Inboard
11	0.81	2.32	20	Inboard
17	1.56	2.56	85	Both
18	0.47	2.13	<5	Outboard
19	1.50	2.83	70	Both
23	0.87	2.56	35	Both
24	1.13	2.41	55	Inboard

According to the Aircraft Maintenance Manual in effect at the time of the last blade lubrication, missing coating at the forward and aft ends of the contact faces on the convex and concave side of the dovetail was permitted up to specified limits. Missing coating within defined areas labeled area X in the manual was permitted indefinitely with repeat inspections conducted at as-scheduled blade lubrication intervals. For blades missing coating within larger areas labeled area Y, the blade could continue in service with repeat inspections and lubrication occurring at no more than 1,500 cycles or 2,000 hours, whichever occurred first. For missing or chipped coatings that was "more than" area Y, the blade was permitted to continue in service for no more than 50 flight cycles or 75 flight hours, whichever occurred first, provided no contact wear was observed on the blade surface in the area of missing coating. The areas defined as areas X and Y are shown as green hatched areas in figure 11. All blades listed in table 1 had missing or chipped coating that after the accident were beyond the limits defined by area Y.

The Cu-Ni-In coating on the dovetails overall had a mix of relatively rougher areas that were light gray to light gold in color mixed with areas that were smoother and dark gray to black in color. Figure 12 shows a view of the dovetail surface as observed on blade 19. Deposits of gray lubricant were generally observed around the edges of the

contact faces. The coating on many of the blades showed evidence of heavy sliding contact with sliding marks and deformation oriented in the radial direction. The outboard edge of the contact area showed deviations outboard at the forward end of the convex side and the trailing end of the concave side. In blades 9 and 11, the outboard edge of the coating was displaced and torn consistent with heavy sliding contact at the forward convex side of the contact face. In some of the rubbed areas, exterior layers of the Cu-Ni-In coating appeared to be fractured away leaving a crater-like appearance. Examples of these areas of coating surface fractures on blade 9 are indicated in figure 10. Later examination of the shims revealed coating material adhered to the interior surfaces of the shim in areas where these crater-like features were observed as will be documented later in this report.

Part numbers, serial numbers, moment weights, and commercial and government entity (CAGE) codes were marked on the root ends of the blades. A view of the root end of blade 23 is shown in figure 13. A search of manufacturing records associated with the blade serial numbers revealed blades 7, 9, 21, and 24 were from the same forging lot as blade 23. Exemplar blade A was acquired from the field as a representative blade from another engine that was also from the same forging lot as blade 23.

b. Fan Blade Shims

Two fan blade shim designs were observed associated with the blades, and differences in the designs are illustrated in figure 14. In the original fan blade shim design, tangs at the forward end of the shim (shown in blue in figure 14) were separated by a gap. A new shim design was introduced in 2008 where the tangs were joined by a bridging piece shown in red in figure 14. Blades 1 through 21 had the older style shim where tangs at the leading edge of the shim were separated by a gap, and shims on blades 22, 23, and 24 had the tabs joined consistent with the newer design introduced in 2008.

Most of the fan blade shims were intact as shown in figures 3 and 4. The shim for blade 21 was cracked at the outboard edge of the contact face on the concave side of the shim and was fractured along most of the inboard edge of the contact face on the convex side. The shim for blade 22 was fractured along the inboard edge of the contact face on both sides of the shim. The pieces outboard of the fractures in shims 21 and 22 were missing. Several of the other shims were cracked or dented at the edges of the contact faces.

The exterior contact faces on the shims were generally different in appearance compared to the noncontact surfaces. The contact faces had variable colors with shades of gray, black, and gold areas observed. Outboard of the contact areas, radially oriented streaks of medium gray lubricant deposits were observed, and relatively uniform medium gray deposits of lubricant were present inboard of the contact areas.

The convex side of the fan blade shim for blade 24 is shown in figure 15 adjacent to the blade dovetail. A pattern matching the area of the disbonded and missing coating in blade 24 was visible on the exterior contact face of the shim.

Shims for blades 2, 3, 4, 7, 11, 17, 19, 23, and 24 were sectioned along the centerline to facilitate a visual examination of the interior contact faces. After an initial examination, the shims were cleaned in an acetone bath using an ultrasonic cleaner.

A view of the interior surface of the blade 23 fan blade shim in the area of the fatigue origin is shown in figure 16 (lower image) after cleaning. For comparison, the corresponding area on the blade 23 dovetail after cleaning with acetone is shown in the upper image in figure 16. Darker areas on the interior surface had a pattern that was essentially a mirror image of the area of the Cu-Ni-In coating on the dovetail. In the shim areas mating to the missing Cu-Ni-In coating, the surface finish was comparable to the areas of the shim outside the contact areas. No wear or contact patterns were observed to indicate prior contact with a Cu-Ni-In coating in the area of missing coating.

The interior surface of the blade 24 shim in the as-sectioned condition before cleaning is shown in figure 17 (lower image). A contact pattern forming a mirror image of the Cu-Ni-In coating on blade 24 in the coating disbond area on the convex side of the dovetail (figure 17, upper image) was observed. Gray deposits of lubricant were observed on the surface of the shim in the area mating to the missing coating area on the blade 24 dovetail. Deposits of gray lubricant were also observed on the shim along the inboard edge of the contact area, and radially oriented trails of gray lubricant were observed outboard of the contact area.

The interior surface of the blade 19 fan blade shim on the convex side of the blade is shown in figure 18 (lower image) after cleaning. A mirror image of the coating contact area was observed on the interior surface matching the area where the Cu-Ni-In coating was disbanded and missing on the convex side of the blade 19 dovetail as shown in the upper image in figure 18. The surface of the shim mating to the area of missing coating on blade 19 appeared shiny and worn consistent with prior contact with a coating.

Among the sectioned shims, shims 2, 3, and 11, had areas where pieces of coating fractured from the blade dovetail and adhered to the interior surface of the shim. Figure 19 shows areas near the leading edge of the blade 2 shim on the convex side (figure 19, lower image) where coating material deposited on the shim interior leaving corresponding crater-like fractures in the Cu-Ni-In coating on the blade 2 dovetail (figure 19, upper image).

c. Fan Blade Platforms

Many of the fan blade platforms showed cracks, deformation, and fractures at the sides contacting the blade airfoils. The blade 22 fan blade platform (the platform trailing blade 23) was fractured and mostly missing. Fracture surfaces of the fan blade platforms were matte gray and rough consistent with overstress fracture.

d. Fan Blade Spacers

The fan blade spacers were present for each blade and were intact. The elastomer strips for each spacer were intact with no damage. Areas mating to the blade shims also

had elastomer on the surface. Some fretting contact damage was observed on the forward face of the spacers. However, no cracks, dents, nicks, or wear, beyond limits listed in the Aircraft Maintenance Manual (Task 72-21-02-200-801-F00) were noted.

5. Fractography: Blade 23

After the initial optical examination described above, the fracture surface was inserted into the scanning electron microscope (SEM) in the as-received condition for an initial examination before cleaning. During the initial SEM examination, the fracture surface and coating were analyzed using energy dispersive X-ray spectroscopy (EDS). The EDS spectra obtained on the fracture surface had elemental peaks consistent with Ti-6-4 plus peaks of carbon and oxygen, and the coating had spectra consistent with Cu-Ni-In plus peaks of molybdenum and/or sulfur, aluminum, and oxygen.

After the initial SEM examination in the as-received condition, the blade 23 piece was cleaned in a bath of acetone using an ultrasonic cleaner. A close view of the origin area after cleaning as viewed with an optical microscope is shown in figure 20. Four distinct crack arrest lines were noted on the fracture as numbered in figure 20. The depths where the crack arrest lines intersected the right dashed line in figure 20 were 0.072 inch, 0.093 inch, 0.139 inch, and 0.228 inch. A possible hint of another crack arrest line between lines 3 and 4 was present particularly near the aft side of the fatigue region, but the boundary was not distinct around the entire crack front as in the labeled arrest lines.

A detailed SEM examination of the fracture surface was conducted on the cleaned fracture surface. An overall view of the origin area as viewed using an SEM is shown in figure 21. The crack arrest lines that were visible by optical microscopy were also visible in the SEM image as indicated in figure 21 (the fourth line was also visible but was outside the field of view in the image shown in figure 21).

At higher magnifications, striations consistent with low-cycle fatigue crack growth were observed. Typical views of fracture features within the fatigue region at various distances from the convex surface of the blade are shown in figures 22 through 26. In general, striation spacing became larger as the depth of the crack increased. However, at the crack arrest lines shown in figure 20, the striation spacing showed an abrupt change where the distance between striations immediately past each arrest line was noticeably shorter than the spacing leading up to the arrest line, consistent with a temporary reduction in crack growth rate as the crack grew after each arrest line had been formed.

Isolated areas of quasi-cleavage fracture were interspersed within the areas of striations at crack depths greater than approximately 0.100 inch. At greater crack depths, the fracture features gradually transitioned to more quasicleavage fracture with isolated areas of striations. Figure 26 shows an area beyond crack arrest line 4 where a mix of striations and quasicleavage fracture features were observed. No discernable striations were observed at distances beyond 0.321 inch from the convex surface. Ductile dimples were observed in the areas of slant fracture beyond the flat fracture region bounded by arrows in figure 8.

Surveys of the fracture surface were conducted along the paths indicated with dashed lines in figure 20 to estimate the number of striations in the region of stable fatigue crack growth. The first survey was taken along the dashed at the left in figure 20. Images from the survey were then independently examined by the NTSB and GE. Similar results were achieved, verifying that there was general agreement with the process and the features that were being counted as striations. Another survey was conducted along the dashed line at the right in figure 20, and the striations were counted by the NTSB and added to the previous data from the first survey.

Striation density as a function of distance from the origin for the blade 23 fracture is shown plotted in figure 27. A power-law curve was fitted to the data as shown by the dashed line in figure 27. By integrating the area under the curve, a total number of striations could be estimated. The locations of the crack arrest lines are also noted in figure 27 as numbered 1 through 4. For crack growth from 0.0005 inch to crack arrest line 4 at 0.228 inch, the estimated number of striations was approximately 14,000. Beyond, crack arrest line 4, an accurate estimate of the number of striations was not possible due to the extent of ductile tearing intermixed with the striations. However, based on the fractography of a crack in blade 24 as described below, it was estimated that crack growth beyond crack arrest line 4 on blade 23 represented approximately 1,000 engine flight cycles. Since one striation is usually associated with one engine flight cycle, the number of engine flight cycles associated with crack growth from 0.0005 inch deep to failure was estimated to be approximately 15,000 cycles.

Crack arrest lines generally represent a change in stress state, environment, or time interval associated with fatigue crack growth. The blade manufacturer has reported that lubrication of the fan blade roots effectively reduces the overall stresses on the blade root in the contact areas. The number of cycles between crack arrest lines on the blade 23 fracture surface were estimated by integrating the area under the curve in figure 27 between each of the crack arrest lines. The estimated number of cycles between arrest lines were 1,020 cycles, 1,709 cycles, and 2,438 cycles from arrest lines 1 to 2, 2 to 3, and 3 to 4, respectively. As noted in the engine history above, the intervals between the last 6 lubrication applications ranged from 1,199 cycles to 1,295 cycles, values comparable to the estimated cycles accumulated between crack arrest lines 1 and 2 and lines 2 and 3 and approximately half the estimated cycles accumulated between crack arrest lines 3 and 4.

6. Metallography: Blade 24

During the group examination at the NTSB laboratory, the group decided to delay a metallographic examination of blade 23 in the area of the fatigue origin to allow time for a complete fractographic examination. However, it was noted that blade 24 had similar damage to the Cu-Ni-In coating, and blade 24 was from the same forging lot as blade 23. Therefore, blade 24 was selected for metallographic examination. A transverse cut was made through the root of blade 24, and a second cut was made at an angle to intersect the end of the first cut, thereby removing a triangular piece shown in figure 28. The outboard end of the triangle was also trimmed to facilitate mounting the sample in a polymer mold for polishing in preparation for metallurgical examination on the face

indicated in figure 28. The polished specimen was then etched with 15/15 reagent (15 milliliters hydrofluoric acid and 15 milliliters nitric acid in 70 milliliters water).

An overall view of the polished and etched cross-section is shown in figure 29. During mounting, the inboard corner on the convex side was deformed in the mounting press, and as a result, the deformed corner was missing from the prepared plane shown in figure 29. The contact faces at the convex and concave sides are indicated with brackets. The Cu-Ni-In surface coating was present on the contact face at the concave side and on the upper half of the contact face on the convex side.

Cracks consisting of a primary crack and many secondary cracks were observed emanating from the surface of the blade near the outboard end of the contact face on the convex side of the blade (circled area in figure 29), which corresponded to the origin area for the blade 23 fracture. As described later in this report, the secondary cracks revealed in the cross-section were found to be part of a network of microcracks⁷ that was present in the vicinity of the primary crack.

Closer views of the cracks in blade 24 are shown in figures 30 through 33. The primary crack was oriented perpendicular to the contact face up to a depth of 0.086 inch. Then the crack turned nearly parallel to the blade root face for an additional 0.025 inch. The total crack depth from the surface to the crack tip was 0.110 inch in the polished cross-section shown in figure 32.

Closer views of the primary crack in blade 24 where it emanated from the surface under the Cu-Ni-In coating are shown in figures 30 and 31. Secondary cracks as indicated in figures 31 through 33 also emanated from the blade surface under the coating. Secondary cracks were noted both outboard and inboard of the primary crack. The coating was disbonded from the surface at the outboard end of the contact face, but the coating remained intact covering the cracks from view during the optical examination of the blade prior to sectioning.

The etched microstructure for the blade showed a nearly even mix of alpha grains (appearing white or light gray) interspersed with areas of alpha plus beta (brownish gray areas), consistent with the expected microstructure for the specified material after etching with Kroll's reagent. No surface anomalies, such as distorted grains or alpha case, were observed on the surfaces.

7. Metallography: Blade 23

A transverse cross-section was cut from the aft end of the dovetail for blade 23 to examine the microstructure of that blade without destroying the fatigue region on the fracture surface. The cross-section was polished then etched with 15/15 reagent to reveal the blade microstructure. Microstructural features were similar to that of blade 24 with a nearly even mix of alpha grains interspersed with areas of alpha plus beta, consistent with the expected microstructure for the specified material after etching with Kroll's

⁷ A microcrack is a crack feature of microscopic proportions usually visible at magnifications of at least 50 times.

reagent. No surface anomalies, such as distorted grains or alpha case, were observed on the surfaces.

8. Nondestructive Inspections

A series of NDI procedures including FPI and other procedures under development at the time (UT and ECI) were conducted on the intact blades from the accident engine and on exemplar blades A and B to inspect for cracks on the convex side near the leading edge. UT inspections from the root surface (base), FPI, and ECI inspections allowed for a complete inspection of the root on both the concave and convex sides, but no confirmed cracks were found outside of the location associated with the origin area for the blade 23 fracture.

A summary of the findings is listed in table 2, and additional details are provided below. Results of the optical examination of the coating described in a previous section of this report and confirmed crack findings as described in the next section are also included in table 2 for quick reference. Not all tests were completed on all blades due to time constraints, availability of inspection methods during the course of the examinations, and results from other steps in the examination process. Dashed lines in table 2 indicate inspections that were not completed during the examination process.

a. Ultrasonic Testing Before and After Stripping the Cu-Ni-In Coating

All the intact blades were inspected by UT before and/or after chemically stripping the Cu-Ni-In coating from the dovetail. Handheld UT inspections were conducted from the inboard root surface (base) of the blade along the length of the dovetail and from the airfoil surface aiming toward the dovetail convex pressure face approximately 2 inches from the leading edge (the fracture origin area on blade 23 and the crack location on blade 24). Several blades were also selected for a UT C-scan to further characterize and document crack indications in three dimensions.

An exemplar blade with an EDM notch at the fracture origin area was used as a reference for verifying the handheld UT inspection technique and for calibrating the signal response. The response from the reference notch measuring 0.100 inch long and 0.050 inch deep was used as the 100 percent reference signal. Percentages for UT inspections listed in table 2 indicate the percent signal returned to the UT transducer relative to the reference notch. Based on the signal response associated with the surface roughness and coating, a crack indication threshold was set at 20 percent of the reference signal.

The UT signal response during initial inspections with the Cu-Ni-In coating in place was generally noisier when compared to the signal response after the coating was removed. In blades 3, 7, 8, 10, 11, 13, 14, 15, 18, 22, 24, and exemplar blade A, UT signal responses initially exceeded the threshold value with the coating intact. However, the signal response during inspection after the coating had been removed was found to be below the threshold in blades 3, 7, 10, 13, 14, 15, 18, 22, and blade A.

A UT signal response exceeding the threshold after the coating was stripped was found in blades 8, 11, and 17. (Blade 24 is not included since it was not UT inspected after stripping, but the signal response from blade 24 as-coated was comparable to that of blade 11 before stripping.) Blades 11, 17, and 24 had cracks that were later confirmed by SEM examination. Blades 1, 2, and 19 had confirmed cracks that returned UT signals below the threshold but were later detected by FPI and/or ECI.

b. Fluorescent Penetrant Inspections

The crack in blade 24 had initiated under the intact Cu-Ni-In coating, and since detection by FPI requires the crack to be open to the surface, the coating on each of the blades was chemically stripped before FPI was performed. Some inspections were performed with the blade surface in the as-stripped condition, and some inspections were performed after the blade surface was etched with a modified Kroll's reagent. The coating on blades 16 and 20 was not stripped during the examination, so FPI was not completed on those blades. Also, no FPI was completed on blade 24 since the crack was examined without removing the coating. In most cases, FPI was not repeated on the etched surface of a blade that showed a negative finding by ECI.

Crack indications were detected by FPI in blades 1, 11, 17, and 19, all of which were later confirmed as cracks. A confirmed crack in blade 2 was not detected by FPI when inspected both before and after etching the surface.

c. Eddy Current Inspections

Nineteen of the 23 intact blades from the accident engine and the two exemplar blades were inspected using ECI. Since coating removal was required for the ECI inspection, blades 16, 20, and 24 were not inspected by ECI for the same reasons FPI was not completed on those blades. Blade 17 had a crack indication detected by FPI, and a destructive examination was completed before the ECI equipment was available for use during the examination. Therefore blade 17 also was not inspected by ECI.

As listed in table 2, all blades with confirmed cracks that were inspected by ECI had ECI crack indications. One blade (exemplar blade B) had an ECI indication that was later found to be associated with a scratch on the surface of the contact face near the forward end of the convex side of the blade. A destructive SEM examination was completed on 7 blades that had a UT signal response below the threshold after stripping (or before stripping when the UT inspection was not repeated after stripping) and also had a negative finding with ECI. Although microcracks were detected in each of the accident blades with negative indications as described below, no confirmed cracks were detected in the examined blades with negative ECI indications.

Table 2. Summary of Inspection Findings

Blade	Coating Spall		UT					FPI		ECI	SEM	Crack Depth (inch)	Crack Length (inch)	Est. Cycles (from 0.0005 inch)	
			Pre-Strip		Post-Strip		C-scan	As-Stripped	Etched						
	CC	CV	Base	Airfoil	Base	Airfoil									
1	Y		N	N	N	-	-	Y	-	Y	Y	0.009	0.057	~4,000	
2	Y	Y	N	25%	N	N	poor	N	N	Y	Y	0.015	0.046	~4,000	
3		Small	N	Y	N	N	-	N	N	N	M				
4			N	N	N	N	-	N	-	N	-				
5			N	N	-	-	-	N	-	N	M				
6			N	N	-	-	-	N	-	N	M				
7			N	Y	N	N	-	N	-	N	-				
8	Y		N	Y	N	Y	-	N	-	N	-				
9	Y		-	-	N	N	-	-	N	N	-				
10	Y		Y	Y	N	N	-	N	N	N	M				
11		Y	80%	>100%	92%	36%	Y	Y	-	Y	Y	~0.060	~0.64	~10,000	
12			N	N	-	-	-	N	-	N	M				
13	Y		Y	N	N	N	-	N	-	N	-				
14			Y	N	N	N	-	N	N	N	-				
15	Y		Y	N	N	N	-	N	-	N	-				
16			N	N	-	-	-	-	-	-	-				
17	Y	Y	-	-	N	25%	Y	-	Y	-	Y	0.023	>0.3	~5,000	
18		Small	Y	Y	N	N	N	N	-	N	-				
19		Y	-	-	N	N	-	-	Y	Y	Y	0.019	0.19	~4,000	
20	Y		N	N	-	-	-	-	-	-	-				
21			-	-	N	N	N	-	N	N	M				
22	Y		N	Y	N	N	-	N	-	N	-				
23		Y	Fracture									-	0.217	1.14	~15,000
24	Y	Y	>100%	>100%	-	-	Y	-	-	-	-	0.126	~0.447	~16,000	
A	N	N	Y	N	N	N	-	-	-	N	N				
B	N	N	N	N	N	N	-	N	-	Y	N				

Notes: Blades identified in red were from the forging lot that included blade 23. Blades identified by letters were exemplar blades from other engines.

“N” indicates no indication above a threshold value (20% of the response from a reference flaw in the case of UT from the base or airfoil).

“Y” or a numerical value represent detected coating spalls, crack features, or crack indications above a threshold value. Numerical values as a percent of the reference signal response are provided for indications of cracks confirmed by SEM examination. For blade B, the ECI indication was a false positive associated with a scratch at the forward end of the contact face.

“M” in the SEM column indicates microcracks were detected during the blade surface examination.

9. Examination of Blades with NDI Crack Indications

Blades 1, 2, 11, 17, 19, 24 and exemplar blade B were selected for additional examination. Blade 24 had a crack that was confirmed by metallographic inspection as described in a previous section, and the remaining blades were selected based on crack indications detected during FPI and/or ECI inspections. Blade 24 was examined by mechanically removing the spalled coating from the surface adjacent to the crack, and the remaining blades were examined with the coating chemically stripped. Most of the stripped blades were etched with modified 15/15 reagent before the examination, but blades 11 and 24 were examined without etching. Surfaces of the blades in the area of the indications were examined by SEM. In most cases, the SEM examination of the surface preceded selection of areas for metallography and for lab fractures to open the cracks, but for blade 24 the SEM examination of the surface followed the metallography and lab fractures.

a. Surface Examination

Sections of the dovetail were cut from blades 1, 2, 11, 17, 19, and exemplar blade B to facilitate SEM examination of the blade surfaces in the areas where crack indications were found. The inboard half of the lab fracture of blade 24 (as described below) was also examined. In each of the accident blades examined by SEM, a network of microcracks was observed on the surface such as shown in figure 34. At the location of the crack indications, some of the microcracks appeared to link together to form a longer crack feature. The cracks and microcracks were all located in the grit blast region where the coating had been stripped from the surface. The network of microcracks was generally 0.047 inch to 0.065 inch wide near the outboard edge of the grit blast region. On blade 24, the network of microcracks was observed mainly near the leading edge of the crack. The length of the network was not measured in every case. However, on blades 1 and 17 the network was approximately 0.5 to 0.6 inches long, and on blade 19 the network was 1.6 inches long. The networks were centered approximately 1.7 inches to 2.0 inches aft of the dovetail forward face.

No cracks or microcracks were detected in the shot-peened surface outboard of the grit blast zone in any of the blades. Also, no cracks or microcracks were observed in exemplar blade B, but longitudinal scratches were observed near the leading edge in the area where the ECI indication was detected, consistent with a false positive reading. Some of the blades were also examined by SEM on the convex side and near the trailing edge. No cracks or networks of microcracks were observed in any of the areas away from the convex side near the leading edge.

b. Metallography

Transverse cross-sections were cut into the dovetails for blades 2, 11, 17, 19, and exemplar blade B for metallographic examination (blade 24 had already been sectioned as described previously). The cuts were generally made away from the center of the crack indication to facilitate the fractographic examination to follow, but a second mount for blade 11 was made through the middle of the crack after the fractographic examination

was completed. In the case of exemplar blade B, the cut was made in the area where cracks were observed in other blades approximately 2 inches from the forward face. One of the microcracks in blade 17 outboard of the primary crack was 0.0036 inch deep. In blades 2, 11, and 19, microcracks were observed along the surface with depths up to 0.0047 inch, 0.0013 inch, and 0.0009 inch, respectively. No cracks were observed in the cross-section from exemplar blade B. Etched microstructures showed features similar to those observed on the blade 24 mount with no surface anomalies.

c. Fractography

A fractographic examination was completed on cracks in blades 1, 2, 11, 17, 19, and 24. First, transverse cuts were made in the dovetails of the cracked blades to isolate the cracked area to be opened. Next, longitudinal cuts were made to remove a triangular section from the middle of the dovetail. Finally, the inboard end of the dovetail was squeezed to open the cracks and produce a lab-induced fracture through the remaining portion of the dovetail. In the case of blade 24, the cracked portion that had been previously mounted for metallography was removed from the mount and processed to produce a lab fracture, but the opened crack did not contain the area of maximum crack depth. The adjacent portion of the dovetail was subsequently sectioned and subjected to a lab fracture, which revealed the deepest portion of the crack. Fractography of the cracks in blades 2 and 11 was completed at GE, and fractography of the cracks in blades 1 and 19 was completed at Safran. Fractography of the crack in blade 17 was completed at both GE and Safran, and fractography of the crack in blade 24 was completed at the NTSB, GE, and Safran.

The lengths and the depths of the cracks are listed in table 2. In the case of blade 11, a cut used to open the crack intersected the crack boundary at the location of deepest penetration, resulting in an estimate for crack depth in that case. In the case of blade 17, the entire length of the crack was not revealed, and for blade 24, the width of the saw cut was estimated to determine the approximate crack length. In blade 11, the primary crack length is shown, but additional microcracks were present adjacent to the aft end of the primary crack.

Optical images of opened cracks are presented in figures 35 through 37. Dashed lines in figures 35 through 37 indicate the fatigue boundaries associated with the primary cracks. The crack surfaces were relatively flat with curving crack arrest lines and boundaries, features consistent with fatigue. All cracks emanated from multiple origins located at the blade surface in the grit blast area of the dovetail approximately 2 inches aft of the forward face. In blades 11, 17, and 19 as shown in figure 36, the crack front had multiple curves frequently separated by ratchet marks consistent with multiple cracks coalescing into a uniform crack front. The smallest opened crack was in blade 2 (see figure 35). The blade 2 crack had a relatively smooth crack front, but slightly smaller cracks were observed on offset planes near the primary crack. A crack arrest line was visible in the blade 2 fatigue region as indicated in figure 35. In blade 1, additional fatigue regions were noted forward and aft of the main crack along a length of 0.19 inch along the blade surface. The largest crack was found in blade 24 as shown in figure 37. Six

crack arrest lines as indicated with arrows in figure 37 were identified within the fatigue region of the blade 24 crack.

SEM examinations were completed on each of the cracks opened by lab fractures in blades 1, 2, 11, 17, 19, and 24. Each of the cracks showed striations consistent with low-cycle fatigue cracks in Ti-6-4. Relatively fine striations were observed near the origin that generally decreased in density with increasing crack depth. A detailed examination of striation density on blade 24 showed crack arrest lines were associated with a change in striation density where striations just past the arrest line were finer than the striations that preceded the boundary, although the general trend was toward a decreasing striation density. On the crack surfaces in blades 17 and 19, fine striations were observed on crystallographic planes near the fatigue boundary consistent with high-cycle fatigue for a short time at the end of crack growth. Similar crystallographic features were observed within approximately 0.002 inch of the fatigue boundary for the blade 2 crack. Microcracks forward of the primary crack in blade 24 also showed fine striations on crystallographic planes within approximately 0.007 inch of the surface. However, a review of reference fractures for low-cycle fatigue in Ti-6-4 showed that fine striations can form on crystallographic planes near the origin under low-cycle fatigue load conditions.

Striation density as a function of distance from the origin was determined independently by the NTSB, GE, and Safran laboratories completing the fractographic examination. In cases where fractures were examined by multiple laboratories, reasonable correlation between data was achieved. The resulting data is shown plotted in figure 38. The results from blade 23 shown in figure 27 are also included in figure 38 for reference. Power-law curves were fitted to data from each of the blades and integrated from 0.0005 inch to the respective maximum crack depth to estimate the number of cycles associated with each crack. Results of the cycle estimates for each crack are listed in table 2 above. The cycle estimates ranged from 4,000 cycles for cracks in blades 1, 2 and 19 to 16,000 cycles for the crack in blade 24. Although the crack in blade 24 had more estimated cycles than the fracture in blade 23, the aspect ratio (length divided by depth) for the blade 24 crack was less than that of the crack that grew to failure in blade 23.

The number of cycles between crack arrest lines on the blade 2 and blade 24 fracture surfaces were estimated by integrating the area under the respective curve for each blade in figure 38 between each of the crack arrest lines (blade 24) and between the last (or only) arrest line and the crack boundary (both blades). Results are listed in table 3, where results from blade 23 are also listed for reference. The estimated values for accumulated flight cycles between crack arrest lines in blades 2 and 24 were comparable to the accumulated flight cycles between lubrication applications also listed in table 3 for reference.

Table 3. Cycles Accumulated between Crack Arrest Lines and Lubrication Applications

Interval	Blade 2 (Est. Cycles between Arrest Lines)	Blade 23 (Est. Cycles between Arrest Lines)	Blade 24 (Est. Cycles between Arrest Lines)	Lubrication Interval (Flight Cycles)
Last	1,100	1,200*	1,228	1,396**
1 st before last		2,438	1,770	1,293
2 nd before last		1,709	1,880	1,219
3 rd before last		1,020	1,291	1,334
4 th before last			1,485	1,295
5 th before last			1,798	1,199

*Estimate based on results of the last interval for blades 2 and 24.

**Last interval is from the last lubrication application to the accident flight.

10. Examination of Blades with Negative FPI and ECI Indications

Blades 3, 5, 6, 10, 12, 21, and exemplar blade A were selected for destructive examination using SEM and metallography. All seven blades had no crack indications resulting from FPI or ECI. Blades 3, 10, and exemplar blade A initially had UT signal responses above the threshold level before the Cu-Ni-In coating was stripped, but then had signal responses all below the threshold after the coating had been stripped. Blades 5, 6, 12, and 21 had UT responses below the threshold in all the inspections.

The dovetails in each of the blades were sectioned to facilitate an SEM examination of the convex side of the dovetail including the area approximately 2 inches away from the forward face. The surfaces were examined in the etched condition to facilitate imaging of crack features. Microcracks were detected in each of these blades examined by SEM. The microcracks were located in the grit blast region where the coating had been stripped from the surface. In blades 5 and 10, the microcracks were relatively isolated and limited in extent with individual cracks measuring up to approximately 0.002 inch long (see figure 39). More microcracks were observed in blade 3 extending up to approximately 0.004 inch long. A network of microcracks was observed on blades 6, 12, and 21 in a band ranging up to 0.04 to 0.06 inch wide within the grit blast region of the dovetail centered approximately 2 inches from the forward face. The length of the network was not measured on blade 21. However, on blades 6 and 12 the network was approximately 1 inch to 1.4 inches long. No microcracks were detected in the shot-peened surface outboard of the grit blast zone in any of the blades. Also, no cracks or microcracks were observed in exemplar blade A.

Metallographic sections through the network of microcracks were taken in blades 3 and 21. Microcrack depths ranged up to 0.0005 inch and 0.001 inch deep in the plane of polish for blades 3 and 21, respectively. A metallographic section was also taken through blade A approximately 2 inches from the forward face, and no microcracks were detected in the cross-section.

The dovetail for blade 21 was sectioned longitudinally to facilitate a lab fracture through the microcracked area. Images of the resulting fracture surface are shown in figures 40 and 41. The lab fracture intersected the microcracks, which appeared as a series of flat features with curving boundaries consistent with fatigue emanating from multiple origins along the convex blade surface. Dashed lines in figures 40 and 41 indicate the fatigue boundaries up to 0.0015 inch deep, and arrows indicate multiple origins at the blade surface.

For comparison, a lab fracture was produced on the concave side of the blade 10 dovetail in an area with no evidence of microcracks. As shown in figure 42, the fracture intersected the grit-blasted surface of the blade at a slant angle, and the fracture surface adjacent to the blade surface had ductile dimples, features consistent with ductile overstress fracture.

11. Electron Backscatter Diffraction

Electron backscatter diffraction (EBSD) was used to examine the grain orientation of alpha grains in metallurgical cross-sections through the dovetail from blades 3 (no crack) and 24 (largest crack). The EBSD work occurred after the crack in blade 24 was opened by lab fracture, so the previously mounted specimen was remounted and repolished for analysis by EBSD. The EBSD work was completed at the Microscopy and Microanalysis Laboratory in the GE Global Research facility in Niskayuna, New York. Grain orientations were mapped on the concave and convex sides of blade 3 near the outboard edge of the grit blast area. For blade 24, grain orientations were mapped at the convex side both inboard and outboard of the crack. Results of the EBSD work were included in the GE Aviation Metallurgical Investigation Report Log No. FA2016-17800, and an excerpt from that report showing the EBSD results is presented in Appendix A.

Maps of grain orientations where colors represented the orientation of the grain showed a relatively random distribution of orientations in the areas mapped. No groups of grains with similar orientations (colonies of aligned alpha grains) were observed. Pole figures also showed a relatively even distribution of grain orientations across the areas mapped with the greatest preference for any one orientation observed on the convex side of blade 3.

Grain contrast maps showed grains that appeared lighter near the surface relative to the underlying grains. The lighter appearance of the grains was associated with compressive stress from shot peening. The depth of the lighter grains appeared to be greater in blade 3 compared to that of blade 24.

12. Residual Stress Measurements

Residual stress measurements obtained by X-ray diffraction at the surface and at varying depths up to approximately 0.008 inch (200 micrometers) were conducted at Lambda Technologies in Cincinnati, Ohio, and at the Safran Materials and Processes Laboratory in Evry Corbeil and Vernon, France. The residual stresses measured at Lambda Technologies were measured parallel to the radial direction while measurements

at Safran were measured parallel to the longitudinal axis. Residual stress measurements were completed on blades 3, 7, 8, 17, and 23 at Lambda, and on blades 5, 6, 19, and 24 at Safran. Residual stress measurements were also conducted on two exemplar blades removed from other engines, referenced as exemplar blades A and B in this report. Residual stress profiles were measured at Lambda for exemplar blades A and at Safran for exemplar blade B. Exemplar blades A and B had each accumulated more than 30,000 CSN, and exemplar blade A was from the same forging lot as accident blade 23. A complete service history was not available for the exemplar blades, but the cycles accumulated since the last overhaul was estimated based on the date of last overhaul and typical usage data. The estimated number of cycles accumulated since the last overhaul was 2,700 cycles to 2,800 cycles for blade A and 14,000 cycles to 15,000 cycles for blade B. The measurements were compared to reference residual stress depth profile data, supplied by Safran, for blades that were peened in accordance with specified parameters.

Residual stress measurements were taken on multiple areas of the dovetail surfaces including convex and concave sides of the pressure face, near the forward and aft ends of the contact face, at the middle and near the outboard edge of the contact face, and outboard of the grit blast area (shot-peened only). To avoid influence from the fracture in blade 23 or the cracks or microcracks in the other blades, residual stresses at the crack location were generally taken near the middle of the contact face, and measurements at the outboard edge of the contact face were taken either forward or aft of the microcrack area. The blades that were tested for residual stress and the locations where stress profiles were obtained are listed in table 4. The second column in table 4 is a reference code used in this report to reference the location on the dovetail where the residual stress profile was acquired. As described in more detail at the bottom of table 4, the first letter in the code indicates the radial position, the middle characters indicate the longitudinal location on the dovetail, and the last letter indicates the convex or concave side of the blade. The locations on the convex side of blade 19 are shown labeled for example in figure 43.

For blades 23 and 24, local chemical stripping of the coating was completed prior to testing for residual stress. The remaining blades had been previously stripped for NDI.

Abnormal residual stress profiles were observed in all blades tested from the accident engine when compared to the reference profile as shown in figures 44 through 46. Minimum residual stress values (most compressive) from the residual stress profile measurements are listed in table 4. In many cases, the minimum residual stress value was higher (less compressive) than the expected minimum value, which is expected to be no greater than -500 megapascals. In table 4, the residual stress profiles with a minimum residual stress value that was above -500 megapascals (abnormal profile) are highlighted in yellow, and the profiles with a minimum residual stress value below -500 megapascals (expected profile) are highlighted in green.

Table 4. Residual Stress Measurement Results

Blade	Profile Location Code*	Test Lab	Residual Stress at a 10-micrometer Depth (MPa)	Minimum Residual Stress Value (MPa)
3	O2.3X	Lambda	-215	-492
5	O2X	Safran	116 (tension)	70 (tension)
6	MFV	Safran	-200	-390
6	M2X	Safran	-235	-310
6	S2X	Safran	-700	-720
6	OAX	Safran	-320	-360
6	MFV	Safran	-309	-370
6	M2V	Safran	-273	-430
6	S2V	Safran	-707	-740
6	OAV	Safran	-386	-500
7	O2.3X	Lambda	2 (tension)	-108
8	O2.3X	Lambda	-75	-247
17	O2.3X	Lambda	-199	-222
17	OAX	Lambda	-234	-400
19	M1.2X	Safran	-380	-520
19	M2X	Safran	-158	-310
19	M2V	Safran	-220	-330
19	M2.5X	Safran	-320	-420
19	M3X	Safran	-370	-390
19	S3X	Safran	-647	-720
19	MAX	Safran	-470	-590
23	M1.8X	Lambda	-56	-244
24	M1.8X	Safran	-184	-370
A	O2.3X	Lambda	-385	-697
B	O2X	Safran	-500	-580
Reference		Safran	-596	-675

*First letter: Radial position where M=middle of contact face, O=outboard edge of contact face, and S=shot-peened only.

Middle character(s): Longitudinal position where A=near the aft end, F=near the forward end, and number=approximate distance from the forward face in inches.

Third letter: Side of the blade where V=concave and X=convex.

Figure 44 shows the residual stress profiles for the fractured blade 23 and cracked blades 17, 19, and 24 acquired near the area of cracking on the contact face on the convex side of each blade. The reference blade residual stress profile is also shown in figure 44 for comparison with the accident blade profiles. The residual stresses in the cracked blades were less compressive than in the reference blade up to a depth of approximately 0.003 inch to 0.004 inch from the surface.

Figure 45 shows the residual stress profiles for blades 3, 5, 6, 7, and 8 that had no cracks. The profiles were acquired on the contact face on the convex side of each blade near the location where cracks had been detected in the other blades. The same reference blade profile is shown in figure 45 for comparison. The residual stress measurements in blade 5 were within the area of microcracks, and measurements in blades 3 and 6 were near the aft end of the microcrack networks. No examination for microcracks was completed on blades 7 and 8. The residual stresses in the accident blades were less compressive than the reference blades at depths of up to approximately 0.003 inch to at least 0.005 inch. Blades 5 and 7 had profiles where residual tension was measured below the surface. The minimum residual stress value in blade 3 was just above -500 megapascals, but the overall profile had an abnormal shape with compressive residual stress values less than -400 megapascals present at depths of 0.008 inch below the surface.

Abnormal residual stress profiles were also measured in blades 6, 17, and 19 in areas of the contact face away from the cracks and microcracks as shown in figure 46 with the reference profile shown again for comparison. In blade 6, abnormal residual stress profiles were acquired in 3 of the 4 locations tested near the forward and aft ends of the contact faces. In blade 17, a residual stress profile was acquired on the contact face at one location away from the area cracking, and that profile had the abnormal shape shown in figure 46. On blade 19, an abnormal profile was obtained on the concave side of the blade opposite the area of cracking.

Residual stress profiles with a minimum residual stress value less than -500 megapascals (highlighted in green in table 4) are shown in figure 47. These profiles had expected shapes comparable to the reference profile also shown in figure 47. All profiles acquired in the shot-peened areas on blades 6 and 19 (profiles 6 S2X, 6 S2V, and 19 S3X) had expected shapes comparable to the reference profile in contrast to the abnormal profiles measured on the contact face just inboard of those locations (profiles 6 M2X, 6 M2V, and 19 M2X). Profiles comparable to the reference profile were also acquired in several locations near the forward and aft ends within the contact faces for blades 6 and 19 (profiles 6 OAV, 19 M1.2X, and 19 MAX). The residual stress profiles for the two exemplar blades as measured near the outboard edge of the contact face on the convex side of the blade approximately 2.3 inches to 2.5 inches from the forward face (profiles A O2.3X and B O2.5X) were also comparable to the reference profile as shown in figure 47.

13. Cu-Ni-In Coating

After completing the optical examination of the Cu-Ni-In coating on the blades described in a previous section of this report, a metallographic inspection of the coating on blade 24 was completed. Views of the coating on the concave and convex sides of the blade are shown in figure 48. Aside from spalling near the crack initiation and missing coating at the inboard end of the contact face on the convex side, the coating was consistent with a plasma sprayed coating with no indications of excessive porosity, unmelted particles, or entrained oxides.

Vickers hardness measurements were conducted on the coating from blade 24 using a 300-gram load as specified for coating quality checks during blade manufacturing and overhaul. On the convex face, hardness was measured near the middle of the contact face where coating remained intact. On the concave side, hardness was measured near the outboard edge, in the middle, and near the inboard edge of the contact face. Also on the concave side, the coating extended around the corner of the dovetail inboard of the contact face, as is allowed in the coating specifications. Hardness was also measured in this area inboard of the contact face where the coating had not been compressed in service.

Results of the hardness measurements are listed in table 5. The hardness measured on the concave side inboard of the contact face had an average value of 147 HV0.3, which was lower than hardness values measured in the contact faces. The hardness in the contact faces ranged from 175 HV0.3 to 198 HV0.3. In 2007 when the coating was last applied to the accident blades, the specified hardness for the coating was a minimum of 120 HV0.3. As of 2008, the specified hardness for the coating is specified to be within a range of 140 HV0.3 to 180 HV0.3.

Table 5. Cu-Ni-In Coating Hardness

Measurement	Hardness, Convex Side, Middle of the Contact Face (HV0.3)	Hardness, Concave Side, Outboard Edge of the Contact Face (HV0.3)	Hardness, Concave Side, Middle of the Contact Face (HV0.3)	Hardness, Concave Side, Inboard Edge of the Contact Face (HV0.3)	Hardness, Concave Side, Inboard Corner (HV0.3)
1	196	183	197	191	159
2	201	179	208	165	174
3	206	164	201	170	110
4	202	165	162	180	138
5	184	182	173	195	154
Average	198	175	188	180	147

Coating thickness was measured using multiple techniques. On blade 23, the exposed outboard end face of the coating at the edge of the spall at the fatigue origin area was imaged and measured directly by optical microscopy. The coating thickness measured approximately 0.007 inch thick.

On blade 24, the thickness of the coating was measured on both the concave and convex sides of the dovetail within the plane of polish. On the convex side, the coating was thickest near the outboard end of the contact face and tapered down in thickness toward the spalled area at the inboard end of the contact face. Adjacent to the crack on the convex side, the coating was approximately 0.009 inch thick. In the area near the

middle of the contact face on the convex side (shown in figure 48), the coating thickness was approximately 0.008 inch. On the concave side, the coating was approximately 0.005 inch thick.

Before stripping the coating from the blades to facilitate NDI, 16 of the blades from the accident engine and exemplar blade A were scanned using a laser coordinate measurement device. After stripping the coating, the blades were rescanned, and the two sets of scan data were then overlaid to determine the differential between the scans, which was associated with the thickness of the coating that had been stripped from the blade.

Results of the coating thickness measurements are summarized in tables 6 and 7 for the convex and concave sides of the blades, respectively. Tables 6 and 7 list the maximum thickness measured in the contact face, where that maximum thickness was located longitudinally, the difference between the maximum and minimum thickness in the contact face (excluding areas of missing coating), and the location where minimum values were located (excluding areas of missing coating). Due to the method associated with determining the coating thickness, the measurements listed in tables 6 and 7 may have an approximate variation from actual values of ± 0.001 inch. Some data was missing for blades 17, 19, and 24 due to metallurgical mount preparation or local stripping of the coating that occurred before both laser scans were completed. According to coating specifications, the thickness of the coating after application should be between 0.0047 inch and 0.0071 inch (0.12 millimeters and 0.18 millimeters). Measurements that exceeded the specified maximum thickness value are highlighted in yellow in tables 6 and 7.

Typical coating thickness maps obtained for cracked blades (blades 1 and 11), blades with microcracks (blades 6 and 12), and an exemplar blade (blade A) are presented in figures 49 through 53. In figures 49 through 52, the thickness scale is the same in each image. Areas of the coating that are displayed in shades of yellow were within the specification range. Areas appearing orange or red were above the specified range, and areas appearing green or blue were below the specified range. In figure 53, green areas of the coating were within the specified thickness range. Areas appearing brown, orange, and red were above the specified range, and areas appearing purple, blue, or gray were below the specified range.

Five of the blades had coating thickness values that exceeded the specification on the convex side of the blade, including exemplar blade A. Nearly all the blades had coating thickness values that exceeded specification on the concave side. Most of the blades from the accident engine had lower thickness values near the forward and/or aft ends of the contact face. Blades 5 and 22 from the accident engine and exemplar blade A had relatively uniform thickness along the length of the convex face. As noted during the examination of the blades by SEM as described earlier in this report, microcracks in blade 5 were isolated and limited in extent, and no microcracks were detected in exemplar blade A.

Table 6. Cu-Ni-In Coating Thickness Data, Convex Side

Blade	Maximum Thickness (inch)	Location of Maximum Thickness	Thickness Differential (inch)	Location of Minimum Thickness
1	0.008	Middle and aft relatively uniform	0.006	Forward end
2	0.007	2 inches from forward face	0.005	Forward and aft ends
3	0.005	Middle and aft relatively uniform	0.002	Forward end
5	0.007	Relatively uniform	0.005	2 inches from forward face
6	0.008	2 inches from forward face	0.005	Forward end
7	0.005	Middle and aft relatively uniform	0.002	Forward end
8	0.007	Middle and forward relatively uniform	0.004	Aft end
11	0.008	2 inches from forward face	0.005	Aft end
12	0.007	Middle and aft relatively uniform	0.003	Forward end
13	0.004	Middle and aft relatively uniform	0.004	Forward end
15	0.006	Middle and aft relatively uniform	0.004	Forward end
17	0.007	2 inches from forward face (missing data adjacent to maximum location)	0.005	Aft end
18	0.007	Relatively uniform 2 inches from forward face and aft	0.005	Forward end
19	0.006	Middle (missing data from forward end)	0.003	Aft end
22	0.006	2 inches from forward face	0.004	Relatively uniform
24	0.009	2 inches from forward face (from metallographic exam)	0.007	Aft end
A	0.008	Relatively uniform	0.001	Relatively uniform

Table 7. Cu-Ni-In Coating Thickness Data, Concave Side

Blade	Maximum Thickness (inch)	Location of Maximum Thickness	Thickness Differential (inch)	Location of Minimum Thickness
1	0.009	Near aft end	0.004	Forward end
2	0.008	Near aft end	0.005	Forward end
3	0.006	Aft end	0.005	Forward end
5	0.009	Middle	0.009	Forward and aft ends
6	0.008	Middle and aft relatively uniform	0.005	Forward end
7	0.009	Middle	0.006	Forward and aft ends
8	0.011	Aft end	0.007	Forward end
11	0.009	Middle	0.006	Forward and aft ends
12	0.008	Middle	0.005	Forward and aft ends
13	0.011	Aft end	0.007	Forward end
15	0.011	Aft end	0.006	Forward end
17	0.008	Aft end	0.005	Forward end
18	0.007	Aft end	0.005	Forward end
19	0.007	Middle	0.004	Forward and aft ends
22	0.009	Aft end	0.007	Forward end
24	0.008	Aft end	0.006	Forward end
A	0.009	Aft end	0.002	Middle and forward relatively uniform

14. Surface Roughness

Using a Mahr MarSurf LD 130 surface analyzer, Safran measured the surface roughness on blades 19, 24, and exemplar blade B. The measurements were conducted on the grit-blast contact face of the dovetail along the radial direction on the convex side. The measured Ra was 2.48 micrometers, 2.38 micrometers, and 3.12 micrometers for blades 19, 24 and exemplar blade B, respectively.

15. June 2007 Blade Overhaul

The blades from the accident engine were last overhauled in June 2007 at GKN Aerospace. The repairs performed included a general inspection, blend repair of the airfoil with restoration of the leading edge, reconditioning of the Cu-Ni-In coating, and shot peening of the blade (airfoil and root). The standard overhaul process at the time was initiated by receiving the blades, removing the dry film lubricant by plastic bead blast, and chemical stripping of the Cu-Ni-In coating. Next, the blades were inspected to determine which repairs were needed including any weld repairs of the leading edge or other damage. Blending operations were then completed followed by visual inspections. Blades were then cleaned and inspected by FPI. After passing inspection, the airfoils and roots were shot peened in an automated chamber, cleaned, and polished. Next, the

blades were masked around the areas of the dovetail to be coated with Cu-Ni-In, and the exposed surfaces were prepared for coating by abrasive blasting by hand. Next, the plasma spray Cu-Ni-In coating was applied by an automated process. After a cleaning step was completed, a dry-film lubricant was applied and baked on the surface of the Cu-Ni-In coating followed by a final inspection.

Quality control checks were included in the overhaul process including the shot peening, grit blast, and plasma spray steps. At the beginning and end of the shot peen process for each set of blades, Almen strips were peened on a simulated blade root and checked for correct bending displacement. For the grit blast and plasma spray process, two sets of 6 test coupons were inserted into the dovetail of a blade modified with slots to hold the test coupons; three distributed along the contact face on the concave side and three distributed on the convex side. The test coupons were grit blast in the modified blade along with the blades being overhauled. During the plasma spray process, the modified blade with a set of 6 test coupons was included with the first and last group of blades (up to 4 blades in a group for the fixture used in 2007). The test coupons (12 total) were then sent to an in-house laboratory for metallographic inspection of the coating cross-section and hardness check. The metallographic inspection included checks for coating thickness recording maximum, minimum, and predominant (visual average) thicknesses for each coupon. The coating was checked for excessive porosity, unmelted particles, and oxides or any cracks in the coating, and the interface between the coating and the blade was checked for excessive separations, voids, porosity, or contamination. Finally, hardness was checked to verify that the minimum hardness value was met or exceeded. If test coupons for the coating process did not pass inspection, the set of blades were resent through the overhaul process starting with chemically stripping the coating.

Records from the 2007 blade overhaul were reviewed. The overhaul records showed that test coupons from the coating process passed the metallographic inspection and hardness tests during the first run through the overhaul process. The 12 test coupons associated with the overhaul were retrieved from storage and reexamined. No anomalies were noted in the repeated visual inspection. At the time of the 2007 overhaul, GKN measured Vickers hardness in the test coupons using a 100-gram load. The hardness tests were repeated on the samples after they were retrieved from storage using a 300-gram load. The average hardness values ranged from 145 HV0.1 to 239 HV0.1 with a combined average of 192 HV0.1 in the 2007 tests. When the tests were repeated at the higher load on the same test coupons in 2017, the values ranged from 158 HV0.3 to 219 HV0.3 with a combined average of 189 HV0.3. All values conformed to the required minimum hardness of 120 HV0.3.

Recorded thickness values of minimum, maximum, and predominant thickness were reviewed for the 12 test coupons associated with the 2007 blade overhaul. The process of determining the values consisted of scanning the length of the coating on each sample with an optical microscope to find the minimum and maximum thicknesses. The lab technician examining the coupons also determined the predominant thickness, which was essentially a visual averaging of the coating across the majority of the surface if the

peaks filled the valleys. The predominant thickness was the value used to determine if the coating thickness met specification. The recorded minimum thicknesses ranged from 0.0039 inch to 0.0055 inch, and the recorded maximum thicknesses ranged from 0.0066 inch to 0.0078 inch. The recorded predominant thicknesses ranged from 0.0051 inch to 0.0066 inch.

The test coupons from the 2007 blade overhaul were obtained from GKN Aerospace, and a montage image of each specimen was created at the NTSB Materials Laboratory. Spot checks of thickness measured on the montage images were consistent with recorded values of minimum, maximum, and predominant thickness listed in the GKN Aerospace lab report from the 2007 blade overhaul.

16. Field Inspection Findings

In this section, findings from fan blade field inspections reported before April 20, 2018 will be documented. Additional field inspection findings reported on or after April 20, 2018 will be documented in Materials Laboratory Factual Report 18-049 associated with an accident (DCA18MA142) that occurred on April 20, 2018 involving a fan blade fracture on another Boeing 737-700 airplane powered by CFM International CFM56-7B engines. The April 20, 2018 accident flight was operated as Southwest Airlines flight number 1380 and was diverted to Philadelphia International Airport, Philadelphia, Pennsylvania after the engine failure.

Shortly after the accident on August 27, 2016 (the subject of this report), a blade inspection program was initiated to assess the condition of a sampling of the fan blades in the fleet. The focus was on older blades with at least 30,000 CSN, but other blades entering the shop for overhaul were also examined. In addition to the normal overhaul process, the Cu-Ni-In coating on the blades was examined visually as the blades were received to assess the condition of the coating. After the coating was stripped, blades were inspected using UT and/or ECI in addition to FPI. By the end of November 2016, over 700 blades had been inspected, and nearly half of those blades had at least 30,000 CSN.

Among the inspected blades, 75 (11 percent) of the blades were found to have some extent of coating loss. In 69 percent of the blades with coating loss, the loss occurred at the trailing edge, and in 27 percent, the loss occurred at the leading edge. None of the inspected blades had coating loss in the area where cracking was observed on the accident blades (convex side approximately 2 inches from the forward edge).

As a result of the increased NDI completed during the blade inspection program, one blade showed an FPI indication at the leading edge corner, but subsequent detailed inspection at the Safran lab revealed the indication was a scratch and not a crack. Among the remaining blades inspected, some blades had a UT signal response above the threshold before the coating was stripped, but no signal response above threshold levels were detected by UT or ECI after the coating had been stripped.

In November 2016 after a tool for routine and repeatable ECI of the blade root was developed and available to the overhaul shops, ECI inspections were added to the overhaul process. By the end of April 2017, approximately 3,000 blades had been inspected by ECI. Although 19 blades had been removed due to ECI indications above the threshold signal response, the responses in each case were associated with surface features determined to be something other than a crack.

In February 2017, 7 blades from a single engine were found to have areas of coating damage when examined at GKN Aerospace and were retained for further examination. The blades were owned by Southwest Airlines and had recently been acquired from Jet Airways. The blade positions on the fan disk were 2, 8, 9, 15, 17, 20, and 21, and for purposes of this report, the blades will be referenced as blades C through I, respectively. Blades D and H were originally manufactured as part number 340-001-022 blades and had the dovetail rework changing their part numbers to 340-001-027. The remaining blades were all part number 340-001-026 blades. Blades D and H each had 31,597 CSN, and blade C had 26,479 CSN. The remaining 4 blades each had 17,015 CSN. Blade D was last overhauled by Snecma Services in 2000. In September 2014, the blade was cleaned and inspected during an engine shop visit, but no repairs were required.

Blades C through I were sent to Safran for further examination. In most of the blades, missing coating was located near the forward end on the convex side and near the aft end on the concave side. Blade C had missing coating along most of the length on the concave side of the blade. Blade G was the only blade that did not have missing coating on the concave side. Blade D had the most coating missing on the convex side with areas of coating missing from the forward end to the middle of the blade length. The extent of missing coating exceeded the area Y limits listed in the AMM manual at the time of the accident in 5 of the 7 blades, including blade D. New limits for missing coating were published in the AMM in June 2017. Under the new limits, the extent of missing coating observed on the 7 blades would have led to a required blade overhaul within 50 engine cycles for all blades except blade F.

Direct coating thickness measurements were conducted around the areas of missing coating on each of the blades. Additionally, coating thickness was measured on blades C, E, and G through I by scanning using a laser coordinate measurement device and then scanning again after the coating was stripped. Results from the thickness measurement showed good correlation between the direct measurements and those obtained from the laser scanning. Coating thickness measurements on the scanned blades showed average coating thickness ranging from 0.001 inch to 0.005 inch. Direct measurements of the coating thickness on blades D and F at the edges of the missing coating in the contact face showed thickness values ranging from 0.002 inch to 0.007 inch. The coating measurements and missing areas generally showed features consistent with wear. Some areas of thickness exceeding specification were measured, but those areas were isolated and located near the edges of the contact area and were likely associated with a local disbond of the coating. On the convex side of blades G and I, the thickness differential between the missing coating at the forward edge of the contact

face and the area approximately 2 inches from the forward face was approximately 0.004 to 0.005 inch.

After the coating was stripped from blades C through I, the blades were inspected using UT and ECI. A signal response above threshold was detected on blade D using ECI, but the signal was found to be associated with a scratch. No other indications were returned from ECI of the remaining blades. None of the blades had a UT indication.

After the NDI was completed, a destructive examination of blades C through I was conducted using an SEM. The surfaces of the dovetail on the convex side approximately 2 inches from the forward face were etched with a modified Kroll's reagent to facilitate an examination for evidence of microcracks. A network of microcracks was detected on blade D as shown in figure 54. A cross-section through the areas of cracks shown in figure 54 was cut and polished for metallographic examination. A crack extending 0.0008 inch deep was observed in the polished plane.

Residual stress measurements were completed at Safran on blades D and F on the convex side of the blade contact face approximately 2 inches from the forward end, and the resulting residual stress profiles are shown in figure 55. Both blades showed profiles comparable to the expected reference residual stress profile (see figure 47) with a minimum residual stress value of approximately -650 megapascals in each blade.

Beginning in January 2017, UT inspections during blade lubrication were initiated at Southwest Airlines. In March 2017, a service bulletin was issued to require UT inspections at each blade lubrication for all fan blades with at least 9,000 engine flight cycles since the last shop visit. The service bulletin was revised in June 2017 to limit the UT inspections to fan blades with at least 15,000 engine flight cycles since last shop visit. In May 2017, a blade with a UT indication above the threshold value was found during blade lubrication inspections at Southwest Airlines. The blade was removed from service and sent to GE for examination. This blade is labeled blade J for reference in this report.

Blade J had originally been manufactured as a part number 340-001-022 blade and had the dovetail rework in 2001, resulting in a part number change to 340-001-027. The blade had accumulated approximately 33,020 CSN. The blade had been overhauled at PTI Technologies during a May 2015 shop visit, and the part number changed to 340-001-039 after the overhaul. Prior to that, the blade had been overhauled at GKN Aerospace during a June 2007 shop visit. The blade had accumulated 2,876 cycles since the last overhaul and had accumulated 18,429 cycles since the overhaul in 2007.

The UT indication was located on the convex side of the blade approximately 2 inches from the forward end. The Cu-Ni-In coating was intact with no apparent areas of wear or missing coating. The thickness of the coating was not assessed.

The coating was stripped to facilitate a direct examination of the blade surface in the area of the UT indication. Next, the area of the UT indication was sectioned from the rest of the dovetail for SEM examination. An initial examination of the surface in the as-

stripped condition was inconclusive, so the surface was etched using a modified Kroll's reagent. An SEM image of the resulting surface is shown in figure 56. A crack approximately 0.153 inch long in the area of the UT indication was observed as shown labeled in figure 56. The crack was surrounded by a network of microcracks. The crack and microcracks were all located in the peened and grit blast region near the outboard edge of the contact face previously covered by the Cu-Ni-In coating. No cracks or microcracks were detected in the adjacent shot-peened only surface.

Next, the dovetail in the area of cracking was sectioned to facilitate a lab fracture to open the crack. Views of the resulting fracture surface are shown in figure 57. A portion of the fracture surface had a flat fracture with curving boundaries, features consistent with fatigue. The crack emanated from multiple origins located at the blade surface as indicated with an unlabeled bracket in the upper image in figure 57. Additional fatigue regions were observed forward and aft of the main crack. Unlabeled arrows in figure 57 indicate the origin areas for these other fatigue regions.

A closer view of the crack feature is shown in figure 57. The lab cut used to open the crack intersected the crack tip at the location of deepest penetration. In the lower image, the dashed lines furthest from the origin area indicate the fatigue boundary that intersected the lab cut. As shown in figure 57, the crack extended approximately 0.061 inch deep and was 0.153 inch long at the surface. A prominent crack arrest line was visible on the surface, also indicated with a dashed line in the lower image in figure 57. The crack arrest line extended up to approximately 0.030 inch deep and was 0.116 inch long at the surface.

The fracture surface was examined using an SEM, and features consistent with low-cycle fatigue were observed. The fracture features in the fatigue region were surveyed to estimate the number of striations within the region of crack growth. Striation density as a function of distance from the origin for the blade J crack is shown plotted in figure 58, and a power-law curve was fitted to the data as indicated by the solid line. Data for accident blades 23 and 24 are also shown for reference, and the curve for blade J was located between the curves for accident blades 23 and 24. By integrating the power-law equation for the curve fitted to the blade J data from a depth of 0.0005 inch to the crack tip at 0.061 inch, the total number of cycles of crack growth was estimated to be approximately 9,000 cycles. By integrating from the prominent crack arrest line at a depth of 0.03 inch to the crack tip at 0.061 inch, the estimated number of cycles between the arrest line and the boundary was estimated to be approximately 2,500 cycles, comparable to the 2,800 cycles accumulated by the blade since the last overhaul.

A transverse cross-section was cut through the dovetail of blade J forward of the crack in an area where microcracks were observed. The polished and etched cross-section showed a typical microstructure for the specified Ti-6-4 alloy with no surface anomalies. Several microcracks were observed emanating from the surface with measured depths up to 0.0016 inch deep.

Next, the blade J dovetail was sent to the Safran Aircraft Engines laboratory for residual stress analysis. Residual stress in the longitudinal direction was obtained on the convex side of the dovetail in the contact face forward and aft of the segment that had been removed for the fractographic and metallographic examinations. Measurements in both locations showed residual stress profiles that had a shape comparable to the reference residual stress profile (see figure 47) with minimum residual stress values of -620 megapascals at the forward location and -650 megapascals at the aft location. Although a profile comparable to the reference profile was obtained, the blade had been shot-peened at the last overhaul that occurred after the crack initiated based on service history data and crack growth estimates derived from the fractographic examination.

In November 2017 and in April 2018, fan blades with cracks were detected during inspections at blade overhaul. For reference in this report, the blades are labeled blades K and L, respectively. Blade K was owned by a domestic operator (not Southwest Airlines) and blade L was owned by a foreign operator.

Blade K was a part number 340-001-026 blade manufactured after CFM International CFM56-7B Service Bulletin 72-0253 was published and was originally manufactured with the tapered dovetail profiles designed to prevent spalling and wear damage to the contact face coatings. The blade had accumulated approximately 19,497 CSN. The blade was rejected due to an ECI response above threshold during the November 2017 overhaul at GKN Aerospace. Reportedly, the blade had been overhauled previously at GKN Aerospace in February 2004 and again in November 2008.

Blade K was sent to the Safran Aircraft Engines laboratory for further examination. The coating had been stripped at the overhaul shop, and the coating condition and thickness were not reported. Crack indications were detected at Safran by FPI and ECI on the convex side of the dovetail near the outboard edge of the grit blasted contact surface approximately 1 inch from the forward face.

The surface was etched with 15/15 reagent and then examined using an optical microscope and an SEM. A network of microcracks was detected within approximately 0.079 inch of the outboard edge of the grit-blasted surface. Some of the cracks were linked together in the area where the FPI and ECI indications were detected.

Next, the dovetail in the area of cracking was sectioned to facilitate a lab fracture to open the crack. A portion of the fracture surface had a flat fracture with curving boundaries, features consistent with fatigue. The fatigue features emanated from multiple origins located at the blade surface approximately 1.1 inch from the forward face. The extent of the fatigue crack was 0.33 inch long at the surface and up to 0.017 inch deep.

The fracture surface was examined using an SEM, and features consistent with low-cycle fatigue were observed. The fracture features in the fatigue region were surveyed to estimate the number of striations within the region of crack growth. A power-law curve was fitted to the striation density data as a function of distance from the origin for the blade K crack, and by integrating the power-law equation for the curve fitted to the

blade K data from a depth of 0.0005 inch to the crack tip at 0.017 inch, the total number of cycles of crack growth was estimated to be approximately 6,800 cycles.

Next, Safran completed residual stress profile measurements on the blade K dovetail. Residual stress profiles in the longitudinal direction were measured on the convex side of the dovetail in the contact face at locations approximately 1 inch, 1.6 inches, and 2.5 inches from the forward face. A residual stress profile on the contact face of the dovetail approximately 1 inch from the leading edge was also measured on the concave side. Abnormal residual stress profiles were measured at each location. Adjacent to the crack location on the convex side, the minimum compressive stress value was approximately -420 megapascals at a depth of 0.0013 inch.

Blade L was a part number 340-001-027 blade originally manufactured as a part number 340-001-022 blade. The blade had accumulated approximately 30,000 CSN to 34,000 CSN. Based on a search of GKN Aerospace records, the blade had been overhauled at GKN Aerospace in August 2009.

Blade L was sent to the Safran Aircraft Engines laboratory for further examination. The coating had been stripped at the overhaul shop, and the coating condition and thickness were not reported. Crack indications were detected at Safran on the unetched surface by FPI and ECI on the convex side of the dovetail near the outboard edge of the grit blasted contact surface approximately 1.7 inch from the forward face.

The surface was etched with 15/15 reagent and then examined using an optical microscope and an SEM. A network of microcracks was detected within approximately 0.063 inch of the outboard edge of the grit-blasted surface located between 1.42 inches and 2.01 inches from the forward face. Some of the cracks were linked together in the area where the FPI and ECI indications were detected.

Next, the dovetail in the area of cracking was sectioned to facilitate a lab fracture to open the crack. A portion of the fracture surface had a flat fracture with curving boundaries, features consistent with fatigue. The fatigue features emanated from multiple origins located at the blade surface. The extent of the fatigue crack was 0.22 inch long at the surface and up to 0.017 inch deep.

The fracture surface was examined using an SEM, but much of the fatigue region was damaged by fracture surface recontact damage. Some areas showing fatigue striations were observed, and striation density measurements in these areas were within the range of data gathered from other examined cracks. However, the extent of available data was insufficient to confidently fit a curve to estimate the total number of striations associated with crack growth.

Next, Safran completed residual stress profile measurements on the blade L dovetail. Residual stress profiles in the longitudinal direction were measured on the convex side of the dovetail in the contact face at locations adjacent to the opened crack and another area near the crack. A residual stress profile on the contact face of the

dovetail was also measured on the concave side opposite from the crack location. Abnormal residual stress profiles were measured at each location on the convex side, and a normal residual stress profile was measured on the concave side. Adjacent to the crack location on the convex side, the minimum residual stress value was -250 megapascals measured at depths of 0.0005 inch, 0.0010 inch, and 0.0028 inch. On the concave side, the minimum residual stress value was -540 megapascals at a depth of 0.0017 inch.

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Senior Materials Engineer

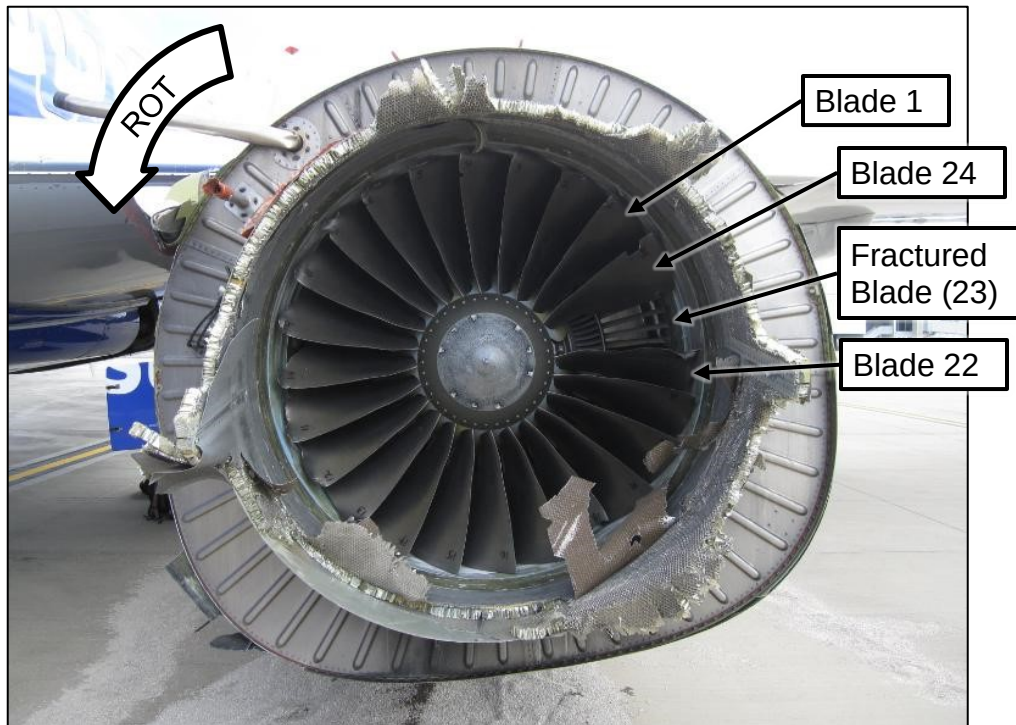


Figure 1. Overall view of the forward end of the number 1 engine after the emergency landing.

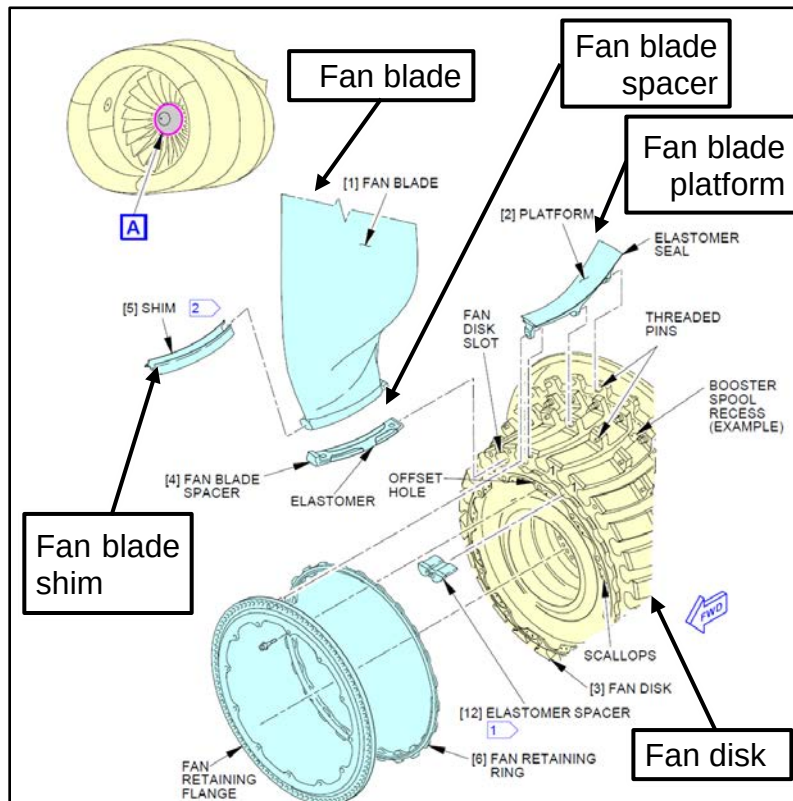


Figure 2. Fan components in the CFM56-7B engine as presented in the Boeing Aircraft Maintenance Manual.

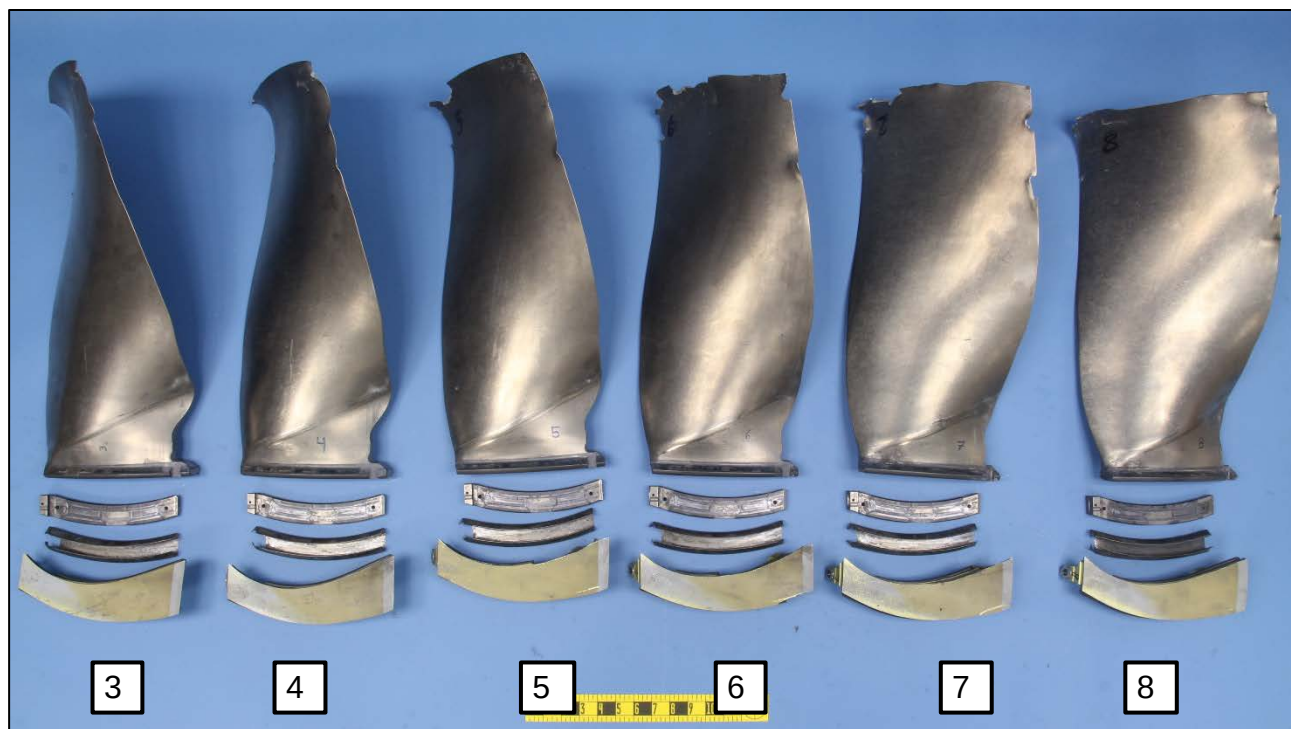
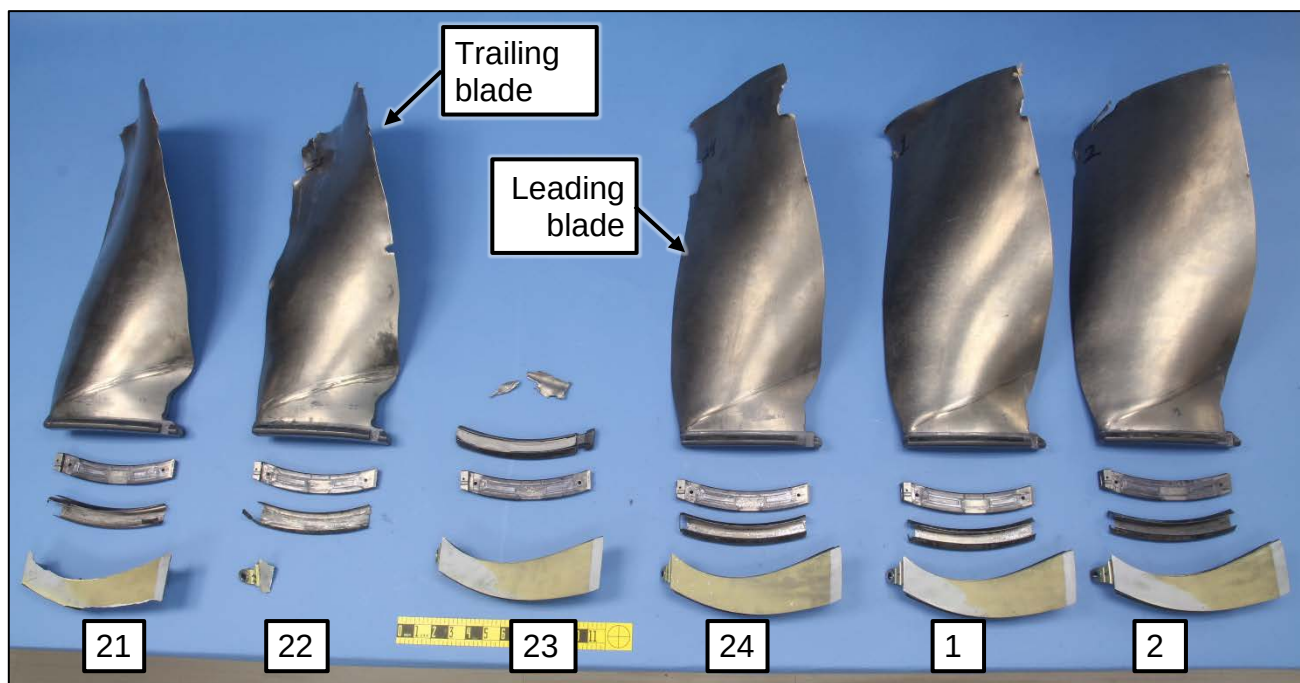


Figure 3. Overall view of submitted fan blades, fan blade spacers, fan blade shims, and fan blade platforms (shown positioned from top to bottom in each image) for blades 1 through 8 and 21 through 24. The shim for blade 23 was received on the dovetail. Blades leading and trailing the fractured blade 23 are indicated.

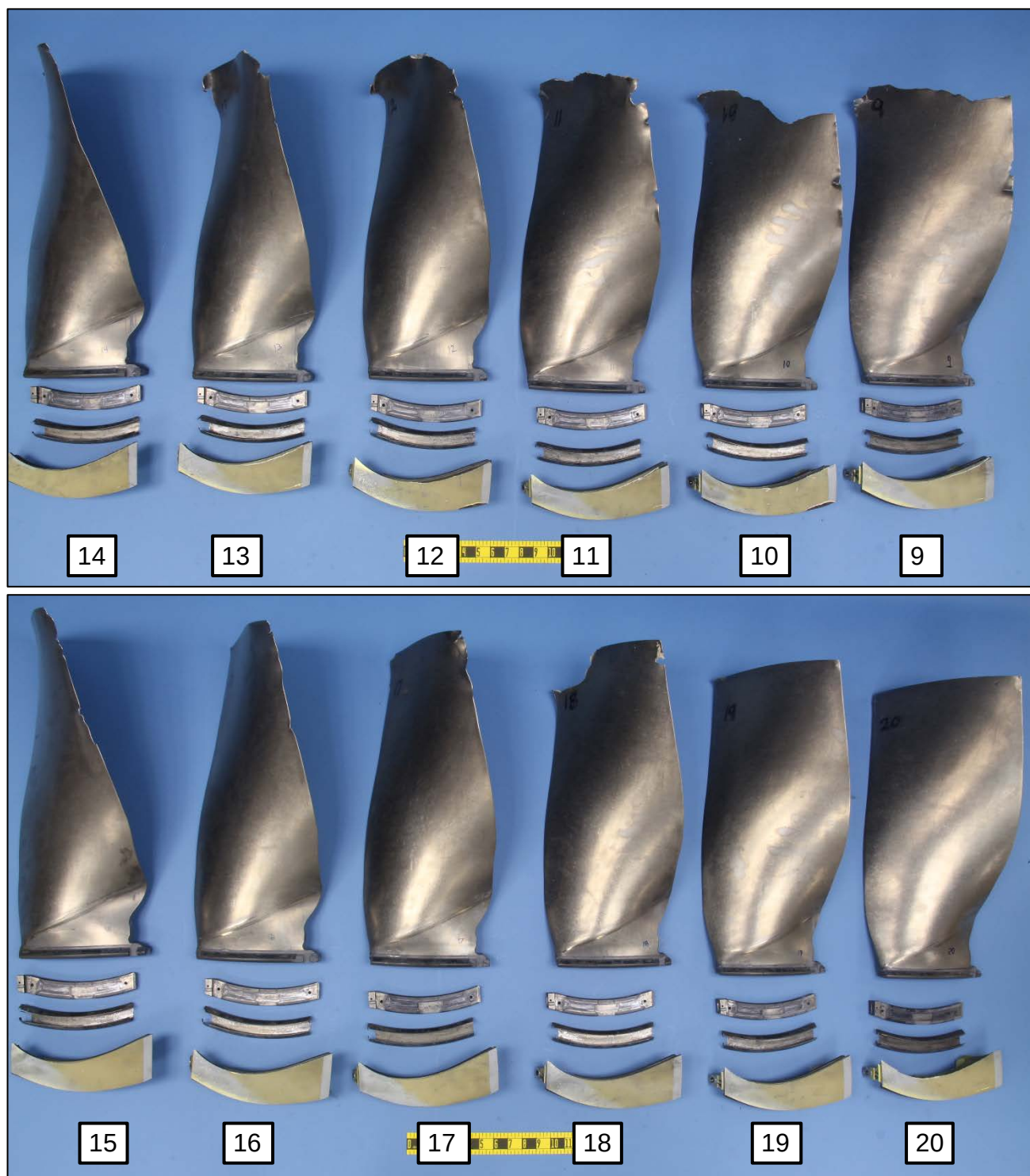


Figure 4. Overall view of submitted fan blades, fan blade spacers, fan blade shims, and fan blade platforms (shown positioned from top to bottom in each image) for blades 9 through 20.



Figure 5. Overall view of the blade 23 fracture surface as received. A bracket indicates the fatigue origin area located approximately 2.1 inches aft of the forward end on the convex side of the blade.

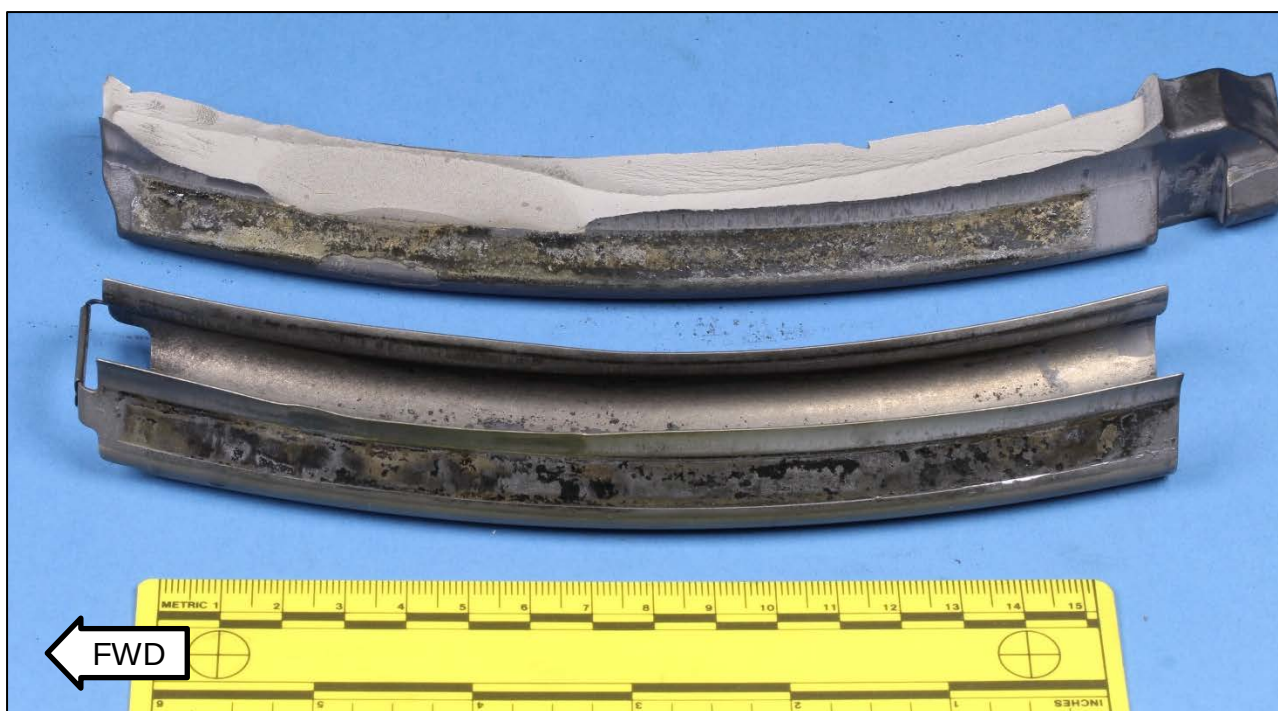


Figure 6. Overall view of the blade 23 dovetail piece and fan blade shim after the shim was disassembled from the dovetail.

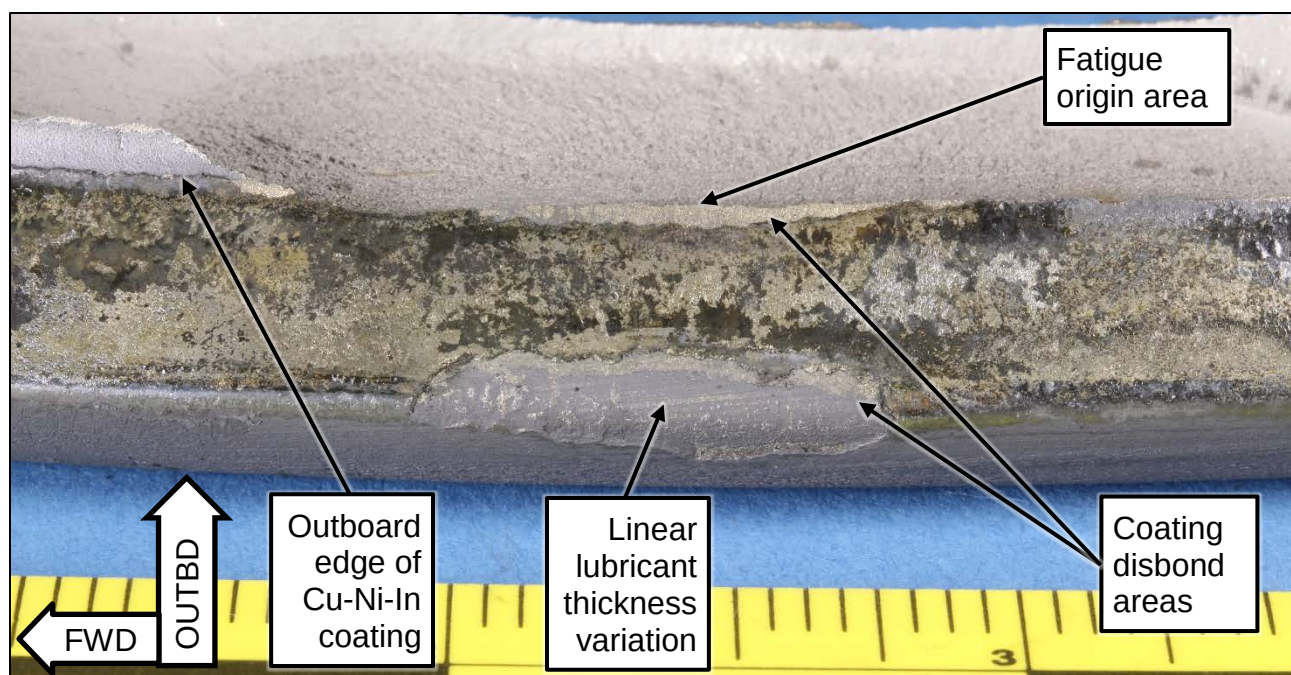


Figure 7. View of the convex face of the blade 23 dovetail at the fatigue origin area.

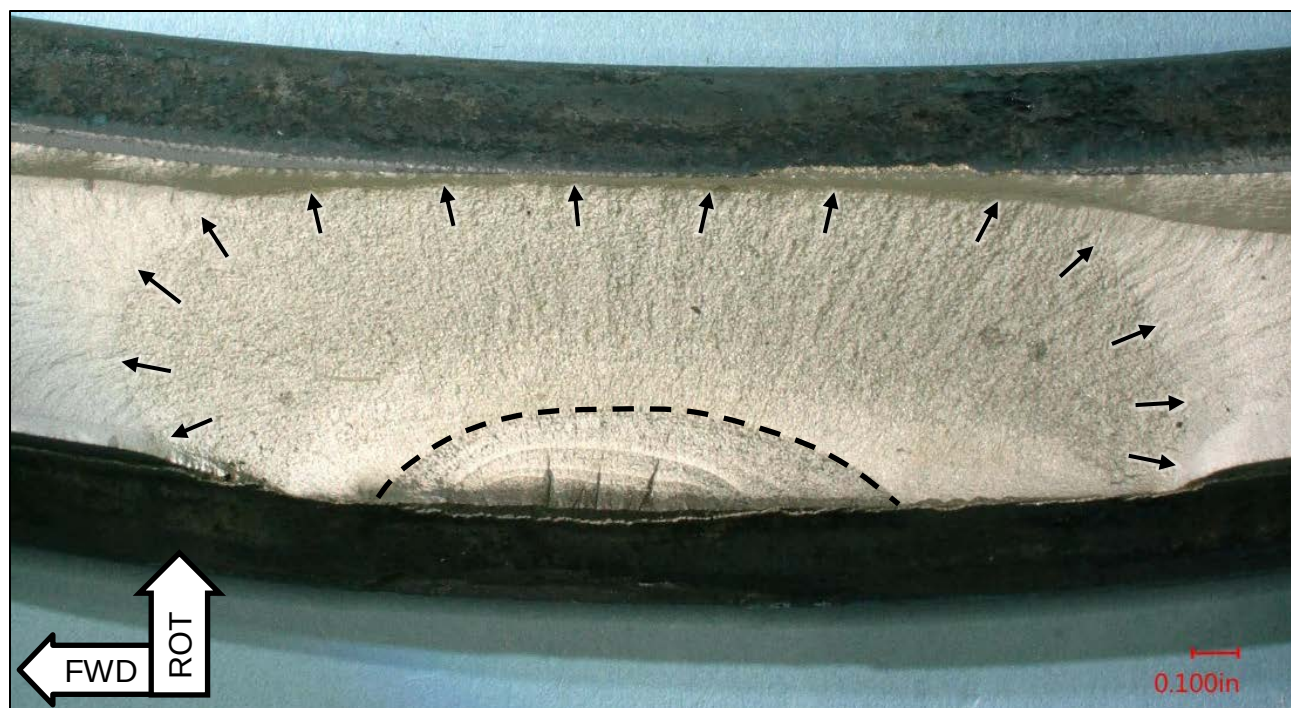


Figure 8. Close view of the fatigue region after removal of the shim. A dashed line indicates the location of the last well-defined crack arrest mark.



Figure 9. View of the dovetail on the convex side of blade 17 near the forward end of the blade. Circled areas were areas with missing Cu-Ni-In coating. An unlabeled bracket indicates the width of the contact face.

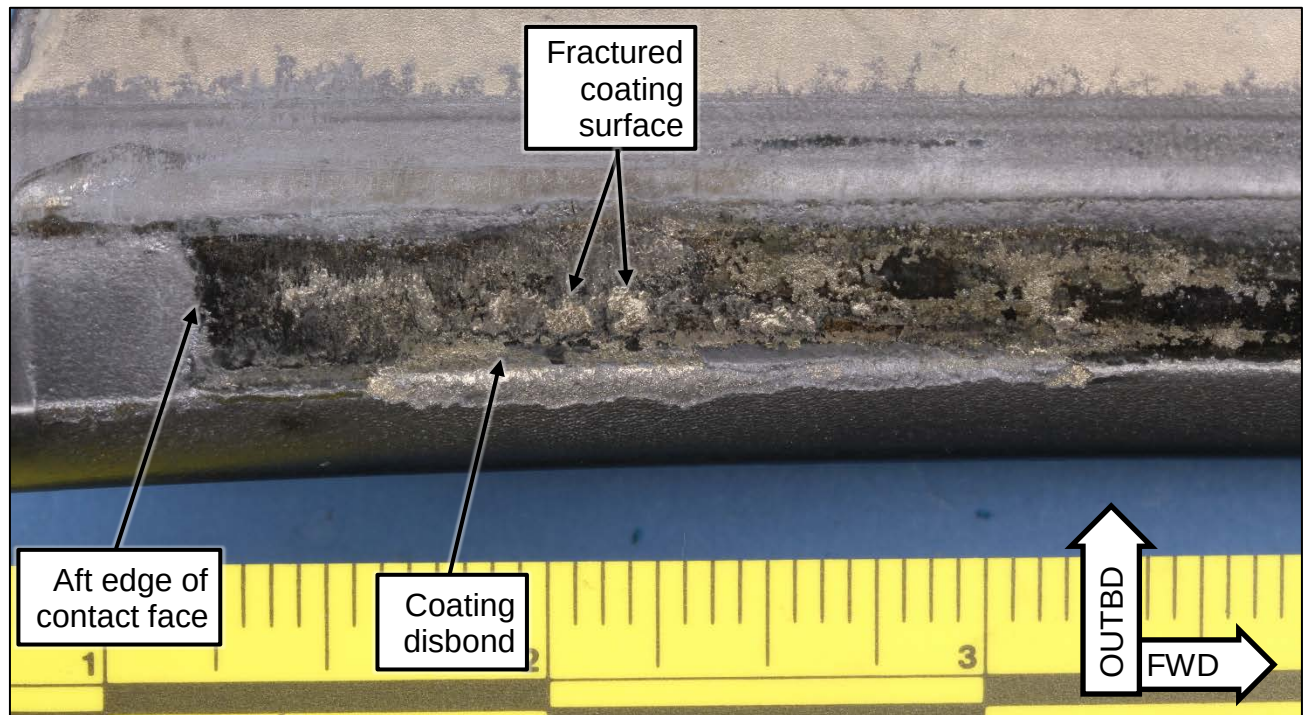


Figure 10. Close view of the Cu-Ni-In coating disbond and flaking of the adjacent lubricant layer on the concave face at the inboard edge of the contact face near the aft end of the contact face.

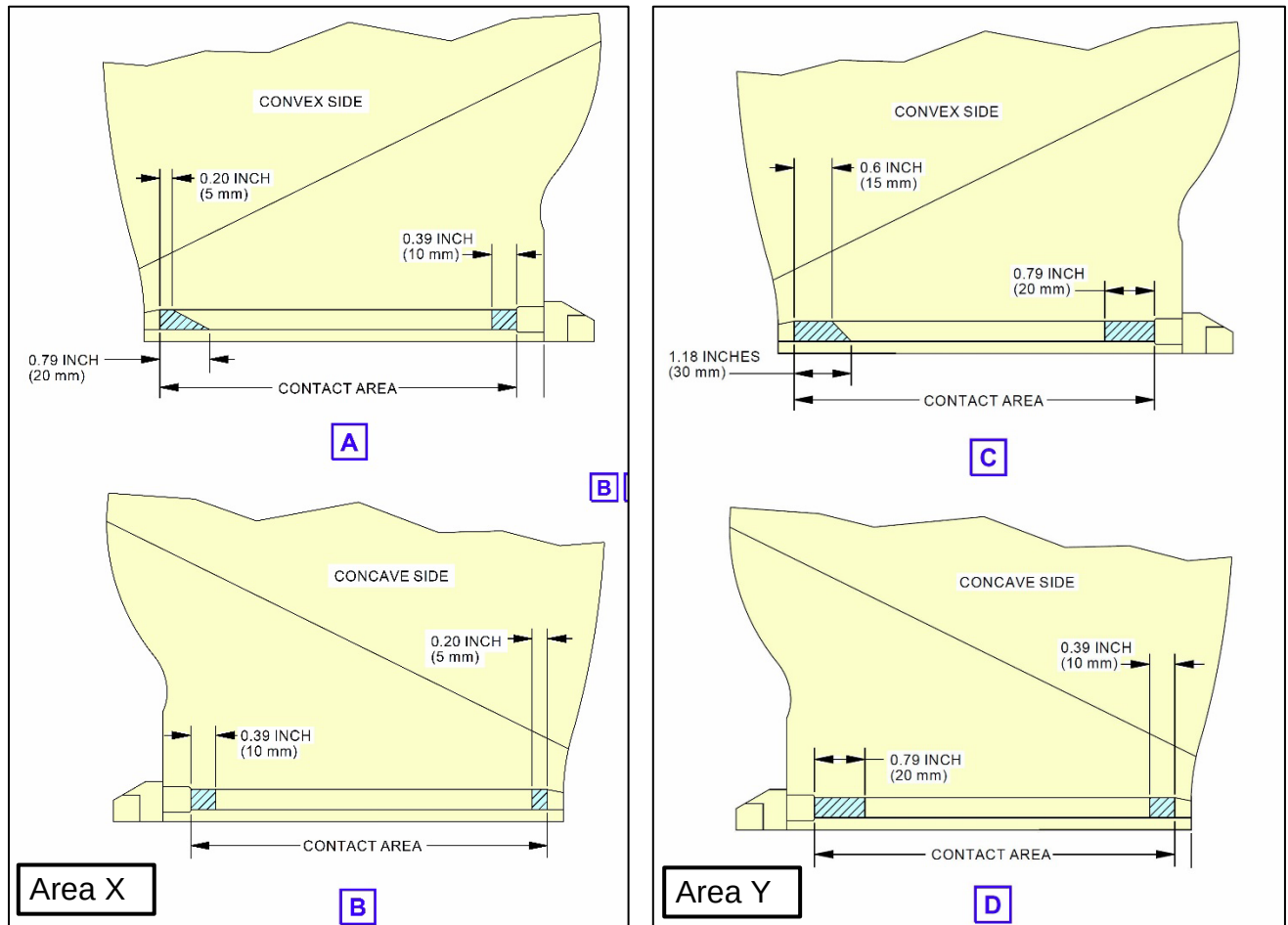


Figure 11. Drawings from the Boeing Aircraft Maintenance Manual in effect at the time of the last blade lubrication showing limits of allowable coating damage on the dovetail contact faces. Areas X and Y were defined by the green hatched areas in the drawings shown on the left and on the right, respectively.



Figure 12. Blade 19 dovetail showing a visual appearance of the Cu-Ni-In coating with a mix of smoother dark gray or black areas mixed with relatively rougher light gold colored areas.



Figure 13. Inboard face of the blade 23 dovetail.

Figure 14. Fan blade shim illustration provided by Safran Aircraft Engines. The original shim design had tabs (shown in blue) separated by a gap. In shims manufactured after the design change introduced in 2008, the tabs are connected with a bridging piece illustrated in red.

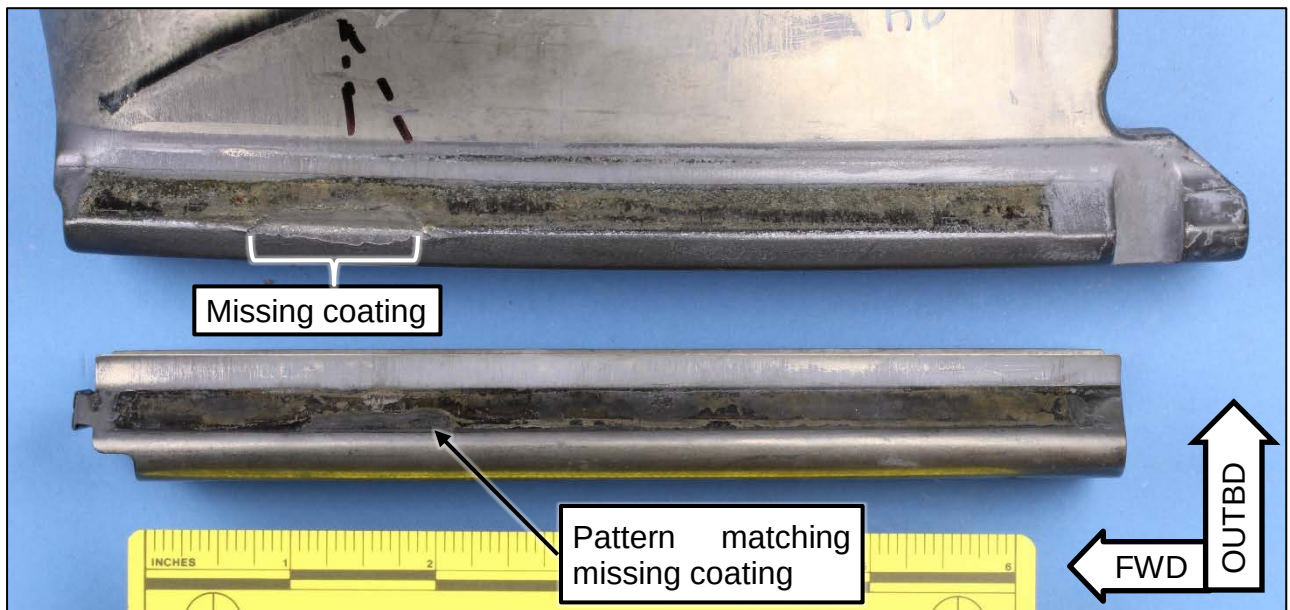
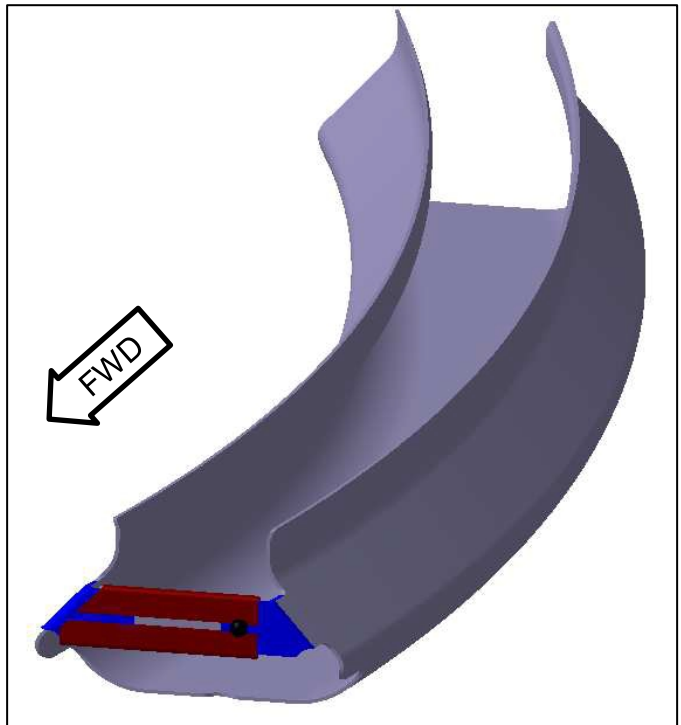


Figure 15. Blade 24 dovetail and shim. A pattern matching the area of missing Cu-Ni-In coating was visible on the shim exterior contact face.

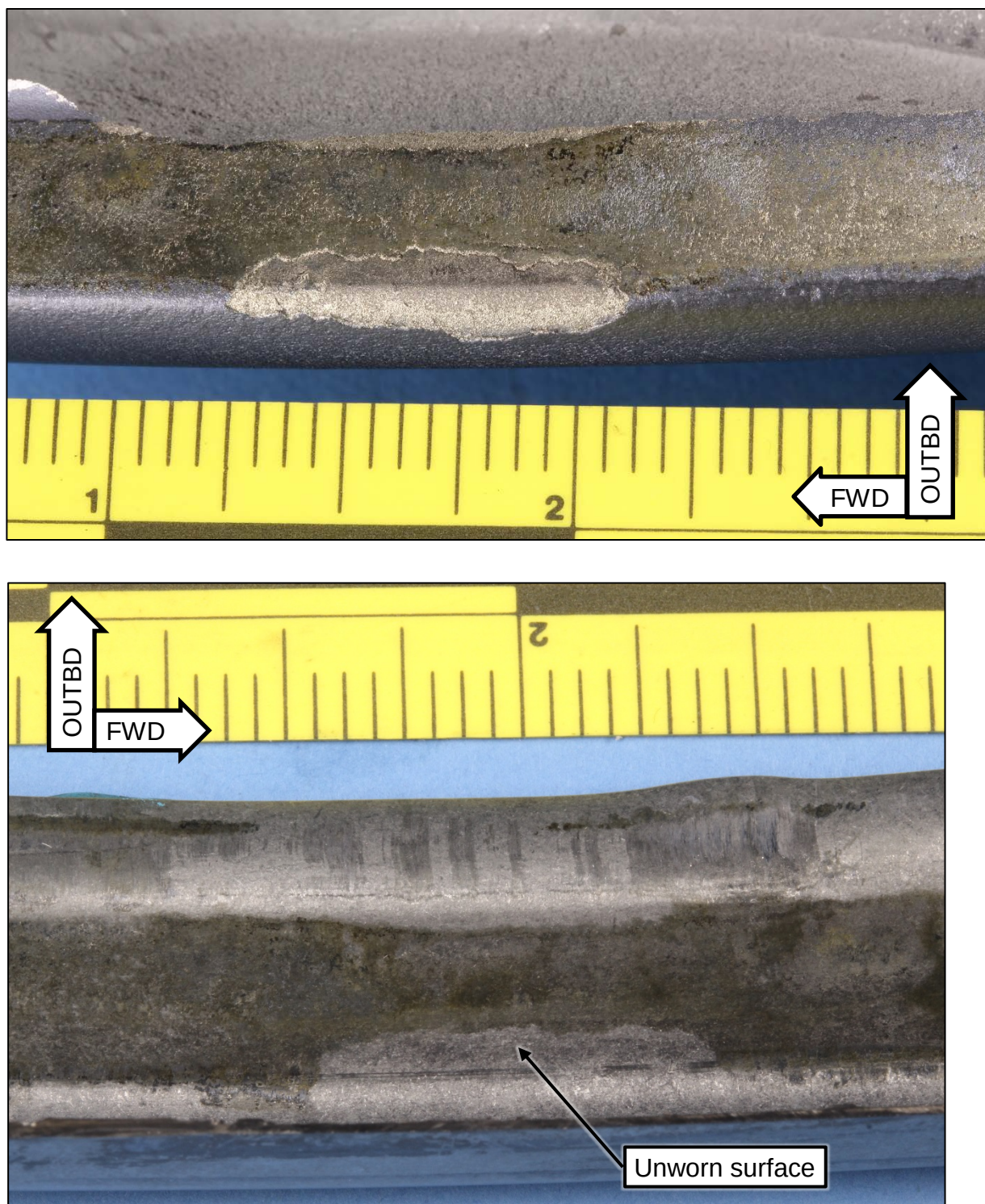


Figure 16. Views of the blade 23 dovetail at the fatigue origin area after cleaning (upper image) and the cleaned mating shim contact surface (lower image).

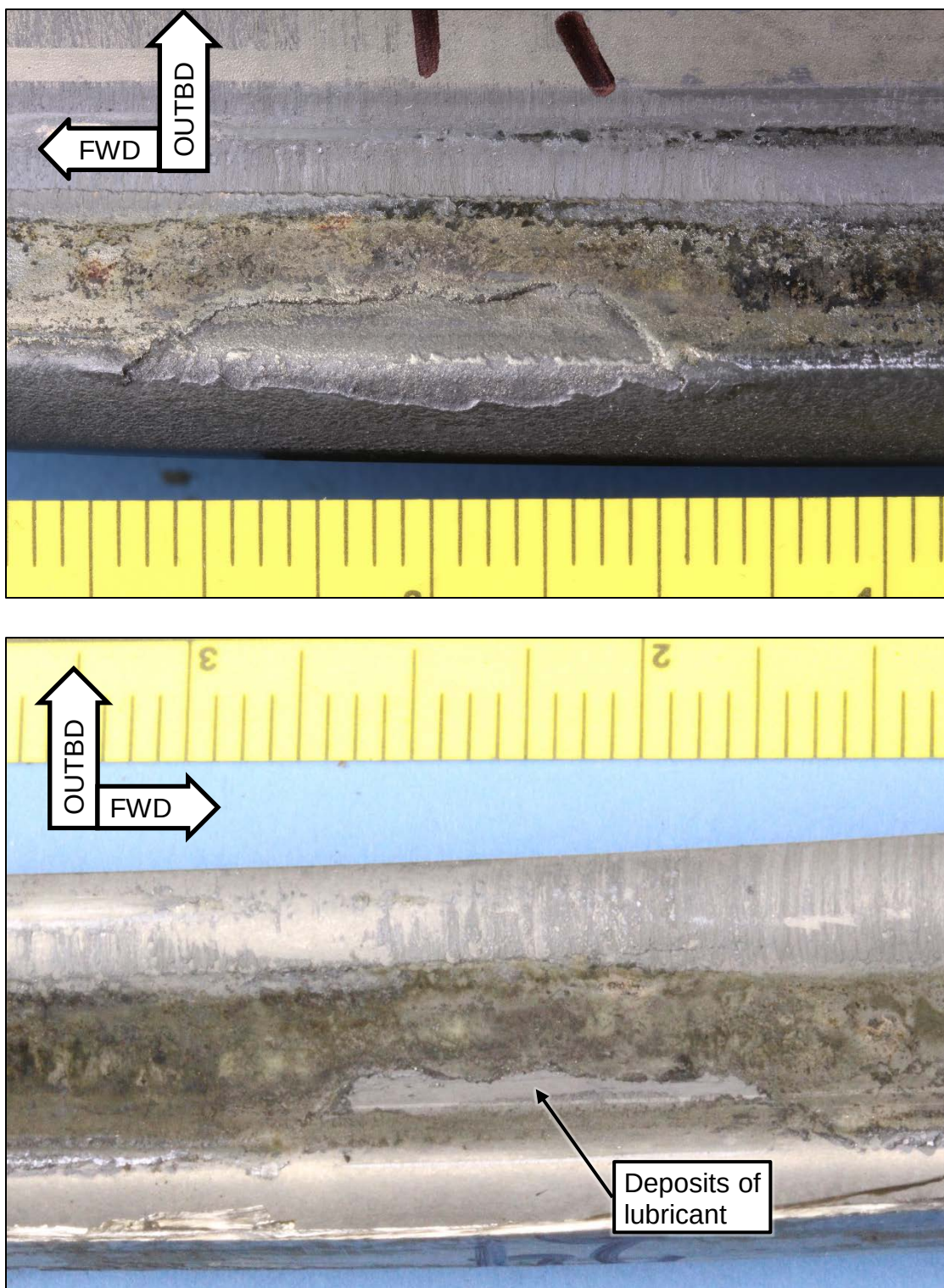


Figure 17. Coating disbond area at the convex side of the blade 24 dovetail (upper image) and the mating contact face on the shim (lower image).

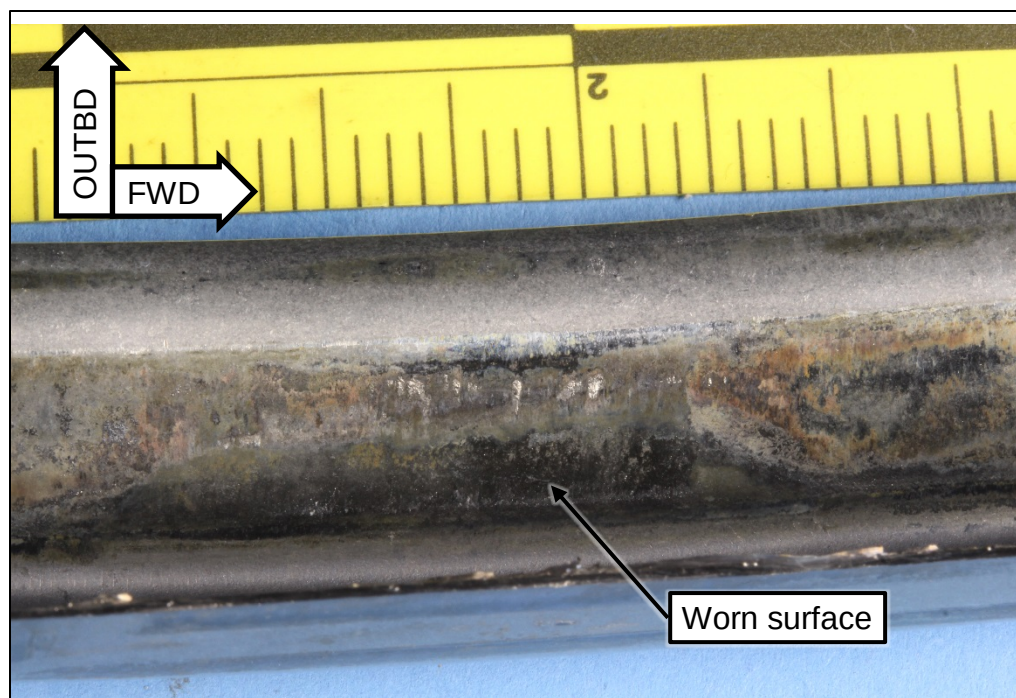
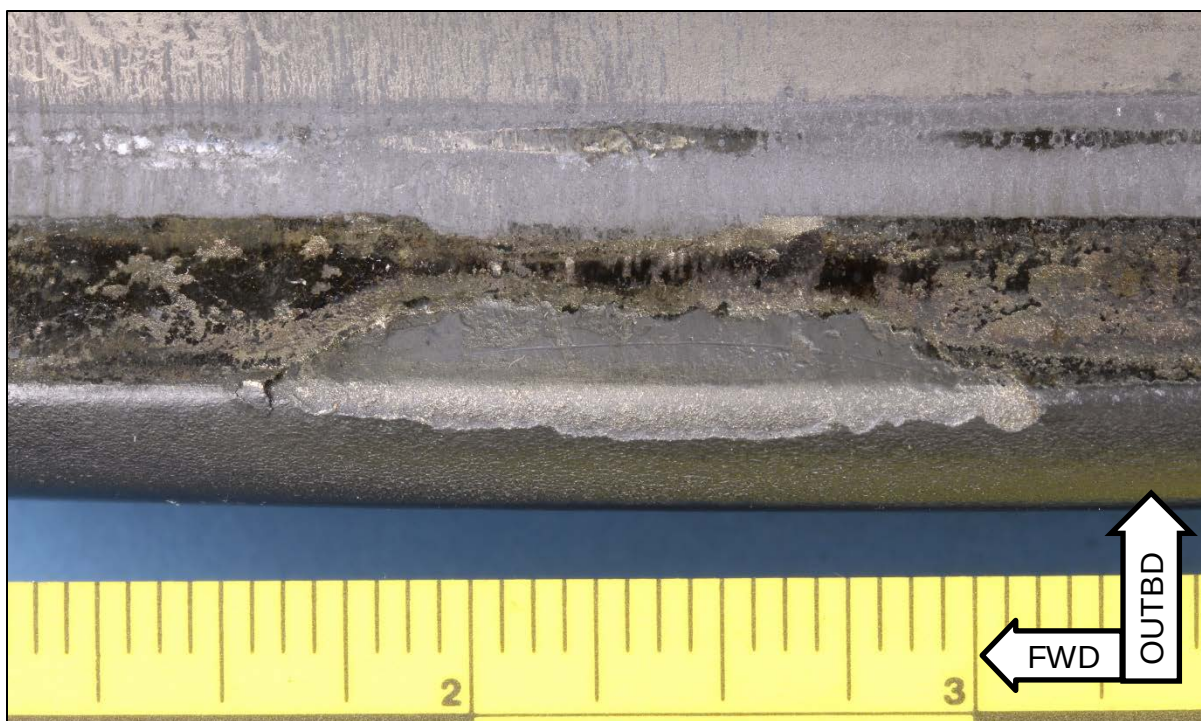


Figure 18. Coating disbond area at the convex side of the blade 19 dovetail after cleaning (upper image) and the cleaned mating contact face on the shim (lower image).

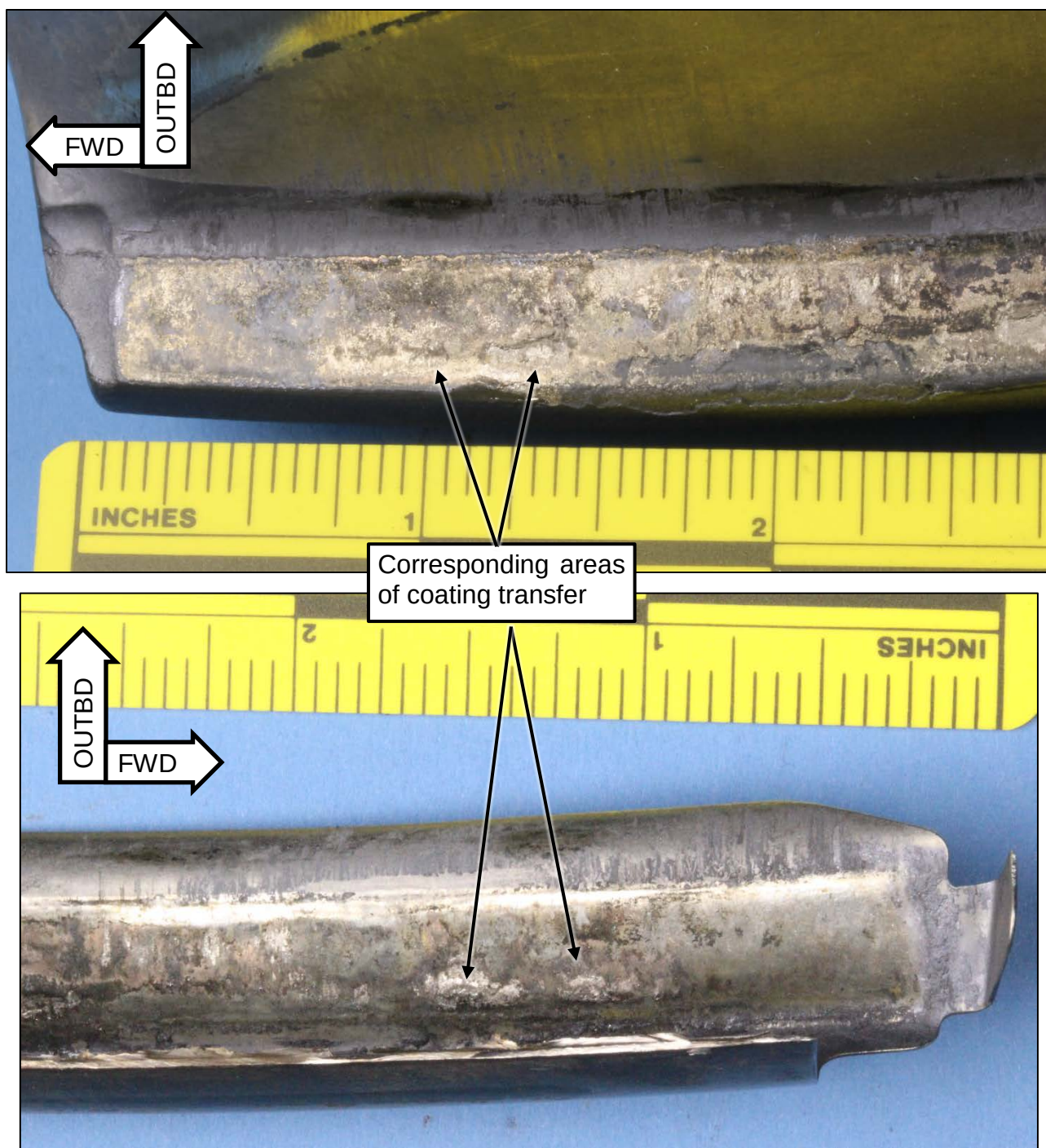


Figure 19. Coating disbond area at the convex side of the blade 2 dovetail after cleaning (upper image) and the cleaned mating contact face on the shim (lower image).

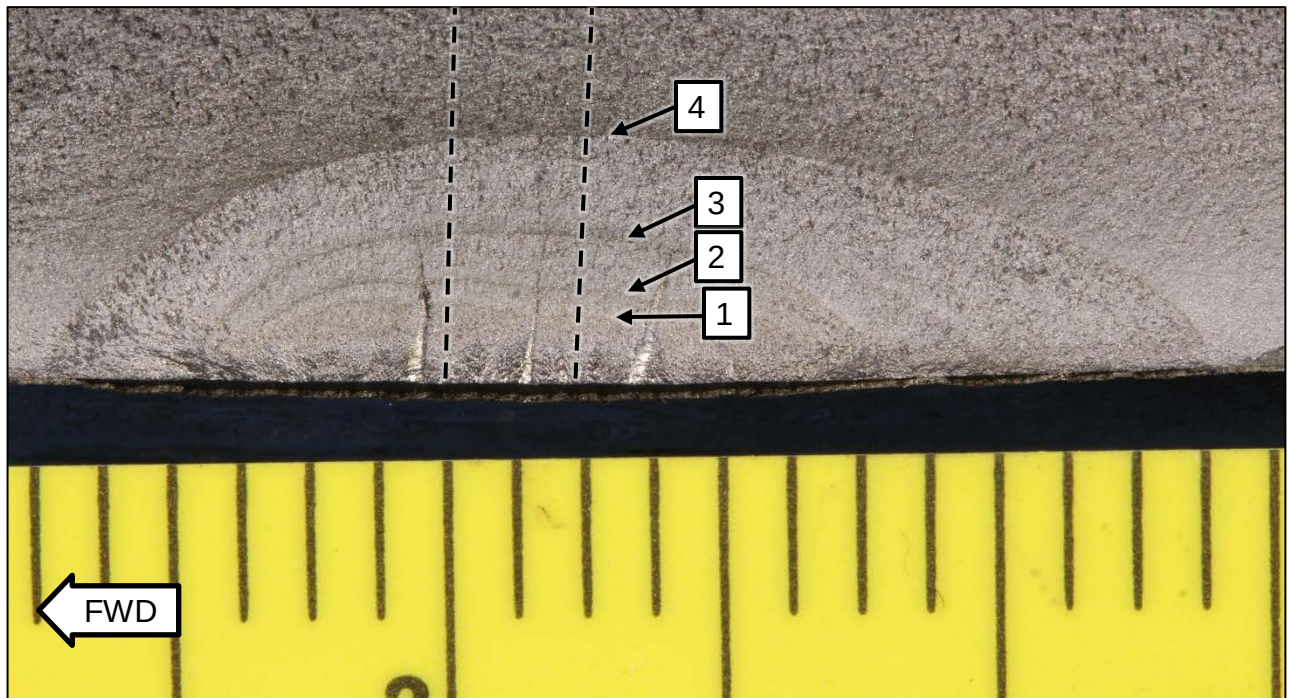


Figure 20. Close view of the cleaned origin area on blade 23. Distinct crack arrest lines are numbered, and dashed lines indicate locations where striation density was counted.

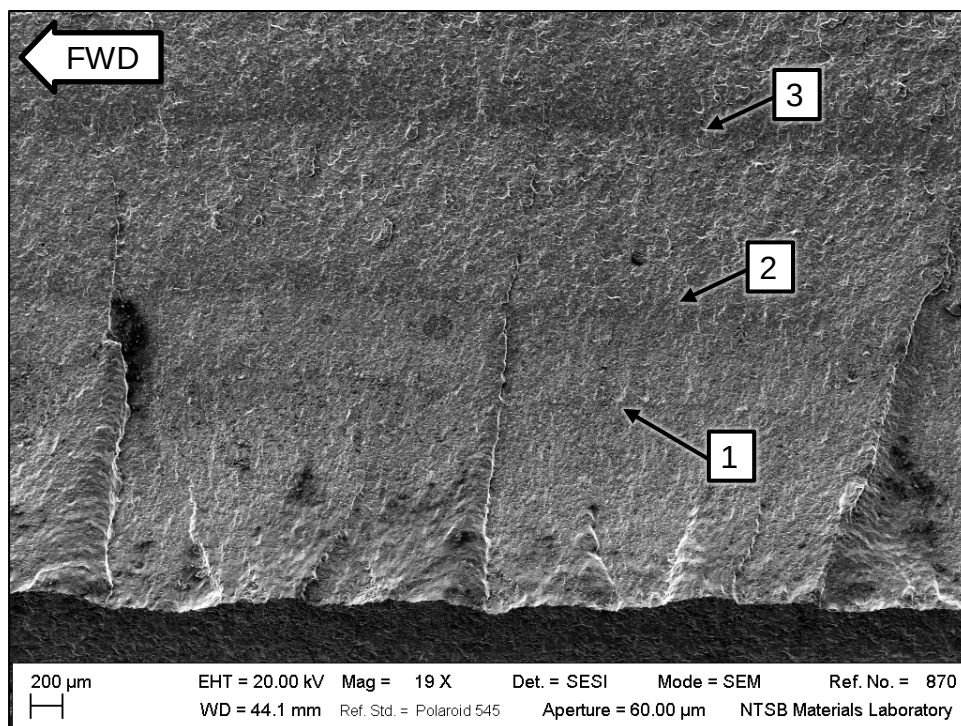


Figure 21. Overall view of the origin area after cleaning as viewed using an SEM.

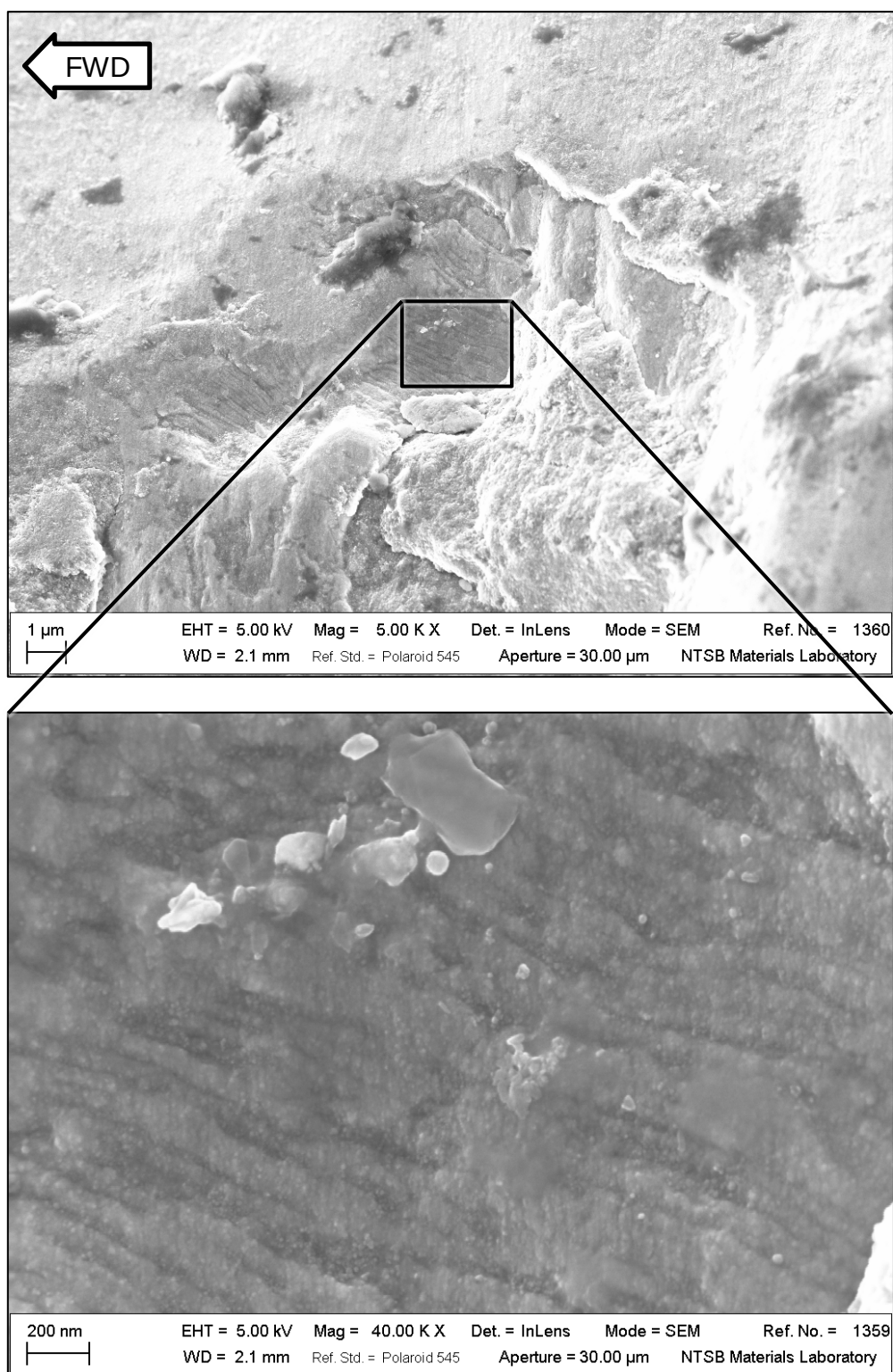


Figure 22. SEM images of the blade 23 fracture surface at 0.0082 inch from the convex surface.

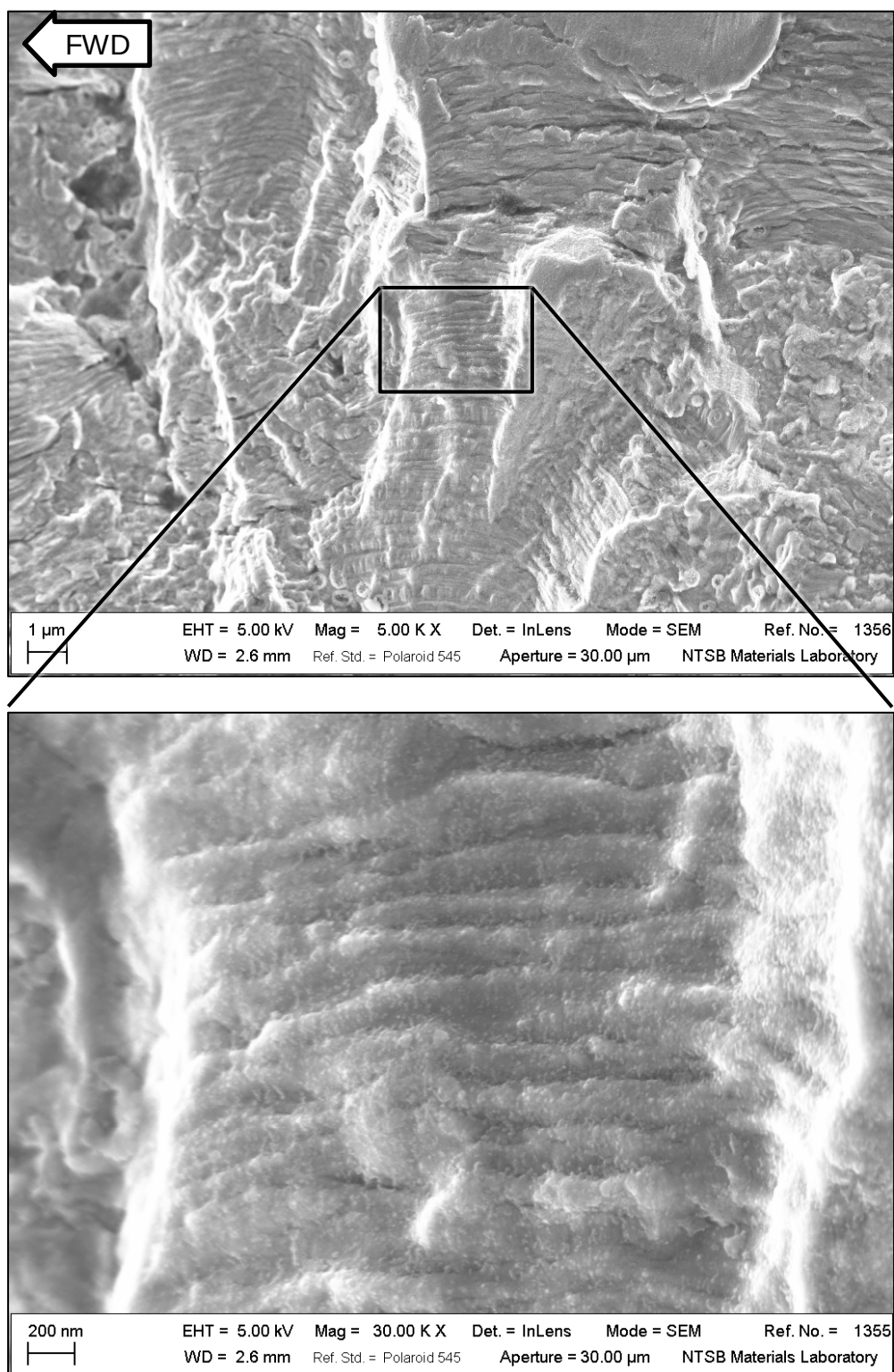


Figure 23. SEM images of the blade 23 fracture surface at 0.031 inch from the convex surface.

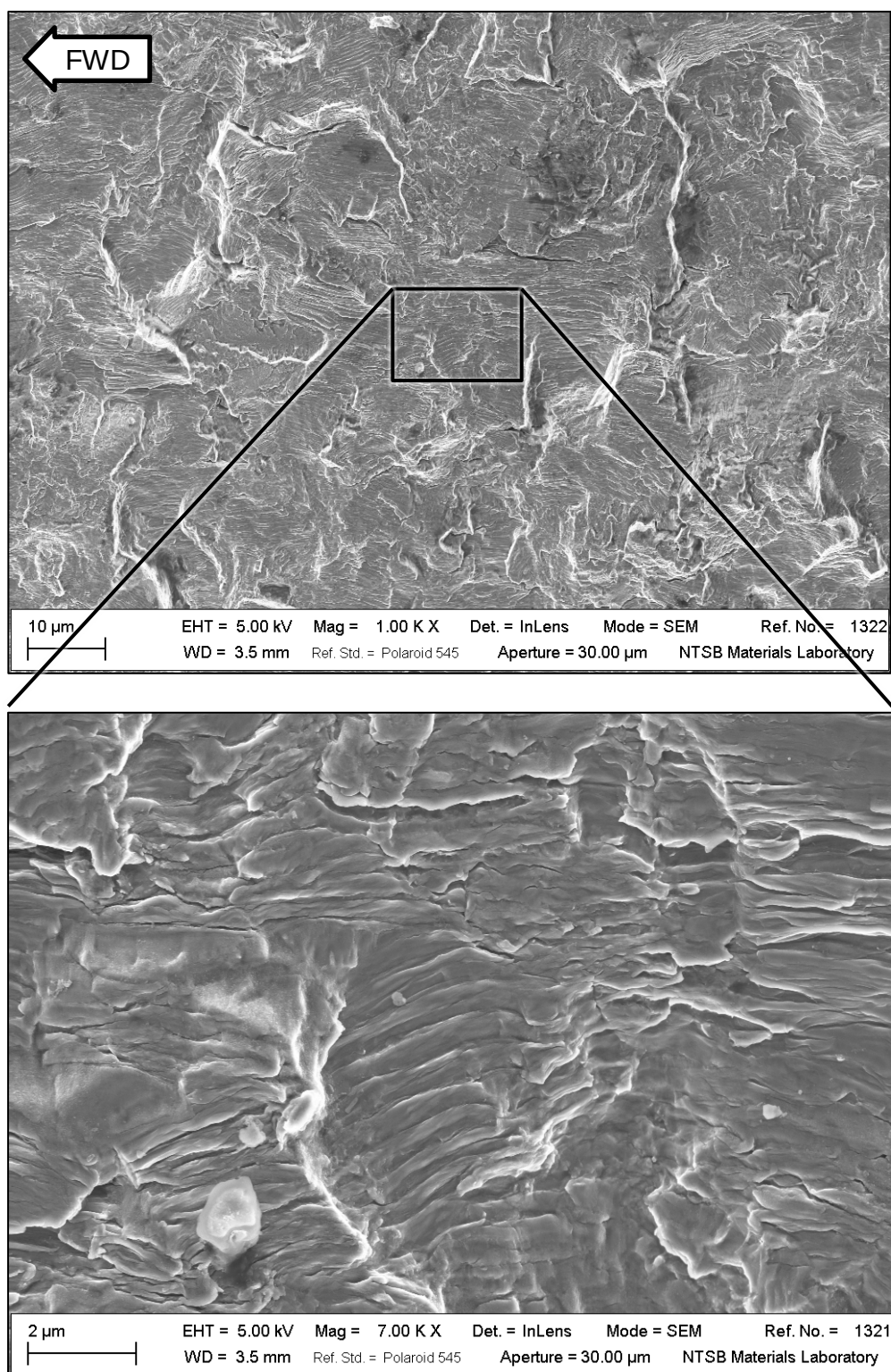


Figure 24. SEM images of the blade 23 fracture surface at 0.098 inch from the convex surface.

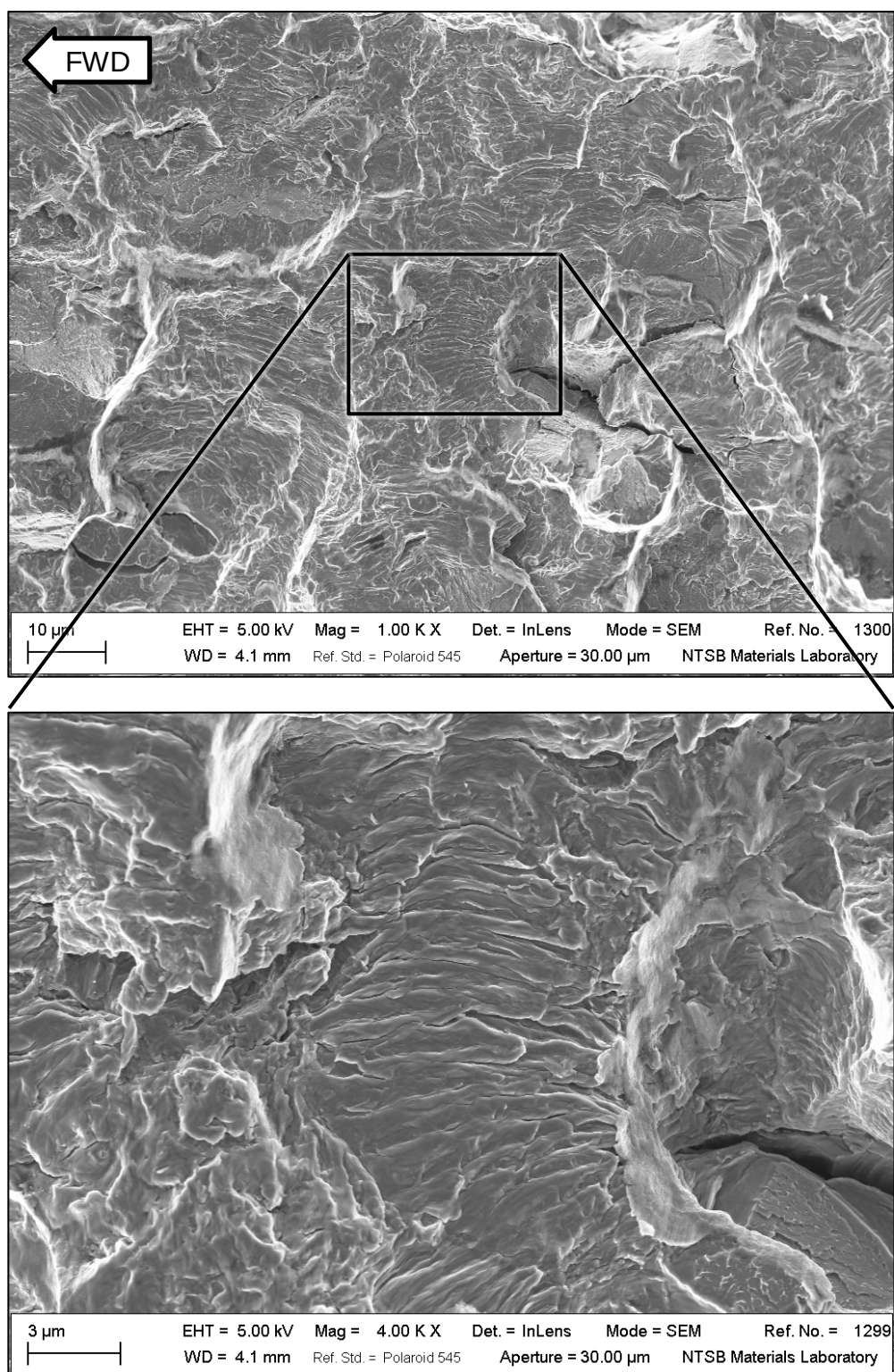


Figure 25. SEM images of the blade 23 fracture surface at 0.187 inch from the convex surface.

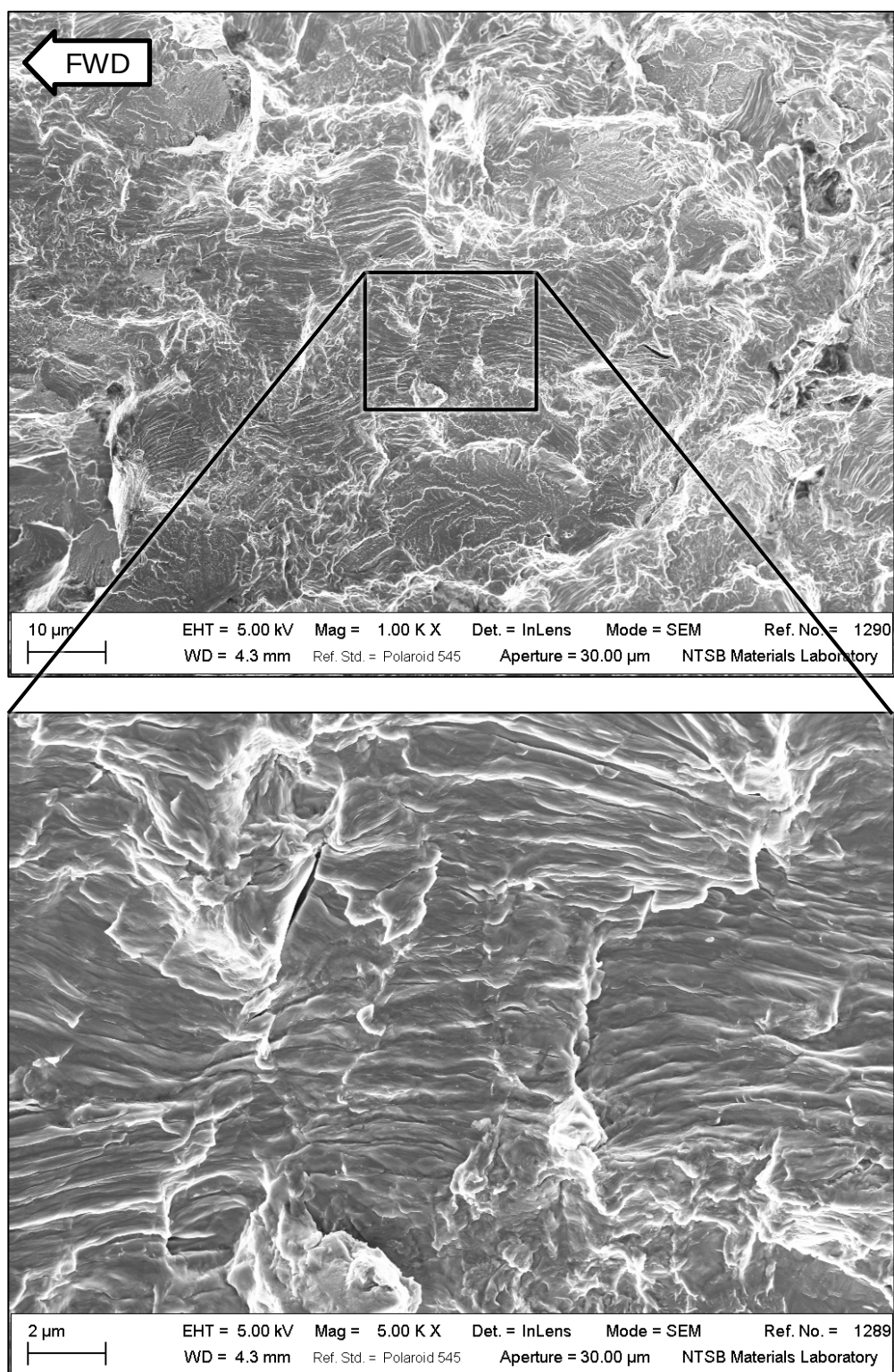


Figure 26. SEM images of the blade 23 fracture surface at 0.249 inch from the convex surface.

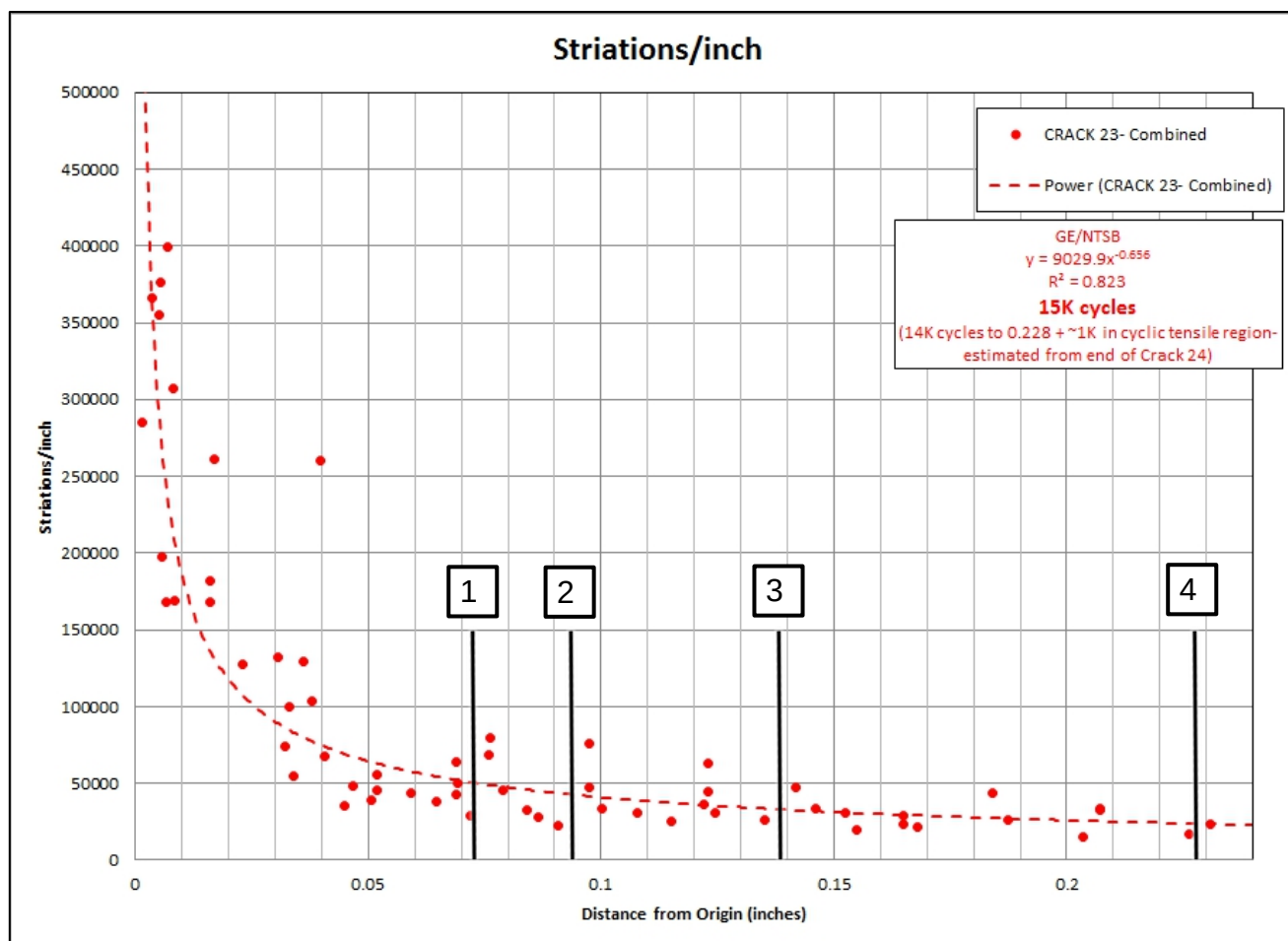


Figure 27. Striation density as a function of distance from the origin for blade 23. Locations of crack arrest lines 1 through 4 are noted on the graph.

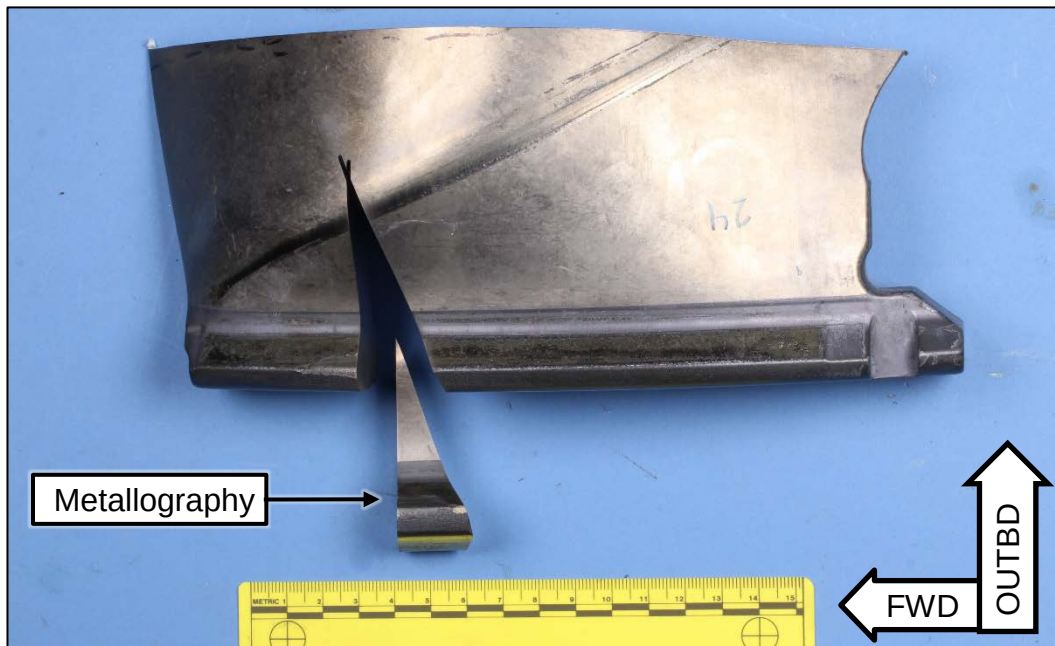


Figure 28. Overall view of the blade 24 piece sectioned for metallography on the face indicated.

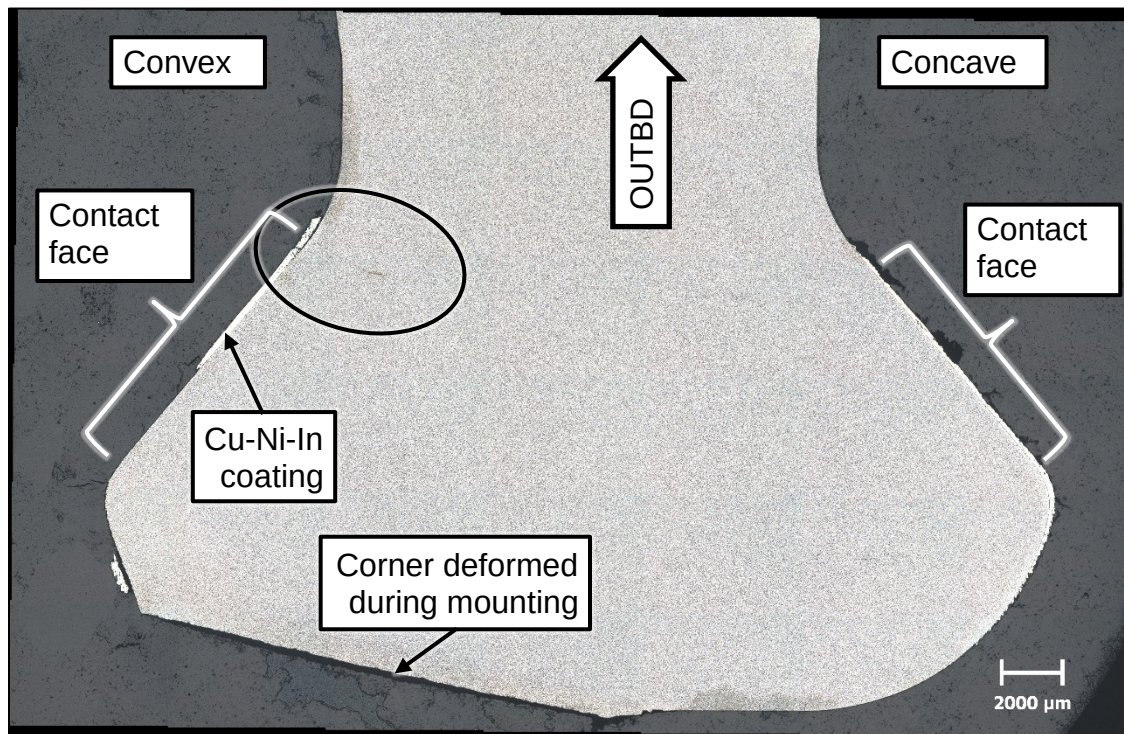


Figure 29. Macrograph showing the polished and etched transverse plane through the dovetail. Cracks were observed in the area circled on the convex side near the outboard end of the contact face. (Etchant: Kroll's reagent.)

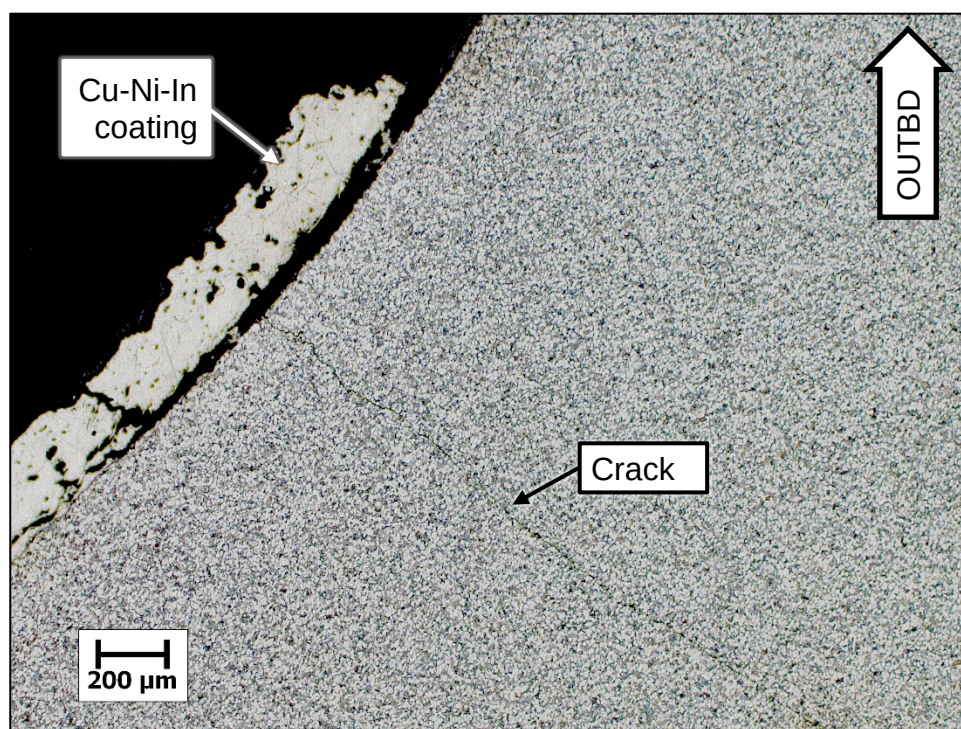


Figure 30. Micrograph of the area circled in figure 28 where the primary crack is indicated. (etchant: Kroll's reagent)

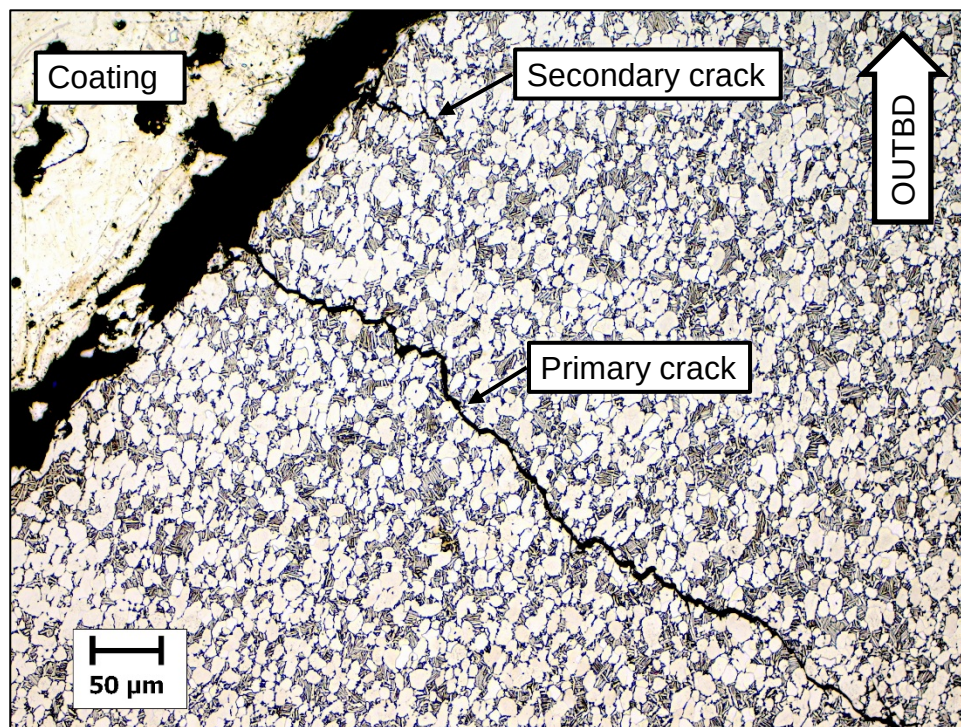


Figure 31. Micrograph of cracks emanating from under the Cu-Ni-In coating on the convex side of blade 24 showing the primary crack and a secondary crack. (Etchant: Kroll's reagent.)

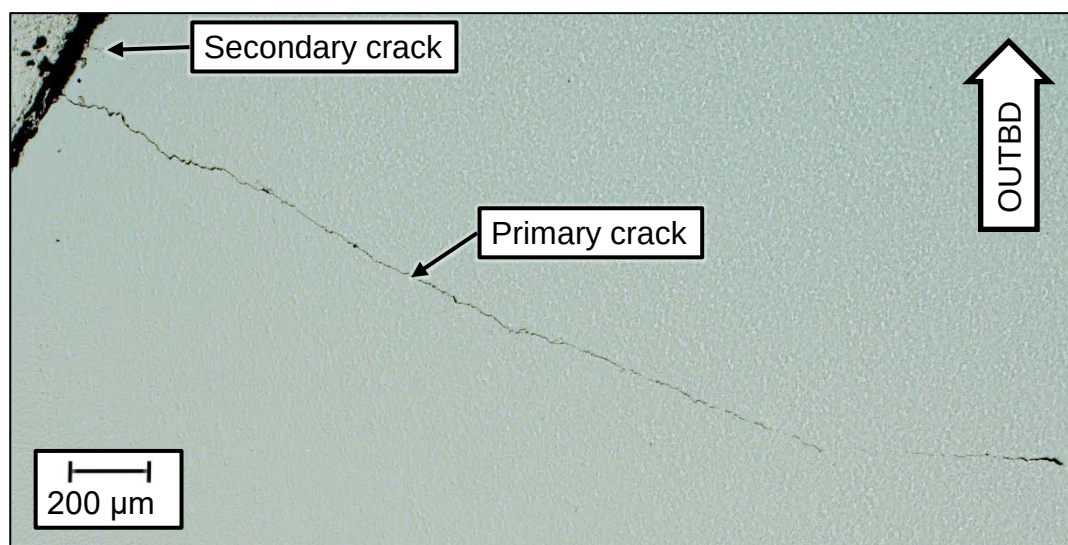


Figure 32. Overall image of the primary crack in the blade 24 cross-section after the surface had been etched with Kroll's reagent and was repolished.

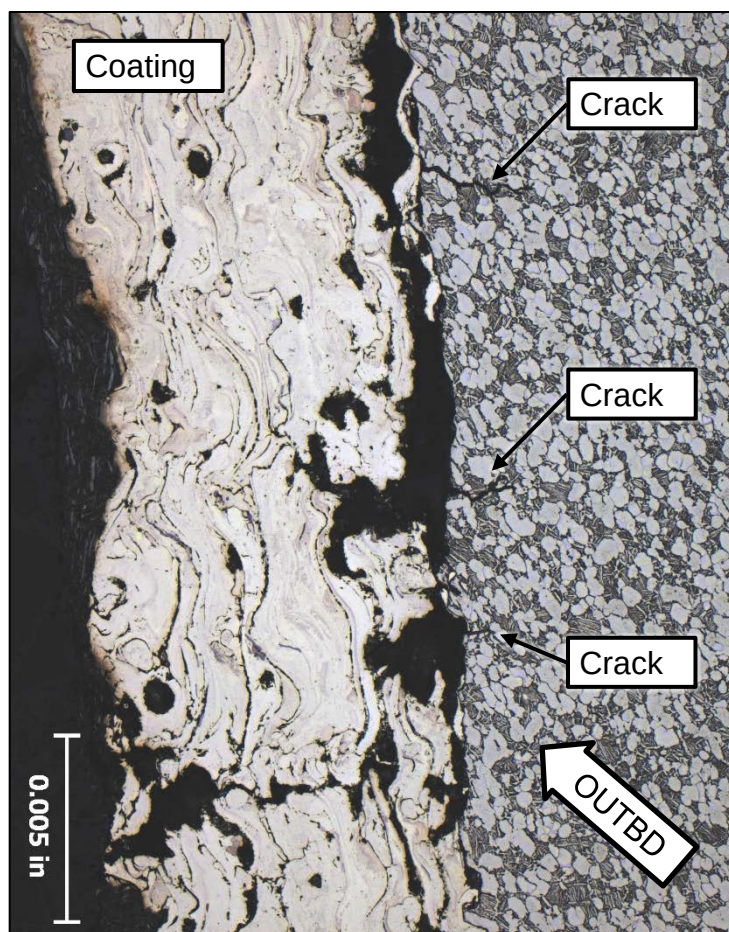


Figure 33. Secondary cracks emanating from the convex surface of blade 24 under the Cu-Ni-In coating inboard of the primary crack. (Etchant: Kroll's reagent.)

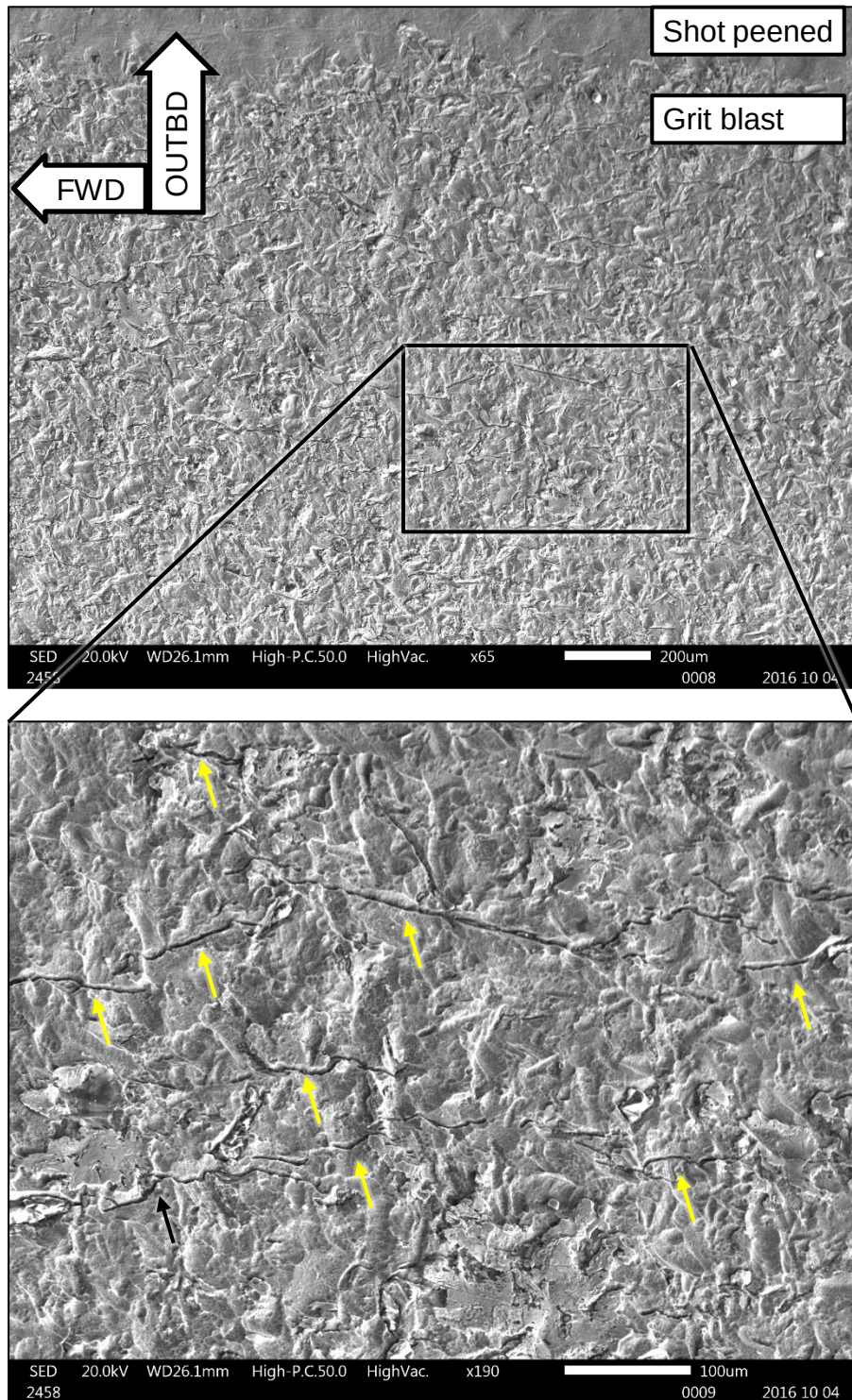


Figure 34. SEM images of a network of microcracks observed on the blade 2 surface viewed at lower magnification (upper image) and at higher magnification (lower image). Unlabeled arrows in the lower image indicate microcracks. (Images provided by GE.)

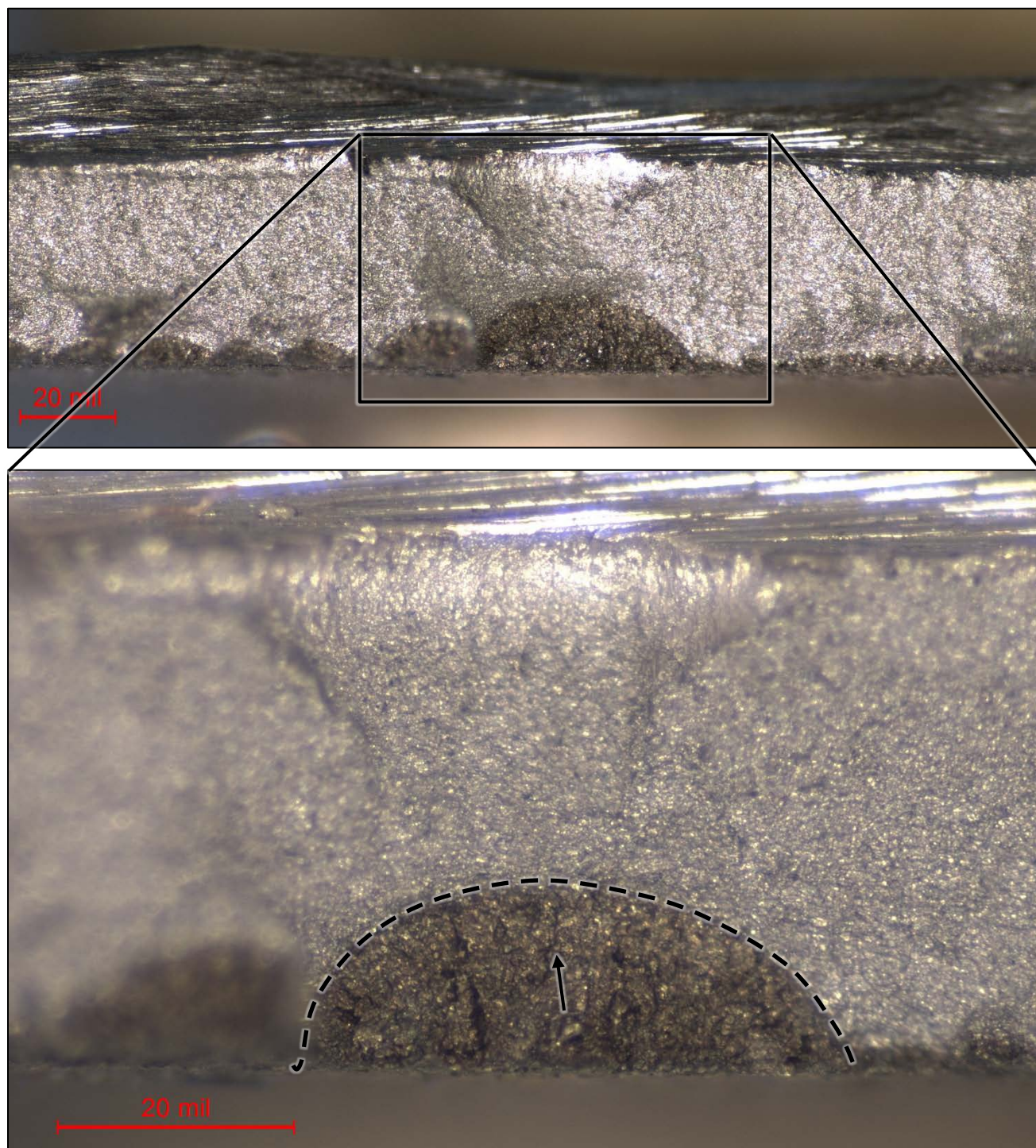


Figure 35. Optical images of the crack opened by lab fracture in blade 2. A dashed line in the lower image indicates the fatigue boundary of the primary crack feature, and an arrow indicates a crack arrest line. (Images provided by GE.)

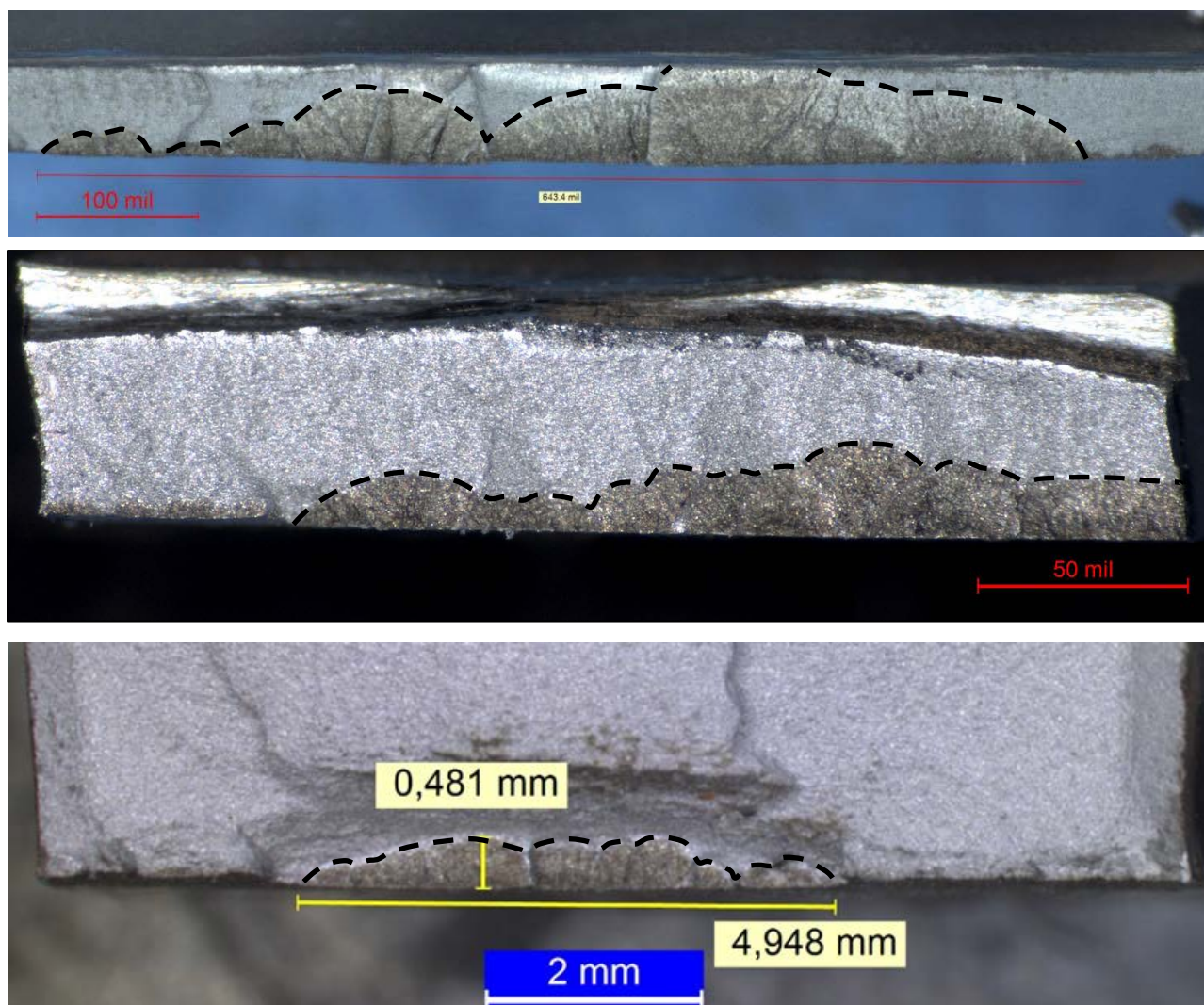


Figure 36. Optical images of cracks opened by lab fracture in blade 11 (upper image), blade 17 (middle image), and blade 19 (lower image). Dashed lines indicate the fatigue boundaries of the primary crack feature. (Upper and middle images provided by GE, and lower image provided by Safran.)

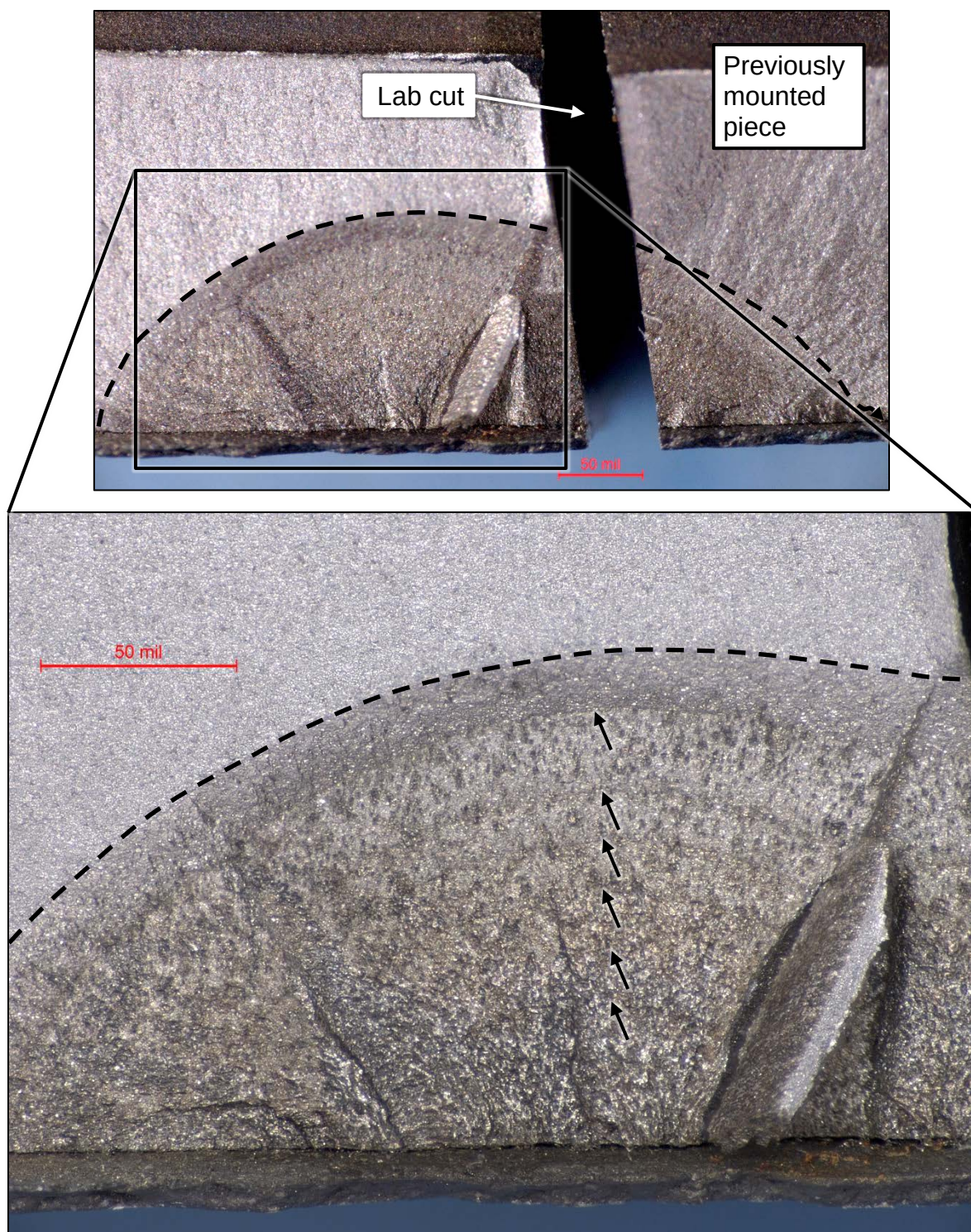


Figure 37. Optical images of the crack opened by lab fracture in blade 24. Dashed lines indicate the fatigue boundary of the primary crack feature, and arrows indicate crack arrest lines. (Images provided by GE.)

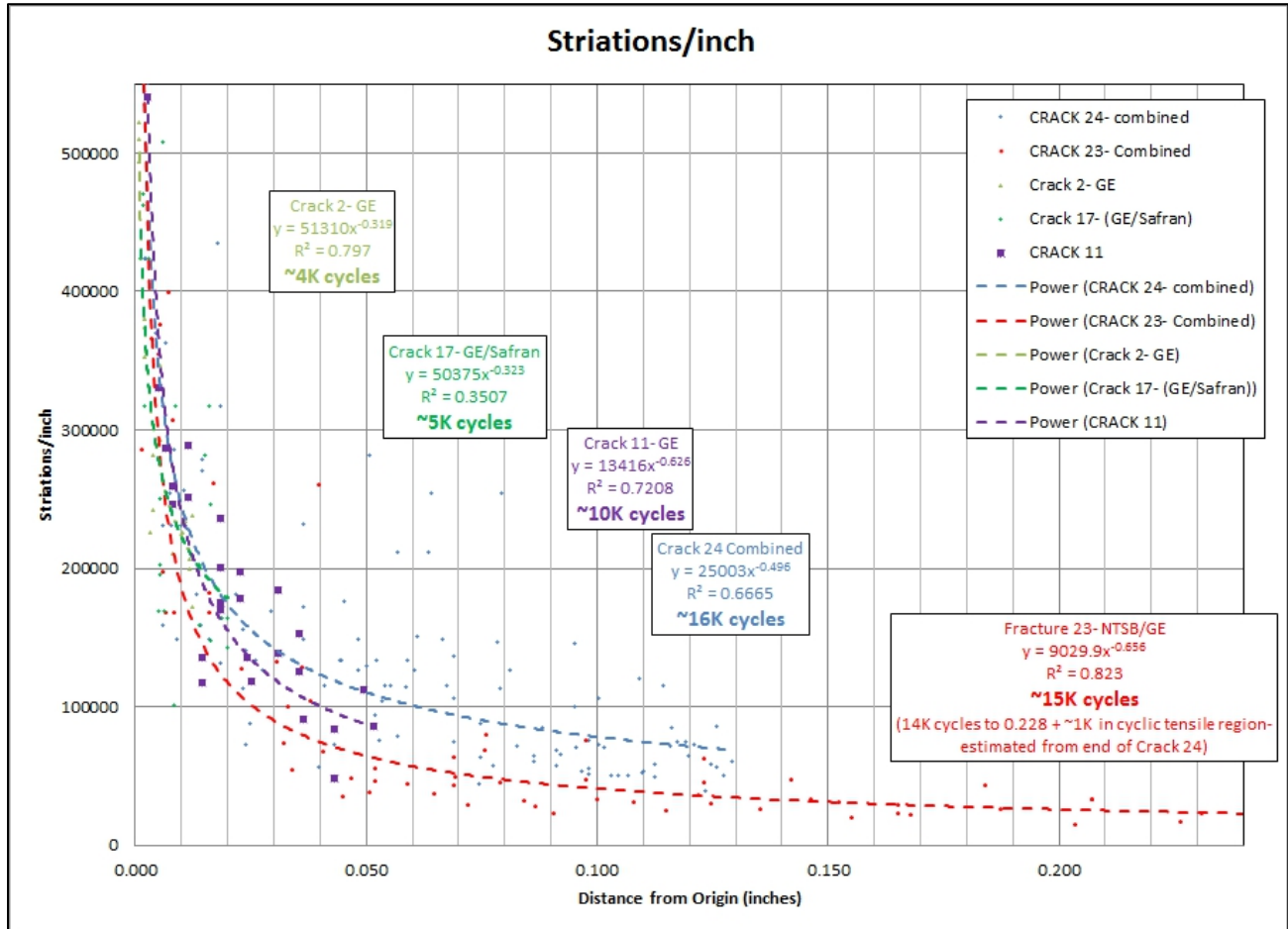


Figure 38. Striation density as a function of distance from the origin for fractured blade 23 and cracked blades 2, 11, 17, and 24. (Chart provided by GE.)

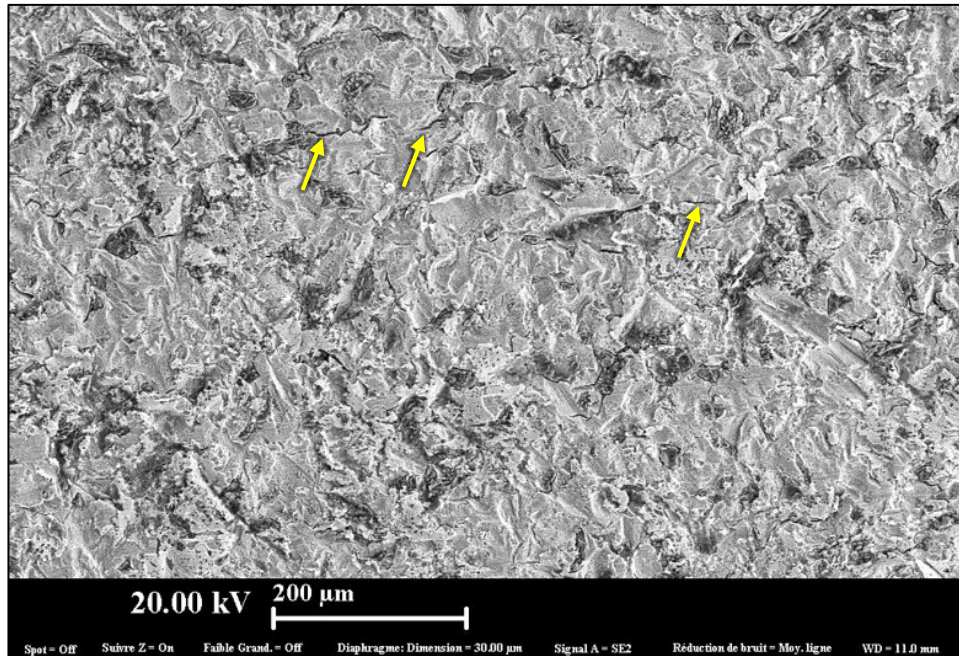


Figure 39. SEM image with unlabeled arrows indicating isolated microcracks on the surface of blade 5. (Image provided by Safran.)

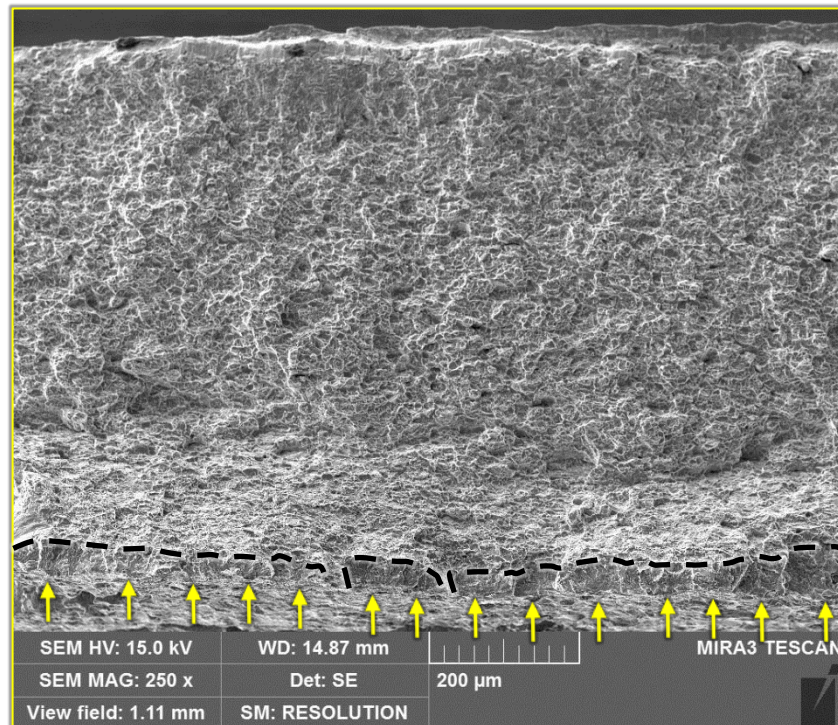


Figure 40. Network of microcracks in blade 21 opened by lab fracture. Dashed lines indicate microcrack boundaries, and arrows indicate origins at the blade surface. (SEM Image provided by GE.)

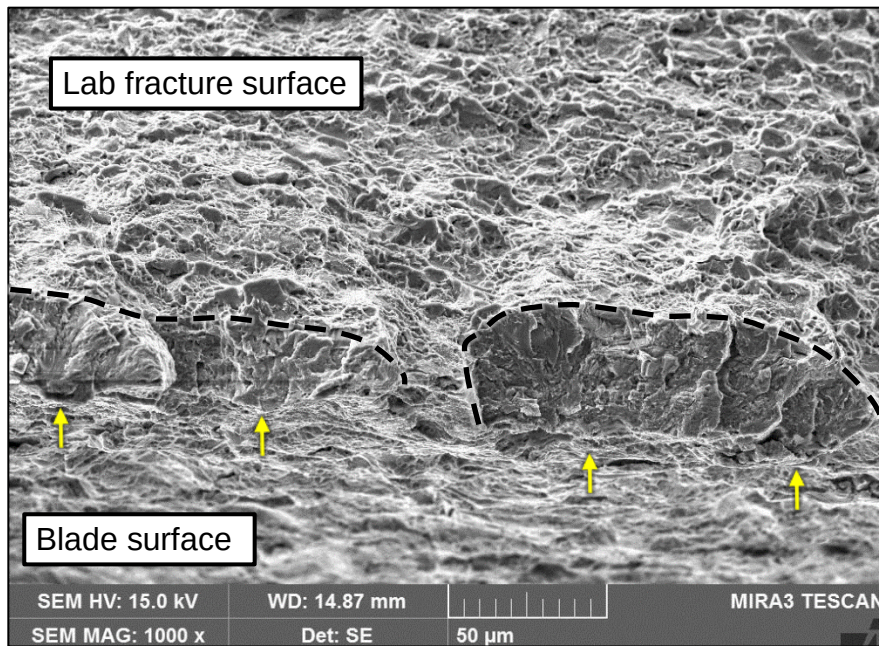


Figure 41. Closer view of microcracks in blade 21 opened by lab fracture. Dashed lines indicate microcrack boundaries, and arrows indicate origins at the blade surface. (SEM Image provided by GE.)

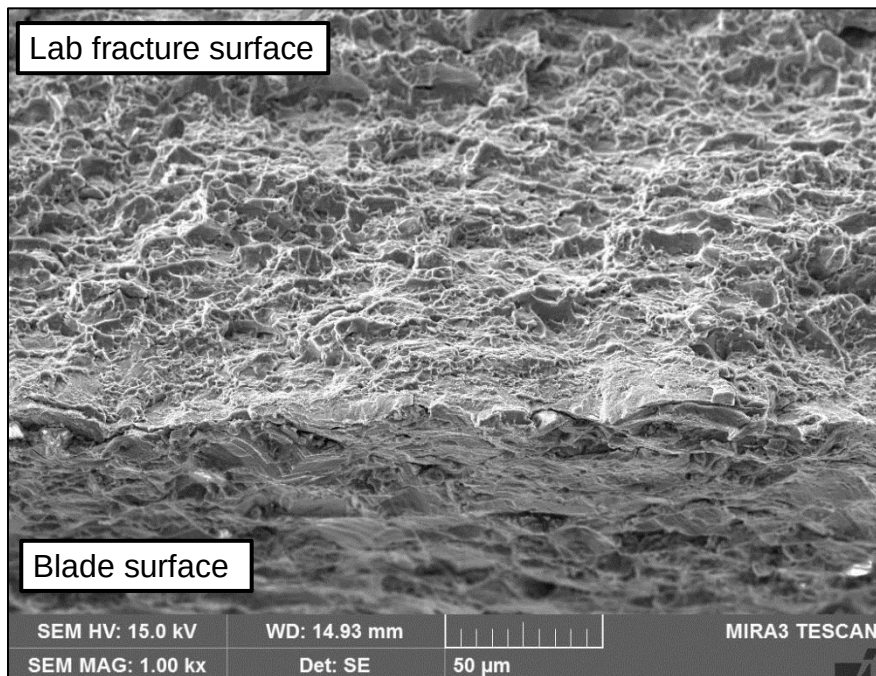


Figure 42. Lab fracture in blade 10 away from microcracks showing slant angles and dimpled fracture adjacent to the surface consistent with ductile overstress fracture. (SEM Image provided by GE.)

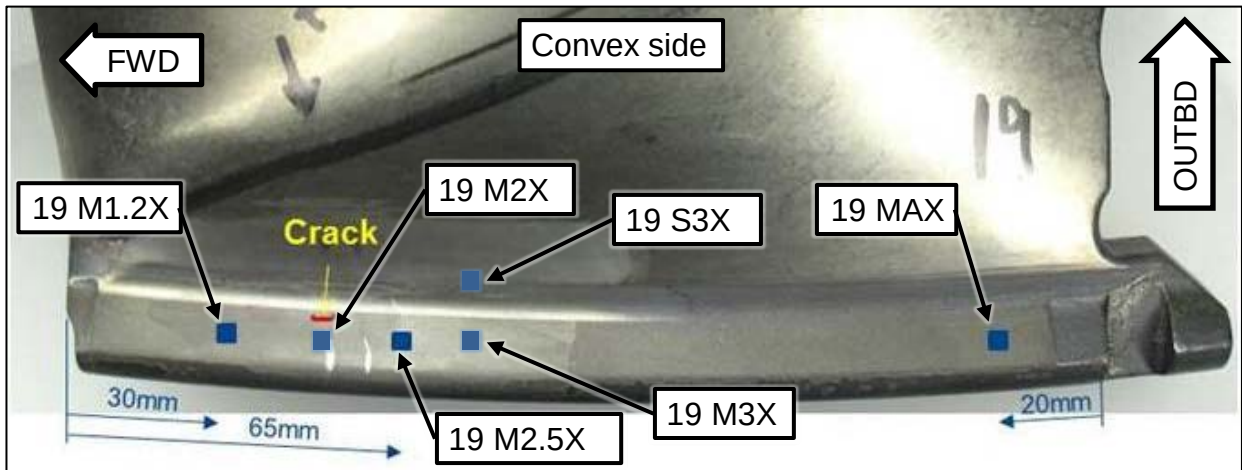


Figure 43. Approximate locations where residual stress profiles were acquired on blade 19 as indicated with blue boxes and labeled with the location reference code. (Image provided by Safran.)

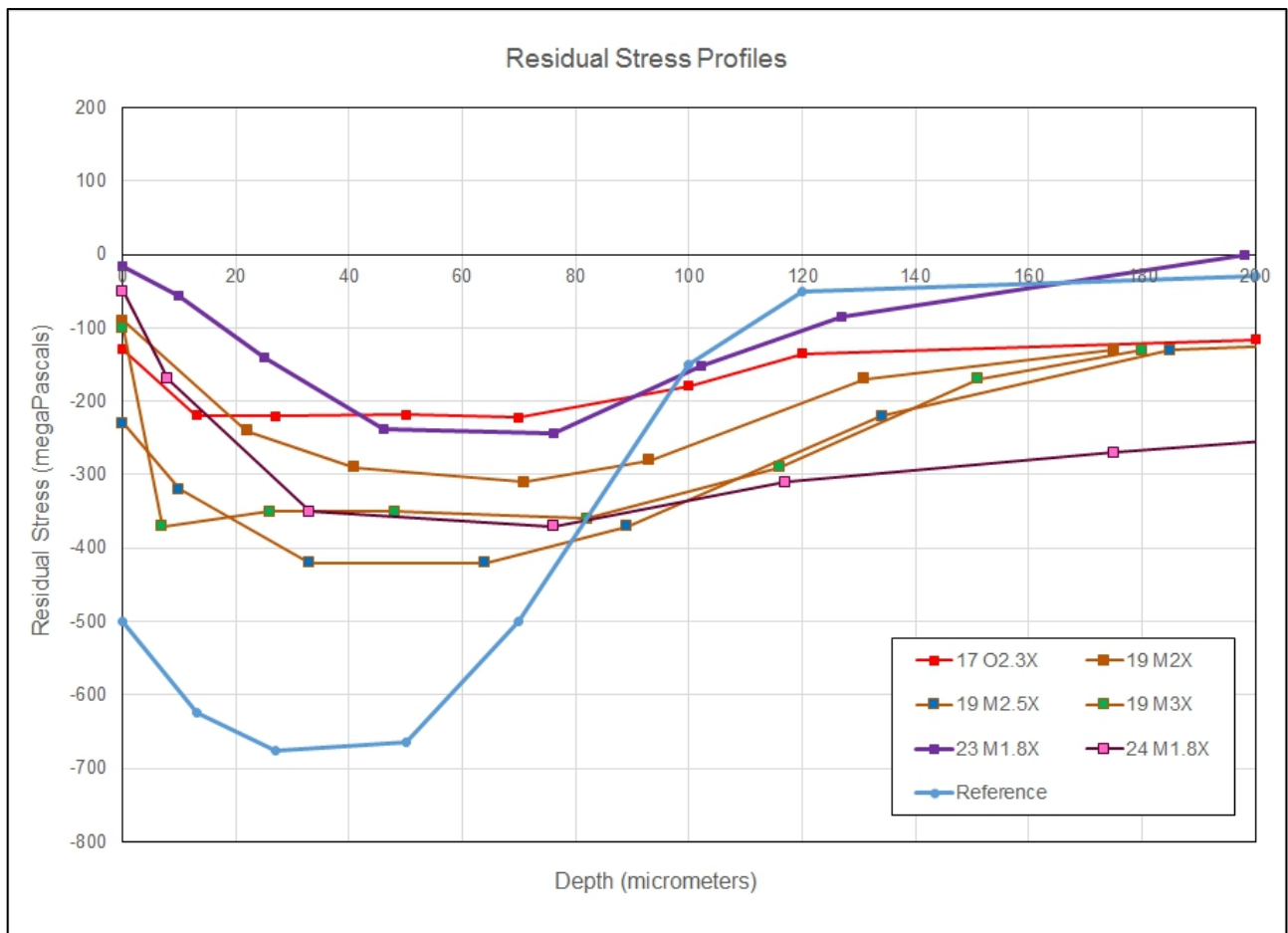


Figure 44. Residual stress profiles for cracked and fractured blades measured on the contact face near the area of cracking. The reference residual stress profile is also shown for comparison.

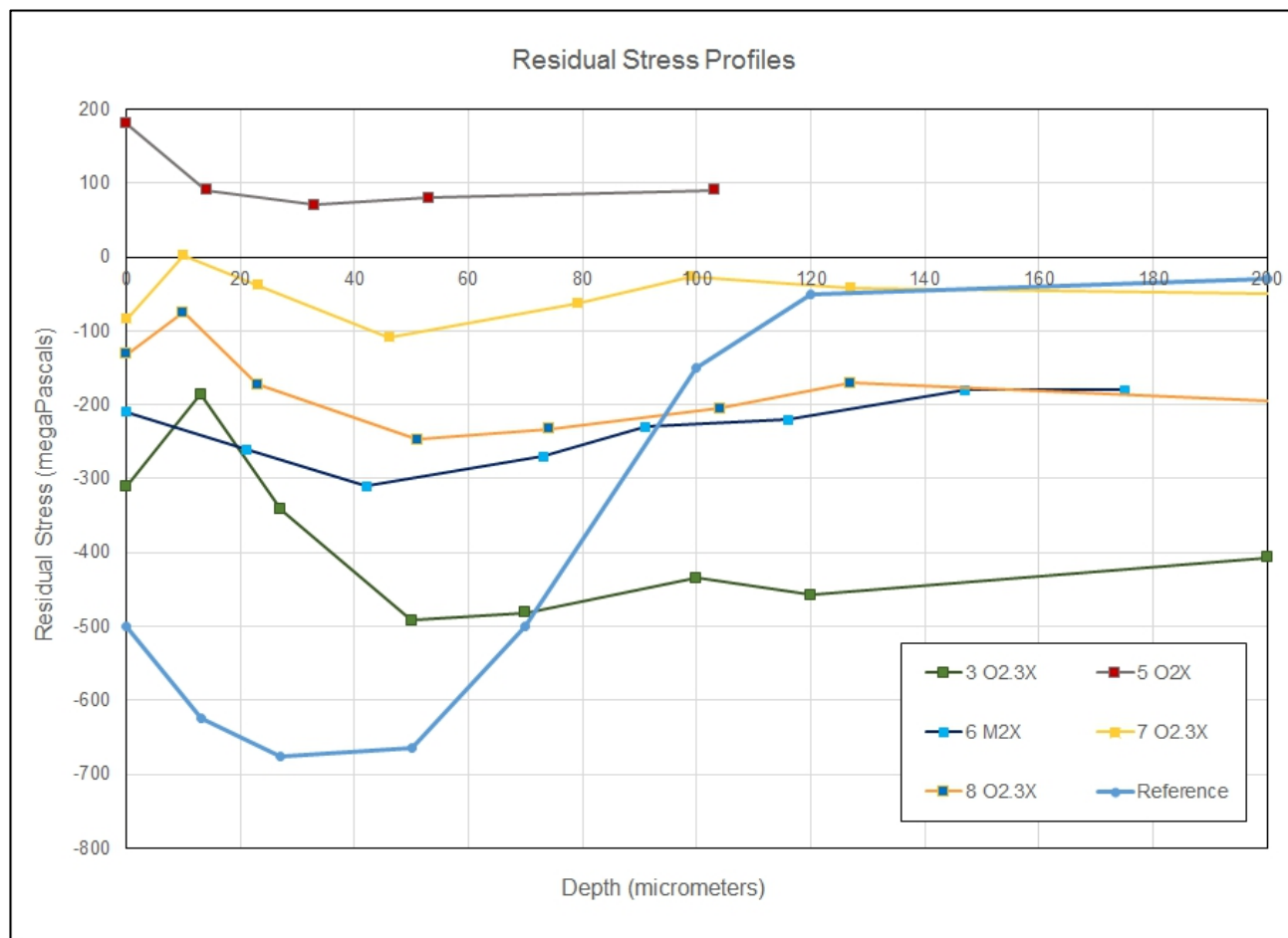


Figure 45. Residual stress profiles for blades without cracks. The profiles were acquired on the contact face in or near the area where cracks had been detected in other blades. The reference profile is also shown for comparison.

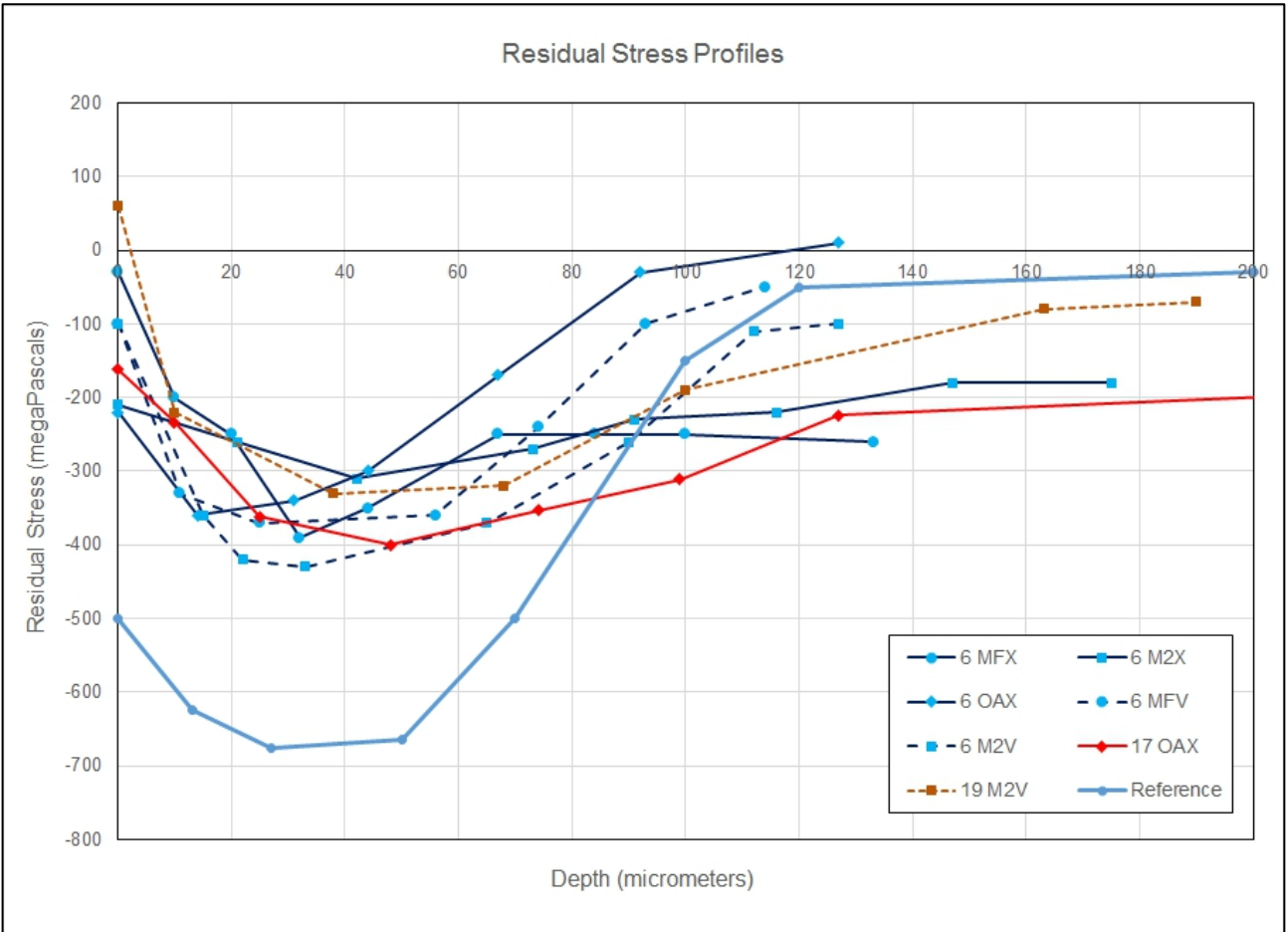


Figure 46. Abnormal residual stress profiles measured on the contact faces in areas away from cracks or microcracks. The reference profile is also shown for comparison.

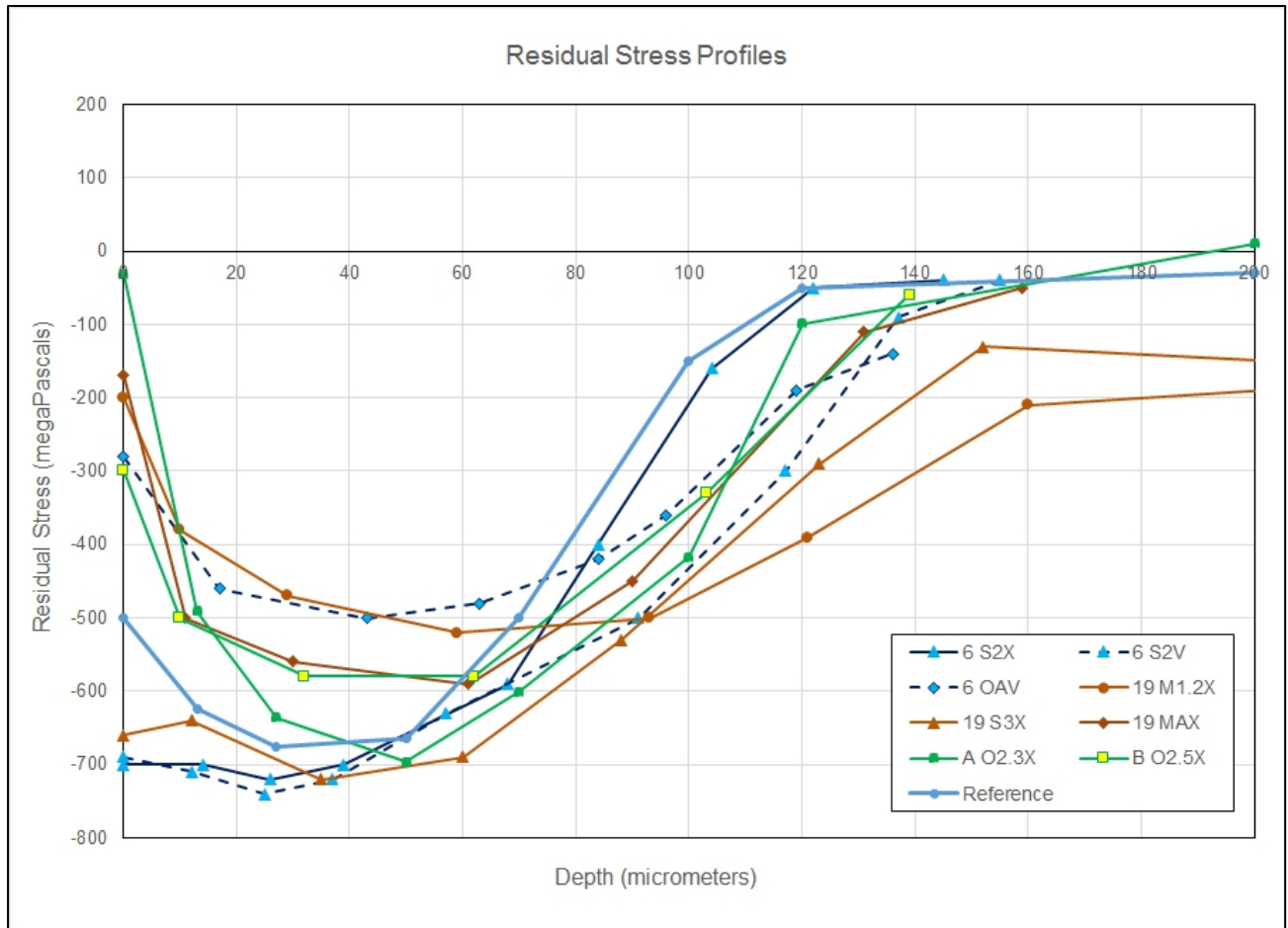


Figure 47. Residual stress profiles where the minimum residual stress value was less than or equal to -500 megapascals (rows highlighted in green in table 4).

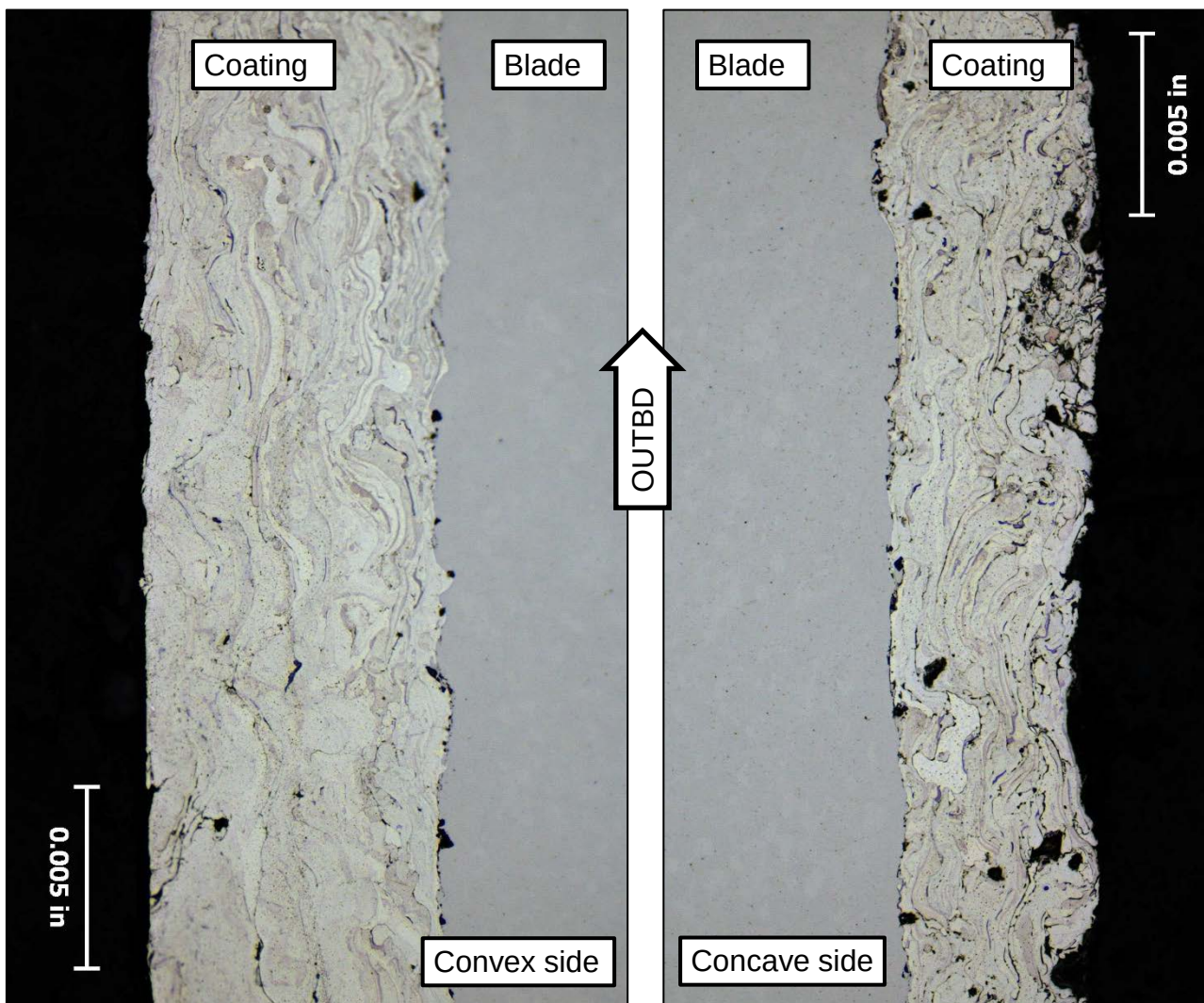


Figure 48. Views of the Cu-Ni-In coating on blade 24 as observed in the metallographic cross-section. The sample had been etched with Kroll's reagent and then repolished.

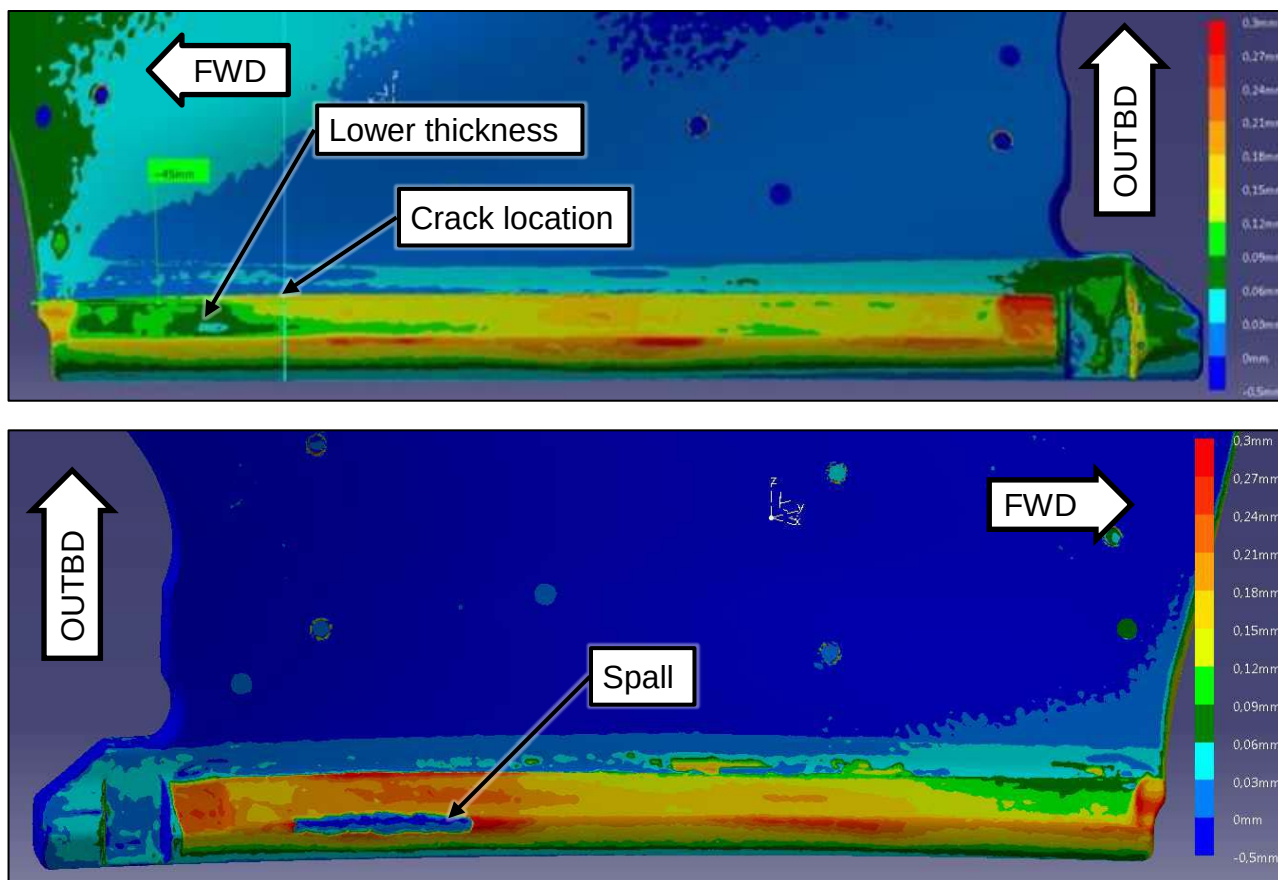


Figure 49. Coating thickness maps for the Cu-Ni-In coating on blade 1 showing the convex side (upper image) and the concave side (lower image). (Images provided by Safran.)

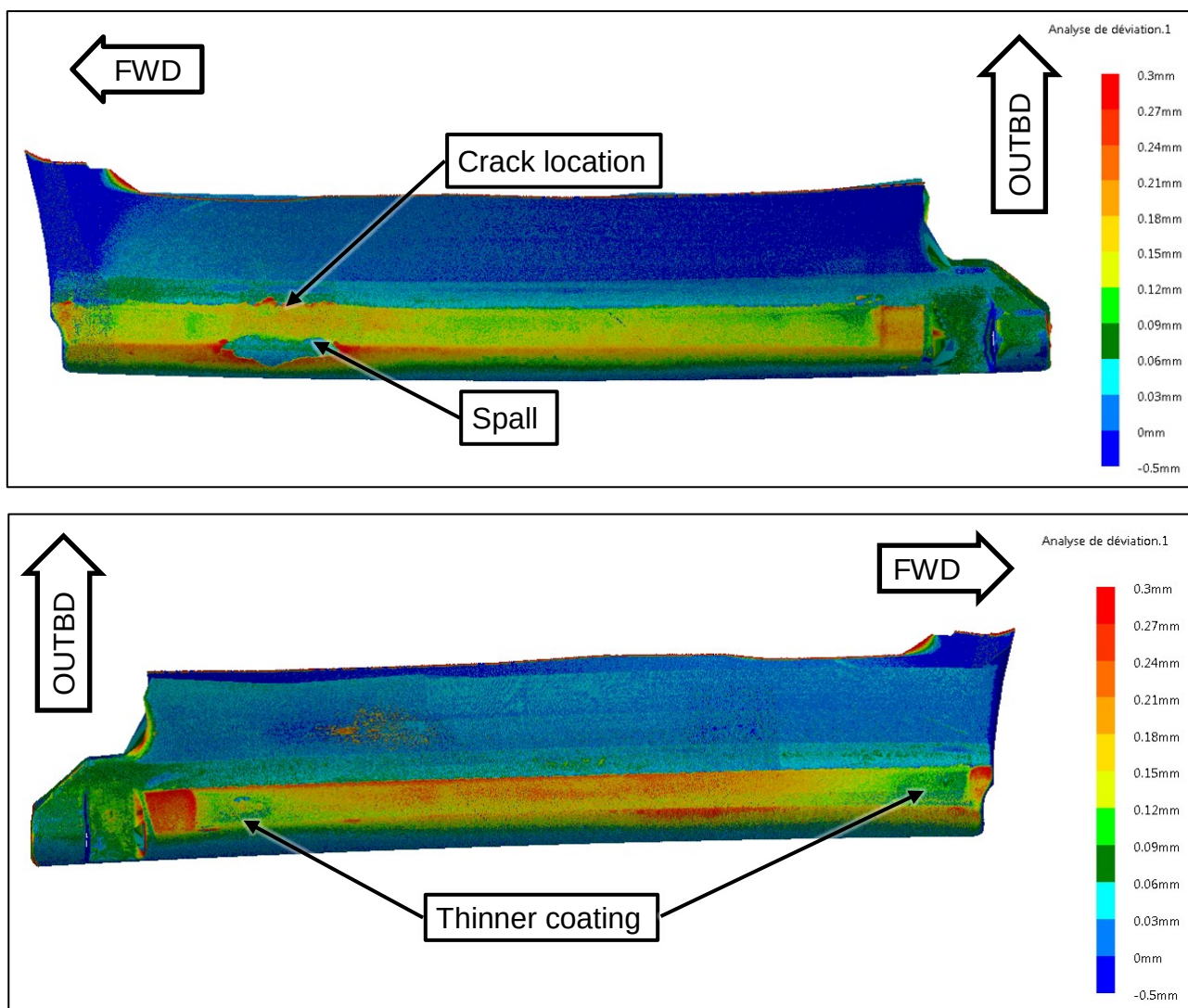


Figure 50. Thickness maps for the Cu-Ni-In coating on blade 11 showing the convex side (upper image) and the concave side (lower image). (Images provided by Safran.)

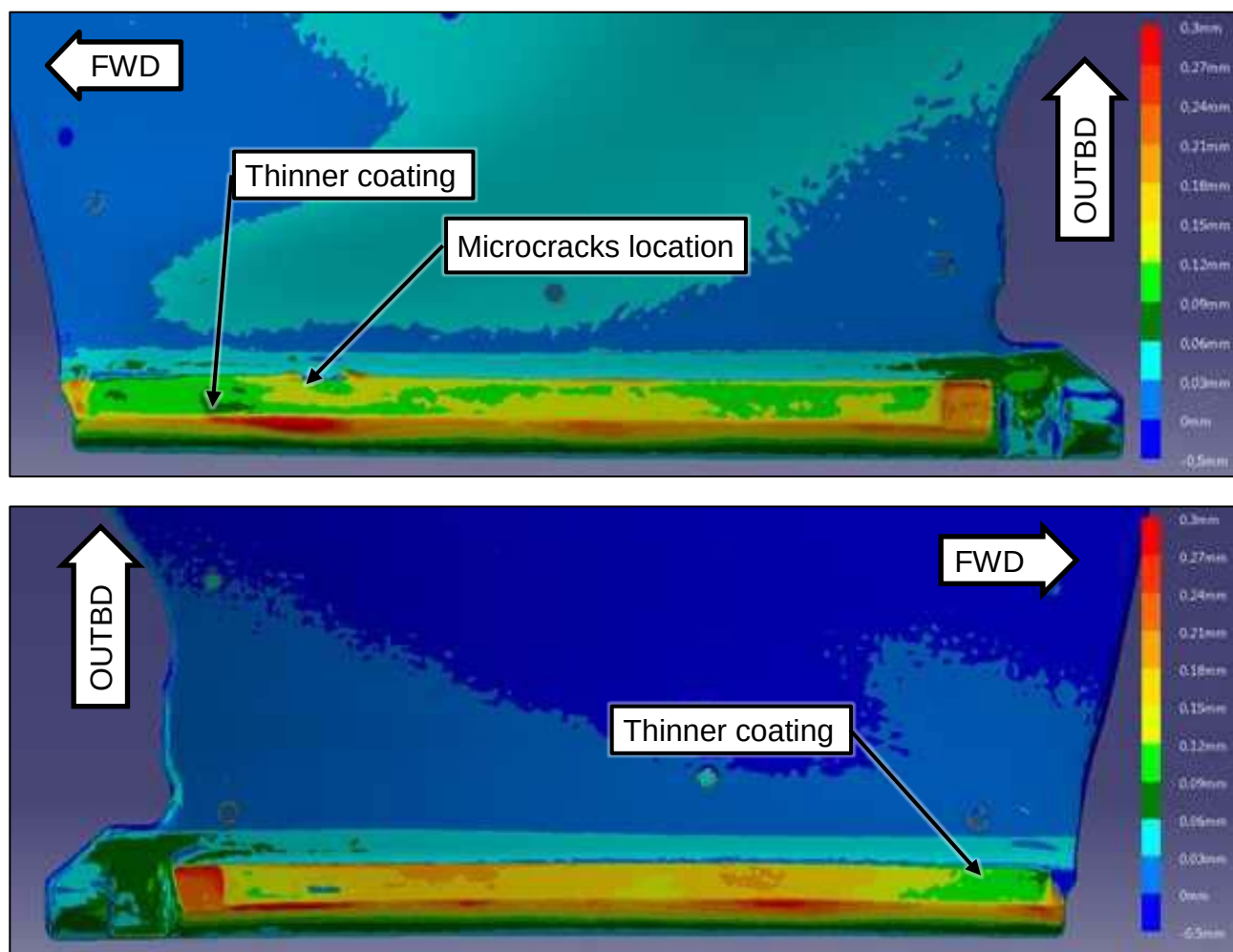


Figure 51. Thickness maps for the Cu-Ni-In coating on blade 6 showing the convex side (upper image) and the concave side (lower image). (Images provided by Safran.)

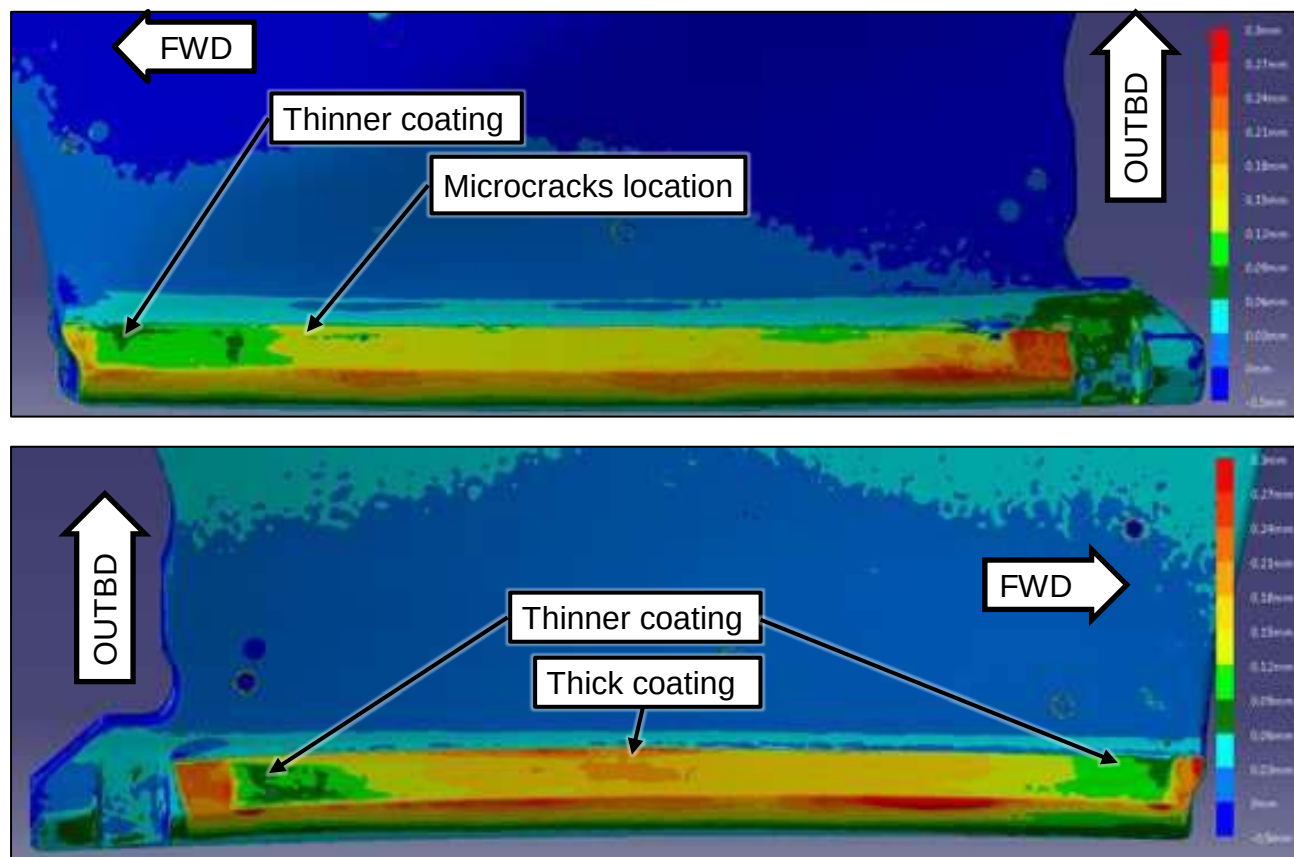


Figure 52. Thickness maps for the Cu-Ni-In coating on blade 12 showing the convex side (upper image) and the concave side (lower image). (Images provided by Safran.)

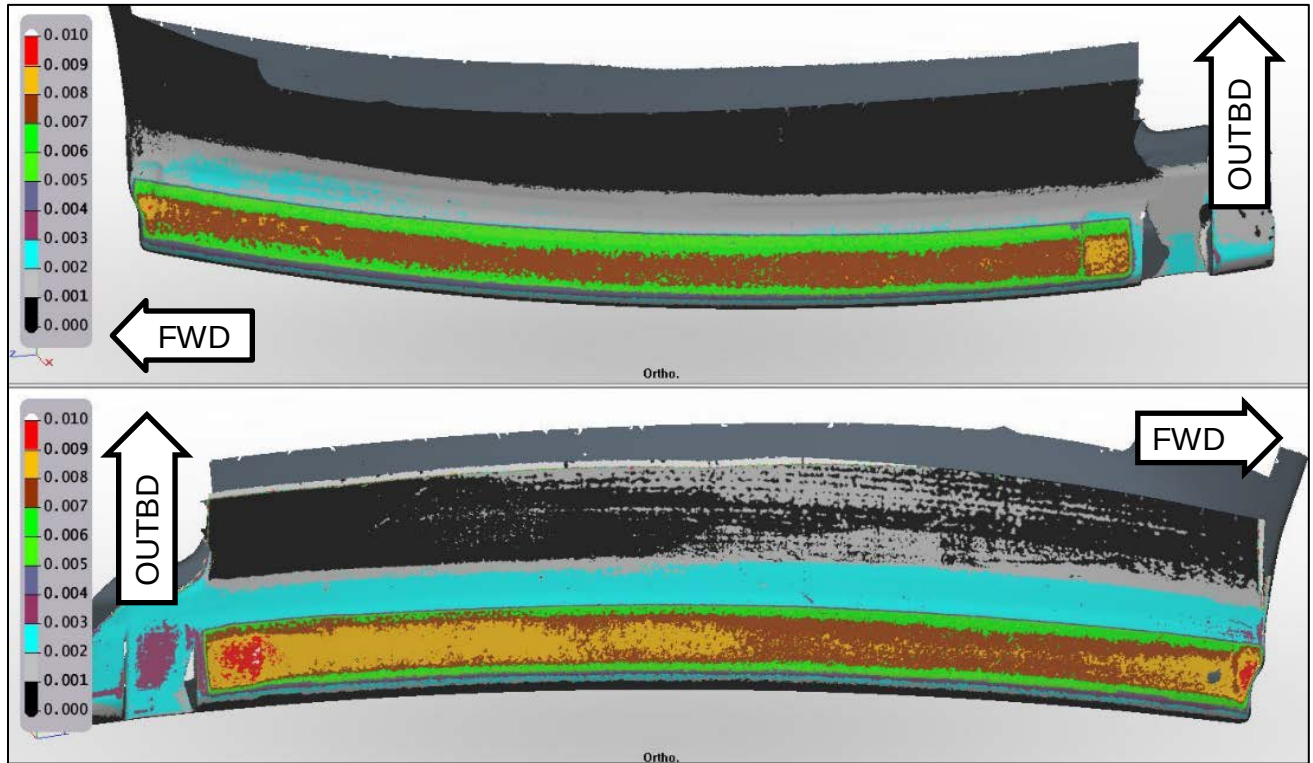


Figure 53. Thickness maps for the Cu-Ni-In coating on exemplar blade A showing the convex side (upper image) and the concave side (lower image). Coating thickness exceeding specification was observed on both sides along the length of the contact face. (Images provided by GE.)

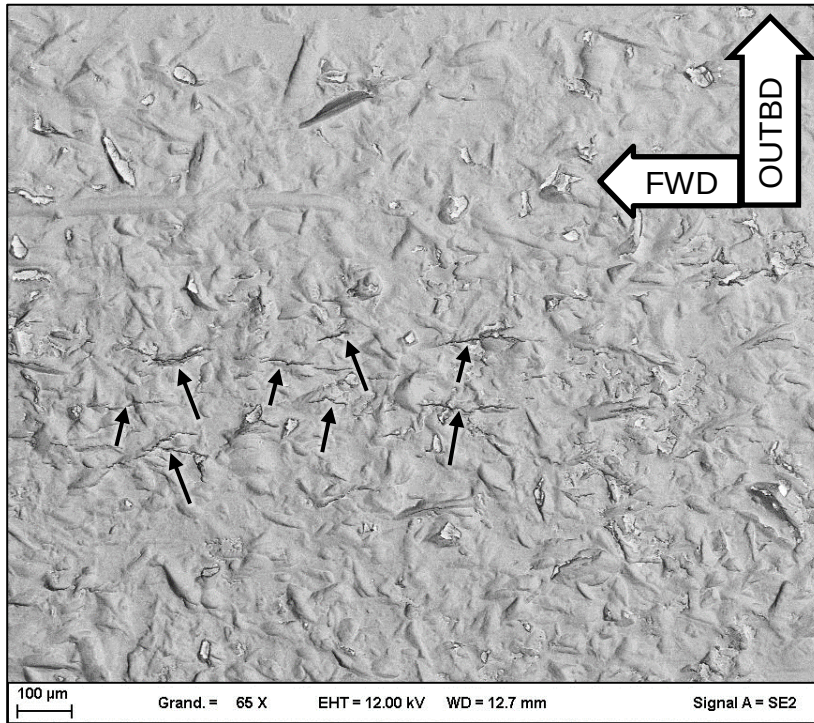


Figure 54. SEM image of the blade D surface after etching with modified Kroll's reagent where unlabeled arrows indicate microcracks. (Image provided by Safran.)

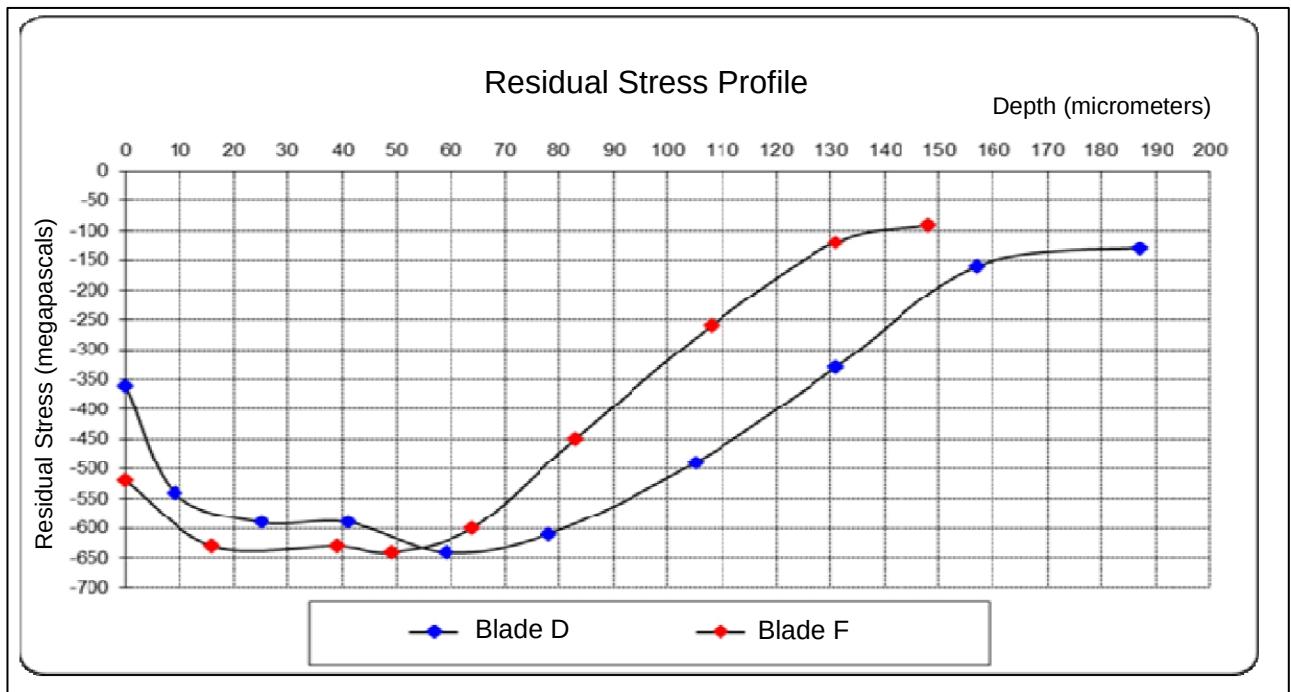


Figure 55. Residual stress profiles for blades D and F. (Chart provided by Safran.)

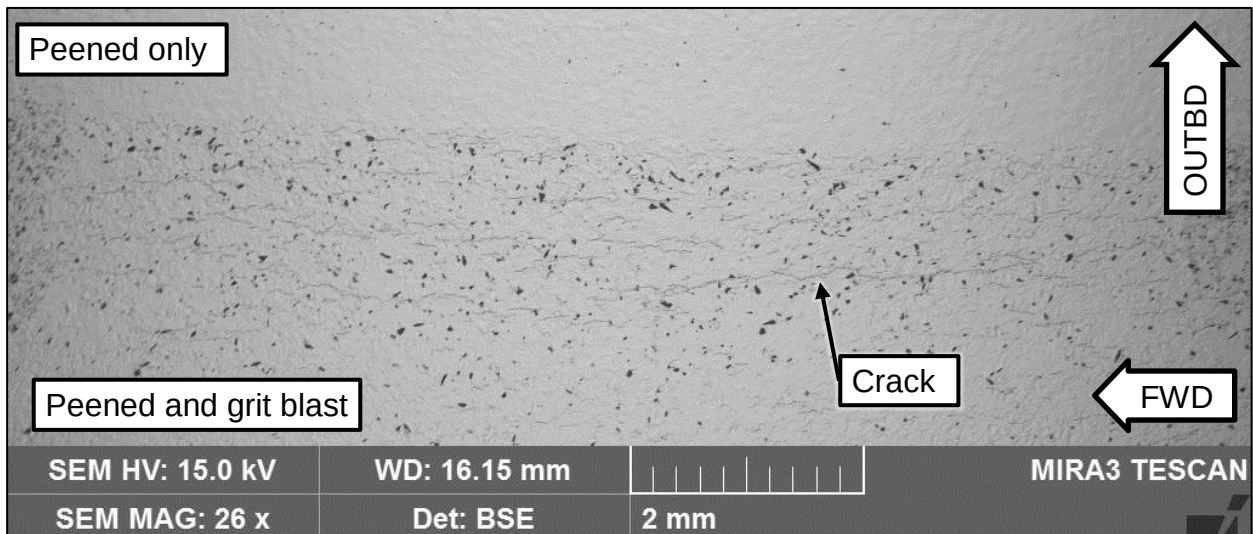


Figure 56. SEM image of the blade J surface after etching with modified Kroll's reagent showing a crack surrounded by microcracks. (Image provided by GE.)

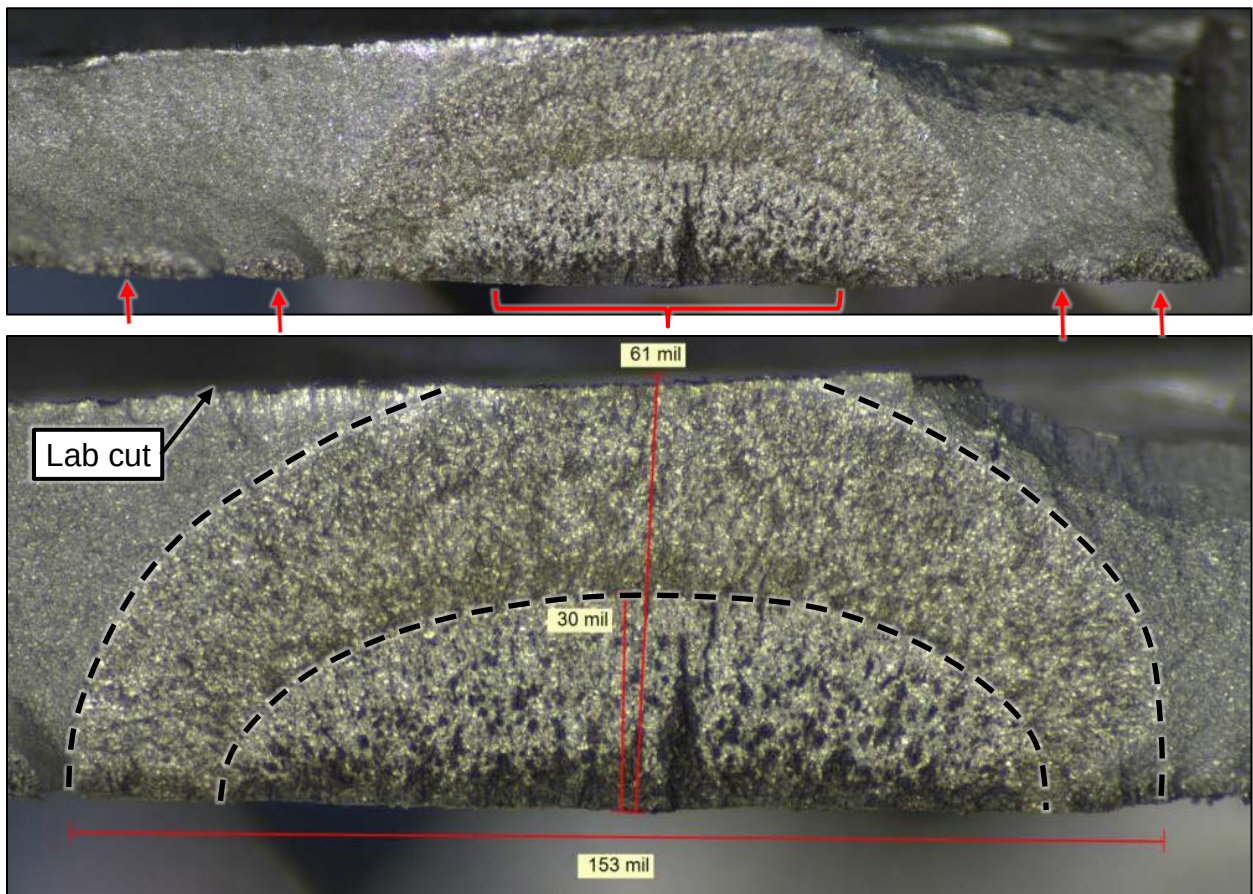


Figure 57. Optical image of the lab fracture (upper image) with a closer view of the fatigue region (lower image). Unlabeled arrows and a bracket in the upper image indicate approximate origin areas for the crack and adjacent microcracks. Dashed lines in the lower image indicate a crack arrest line and the fatigue boundary for the crack. (Images provided by GE.)

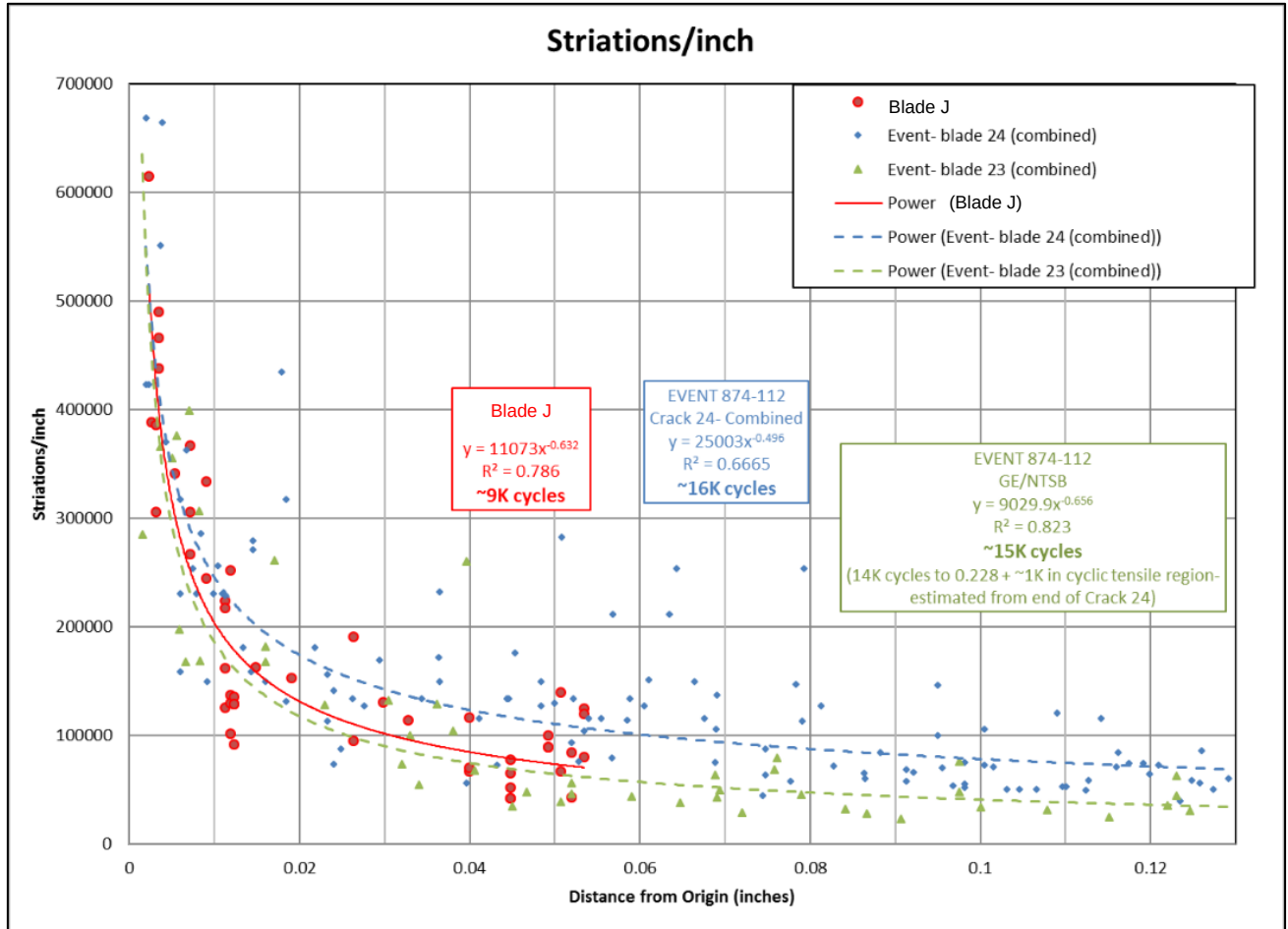
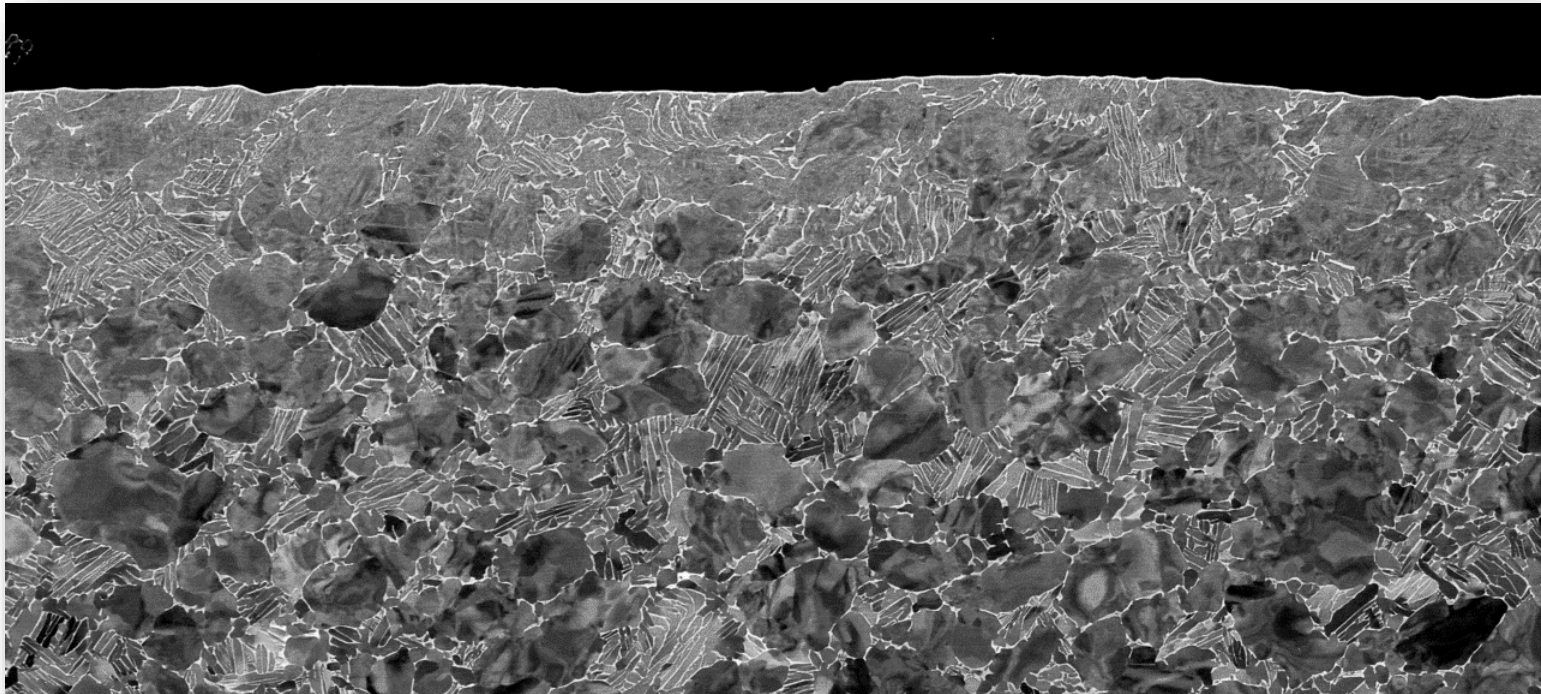


Figure 58. Striation density as a function of distance from the origin for cracked blade J. Data from accident blades 23 and 24 are also shown for reference. (Chart provided by GE.)

D. APPENDIX A: ELECTRON BACKSCATTER DIFFRACTION RESULTS

EBSD ANALYSIS

SWA Ti64 Fan Blade Event



Ian Spinelli/William Rossey

November 9, 2016

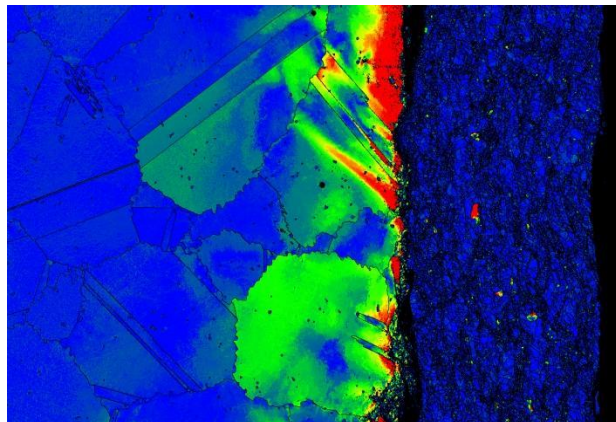
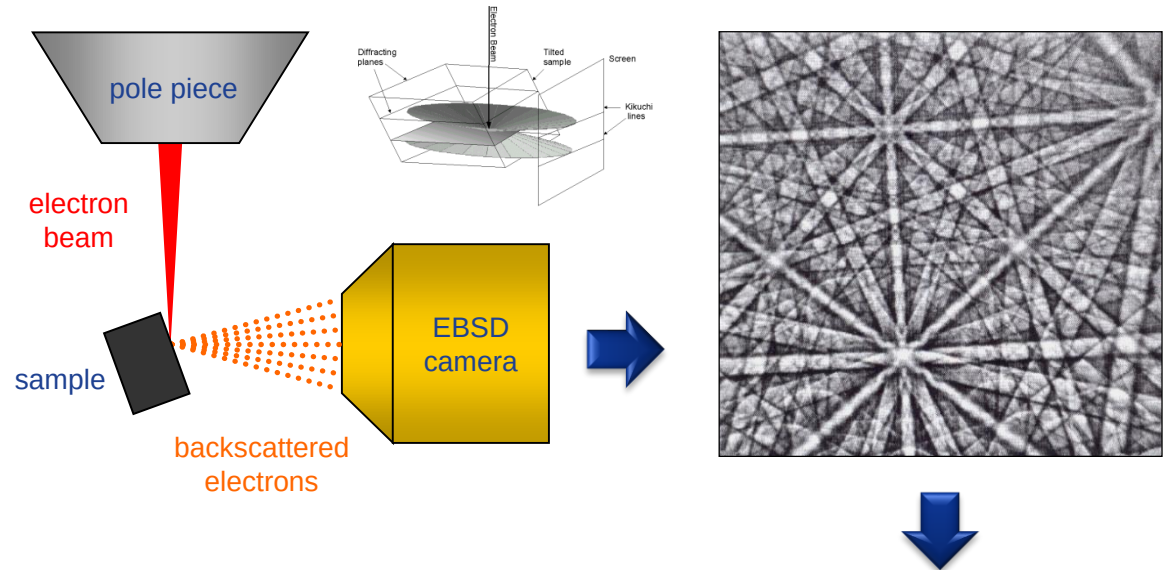
Electron Backscatter Diffraction

How Does It Work?

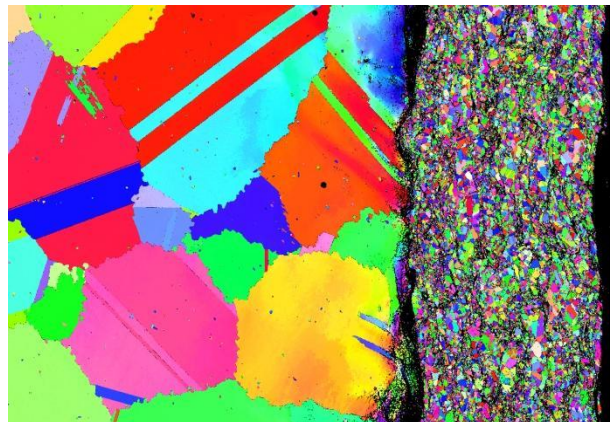
An SEM is used to accelerate electrons towards a grounded, conductive sample. These electrons penetrate the sample and are scattered within the material. Some of the electrons exit and collide with the CCD screen of an EBSD camera. Electrons scattered by the periodic crystalline lattice appear as pairs of parallel light and dark lines (Kikuchi Lines). Crystallographic information (orientation, retained strain, etc.) can be derived from the pattern of these lines.

EBSD at a Glance

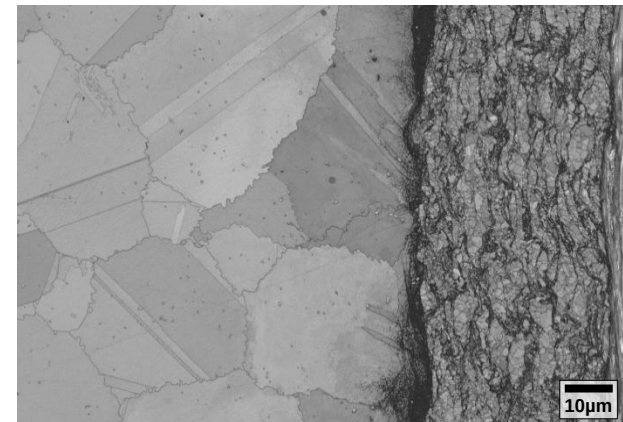
Generally, a FEG-SEM examining a Ni, Ti, or Fe alloy can achieve a spatial resolution of ~25nm in X, ~50nm in Y. The diffraction signal is generated in approximately the top 25nm of the sample, so it is necessary to have a flat surface with a high quality prep (i.e., vibratory polishing).



Grain Misorientation – color at every point indicates level of misorientation from its parent grain orientation. Grains with colors closer to the red end of the scale have more retained work



Inverse Pole Figure - colors in IPF maps identify the crystallographic orientation (e.g. red is 001) parallel with the specified sample axis



Band Contrast - indicates the contrast/quality of the diffraction pattern observed at every point of the map (black=worst, white=best).

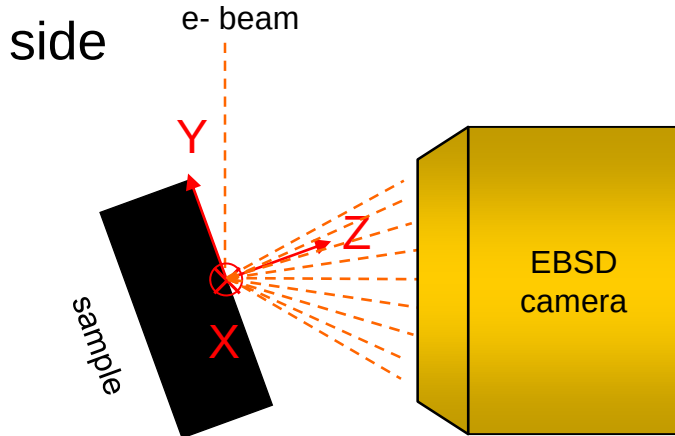
Appendix C1: Description of EBSD analysis.

Summary

Blades 3 & 24

Mapping

- JEOL 6610 SEM
- 20kV accelerating voltage
- Step size = $0.5\mu\text{m}$ at MAG = 200X
- 1269 points in X and 951 points in Y
- Only mapped alpha (hexagonal)

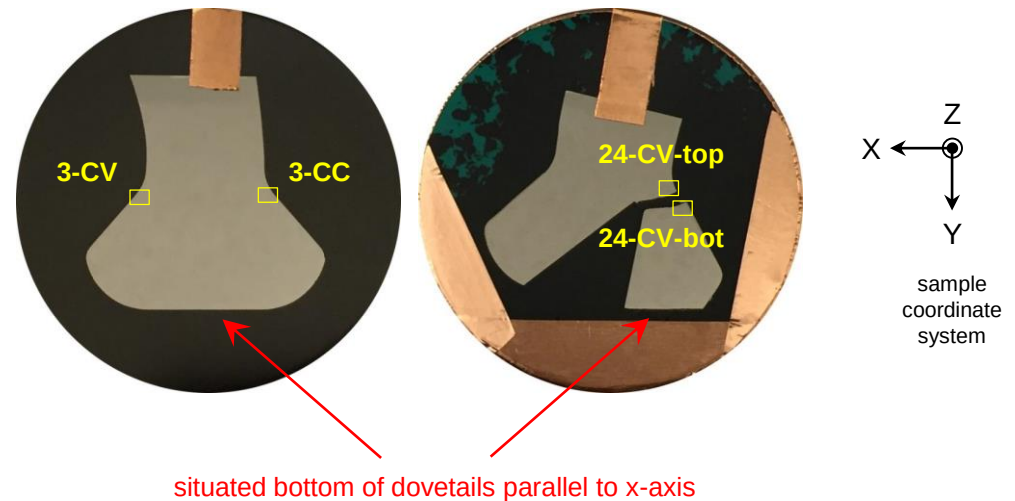


Observations

- Mapping hit rates were poor:
 - Pt-coated Blade 24, which attenuates signal
 - Ignored beta
 - Coarse step size misses finer alpha laths
- Some level of texture in both areas for both blades
- BSE images indicate Blade 3 may have a better peen than 24. Need higher-res EBSD to evaluate further.

	Hit Rate
Blade 3 CV	60%
Blade 3 CC	48%
Blade 24 CV top	35%
Blade 24 CV bot	39%

top



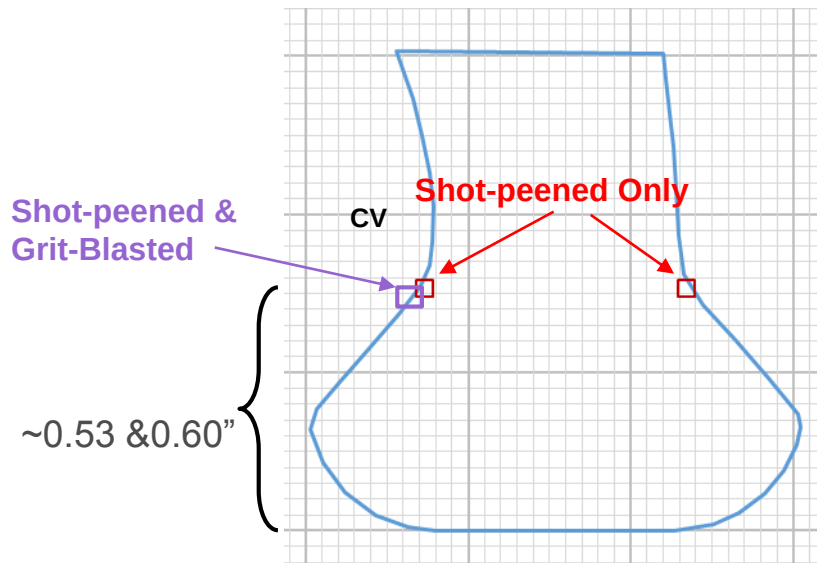
Appendix C2: Description of the set-up for examining the metallographic mounts of blades 3 and 24.

BSE/EBSD Locations

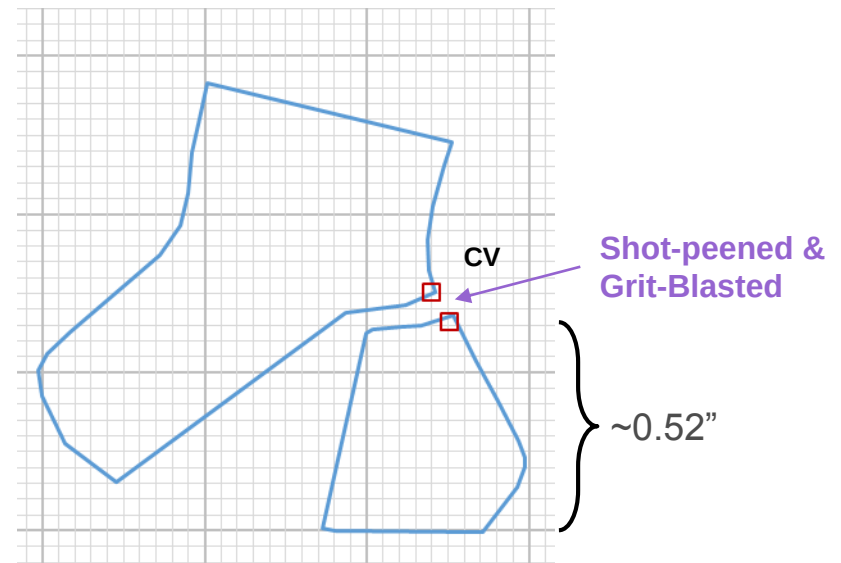
plotted from stage coordinates

* grid squares are 1x1mm

Blade 3



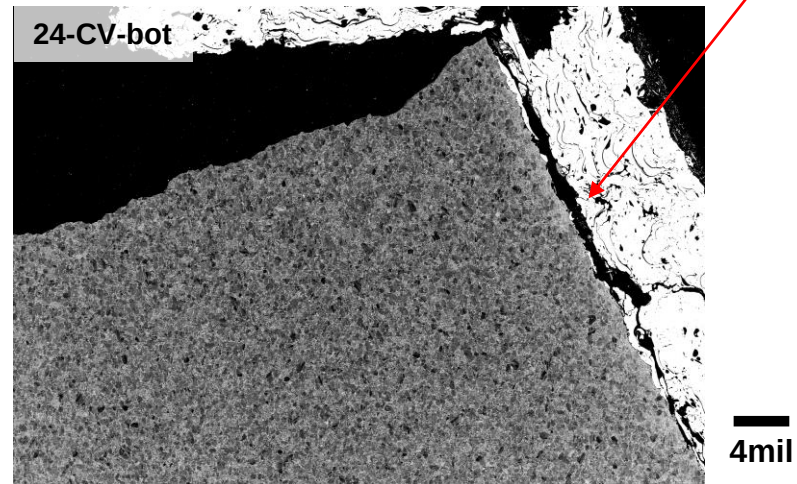
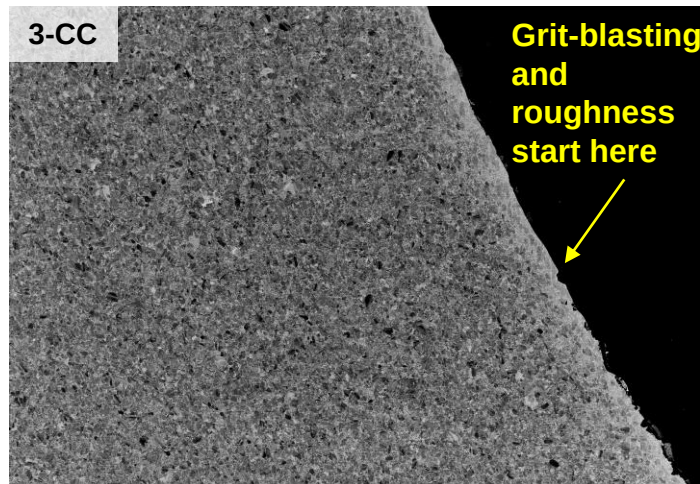
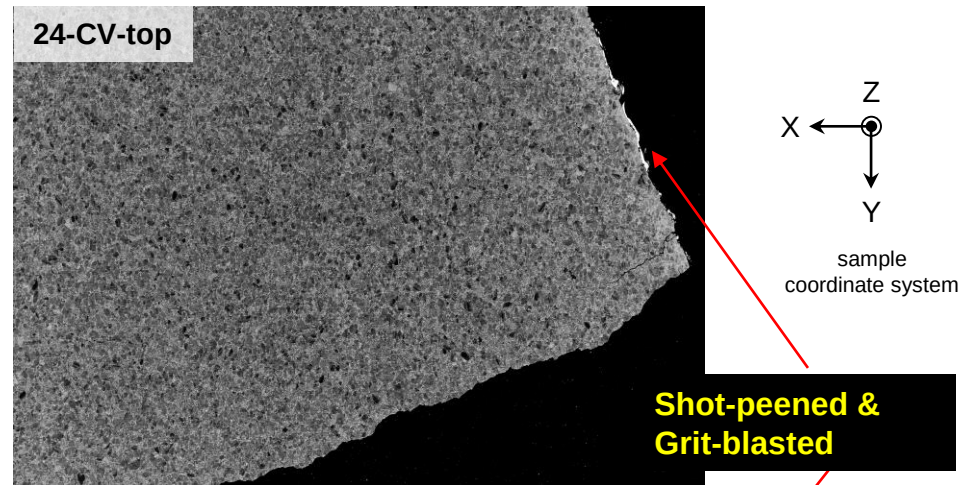
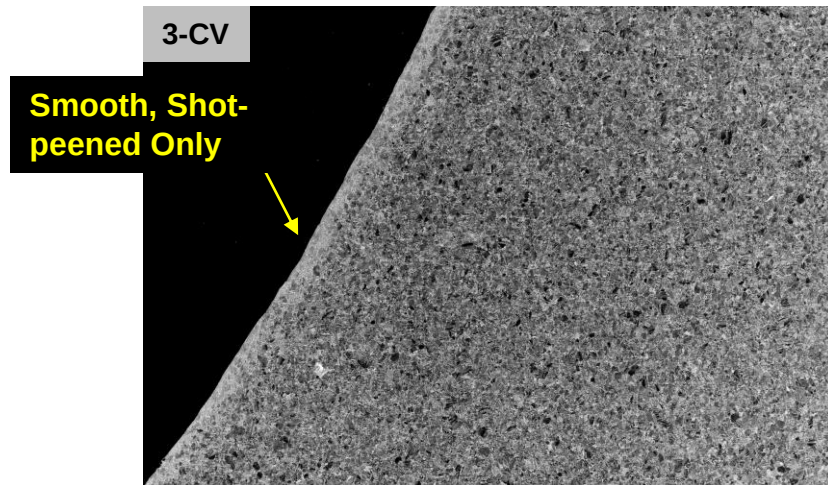
Blade 24



Appendix C3: Description of the set-up for examining the metallographic mounts of blades 3 and 24.

Backscattered Electron Images

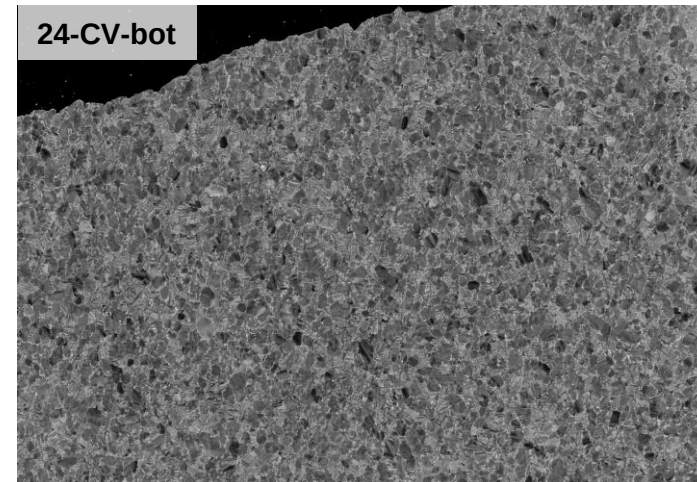
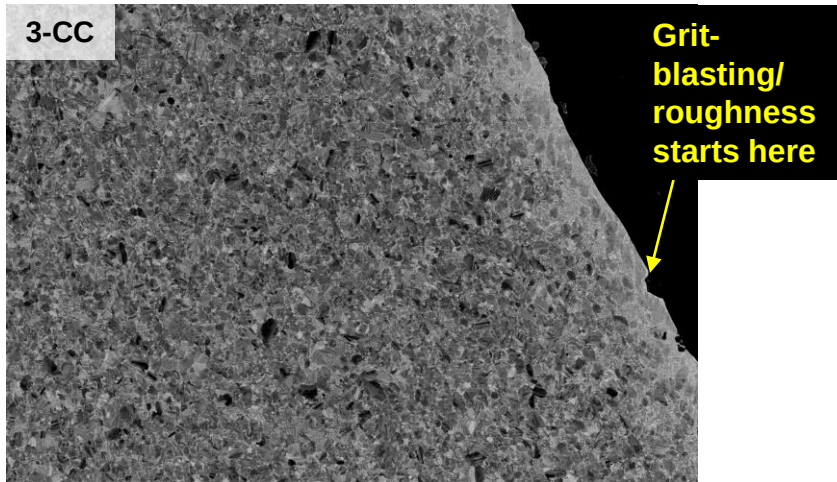
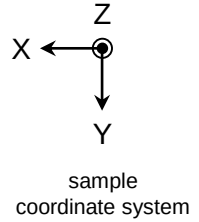
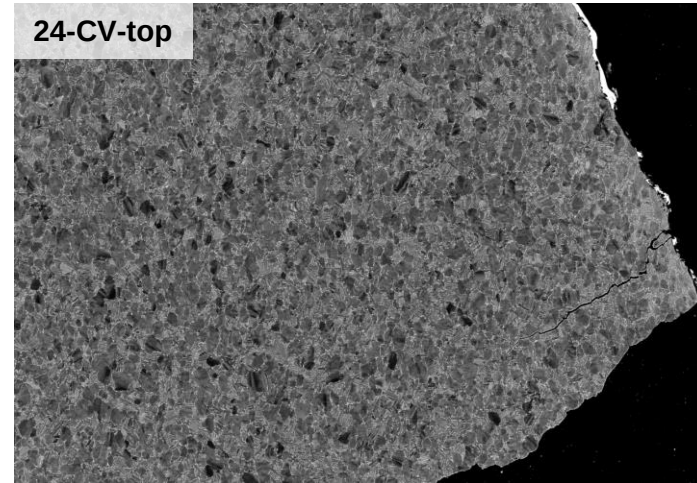
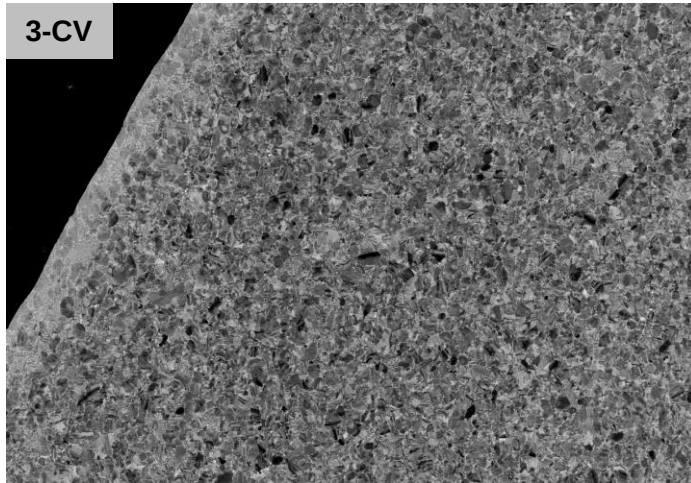
100X



Appendix C4: SEM backscatter images showing the changes in the near-surface response in the different areas of the two blades.

Backscattered Electron Images

200X

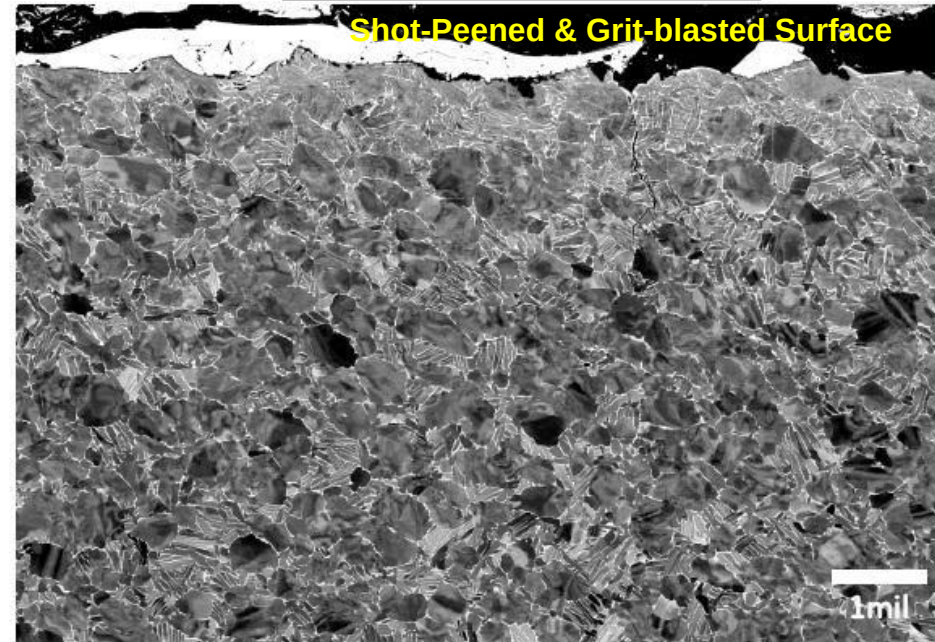
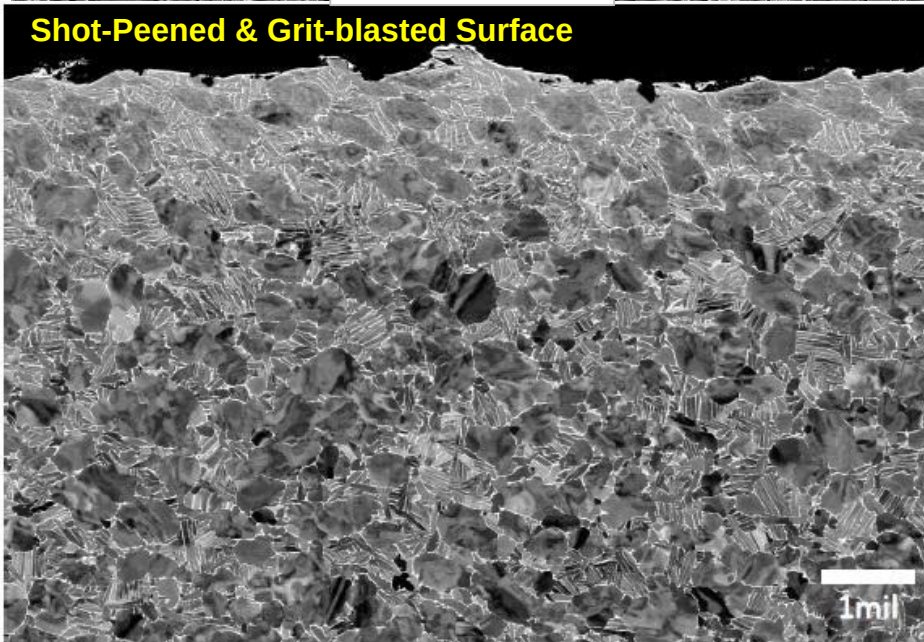
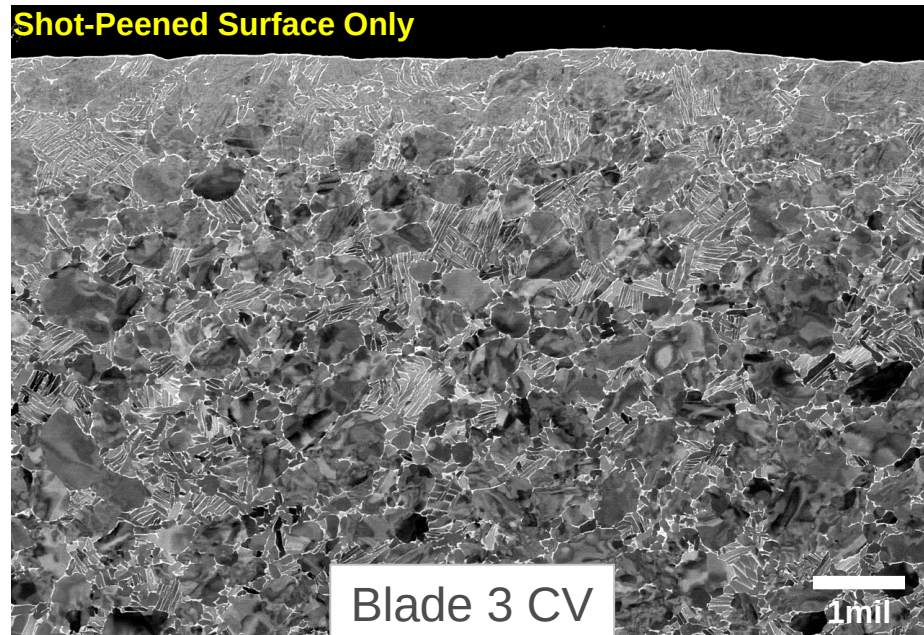


2mil

Appendix C5: Higher magnification views of the same regions.

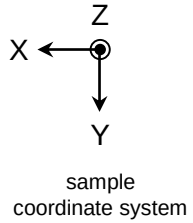
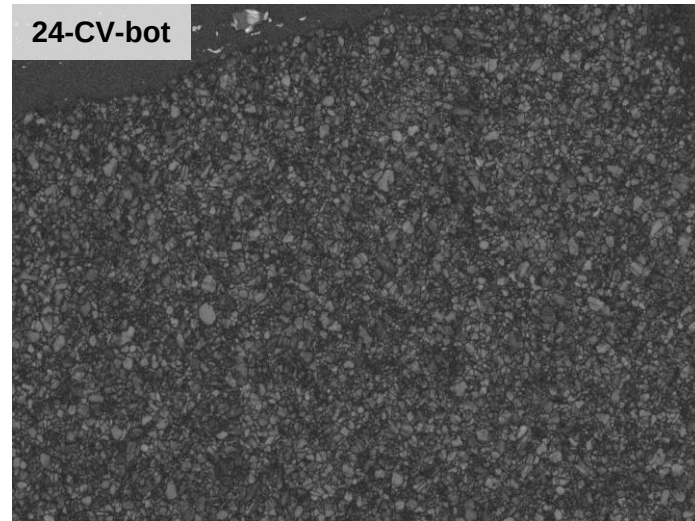
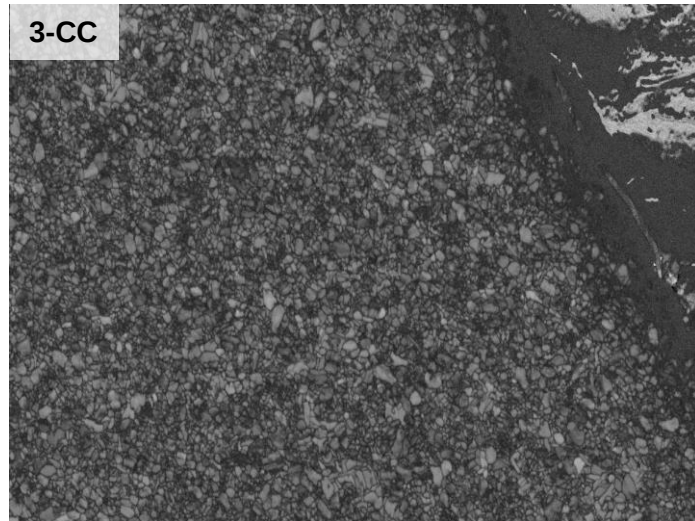
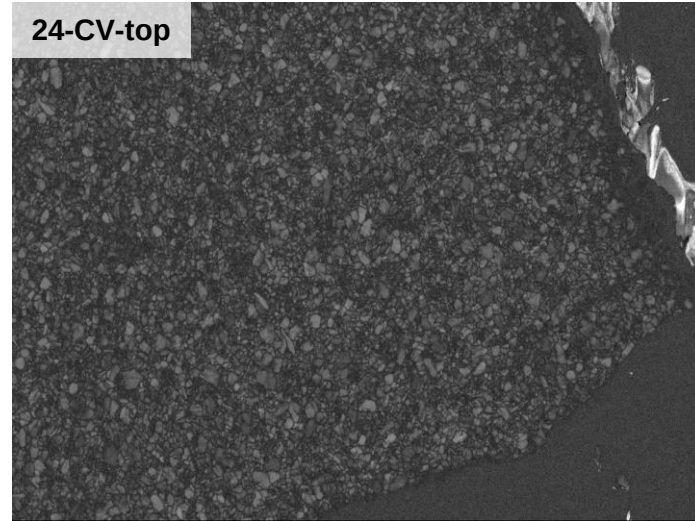
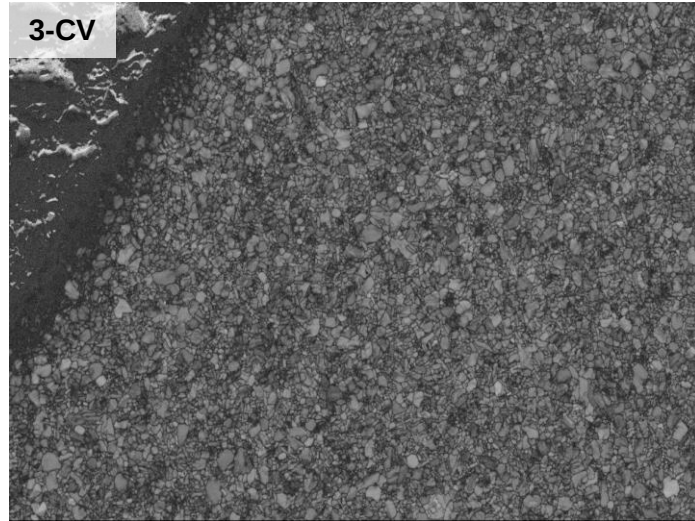
Backscattered Electron Images

Appendix C6 Backscatter images of a shot-peened only area of blade 3 as a baseline (top left) and direct comparison views of shot-peened and grit-blasted areas for blade 3 versus blade 24 in the pressure face (lower images). Both areas of blade 3 appeared more compressive (whiter) than blade 24.



Blade 24 CV Bottom

Band Contrast Maps

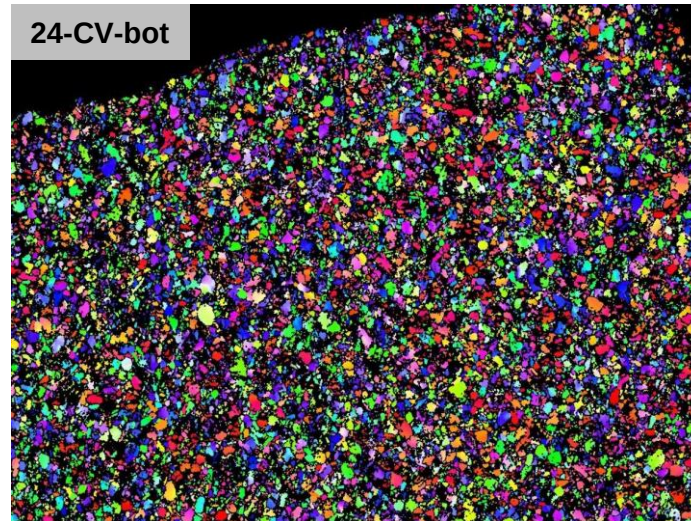
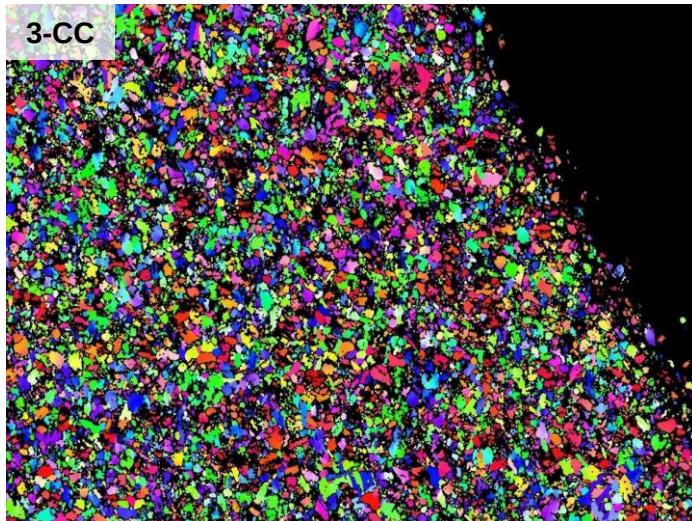
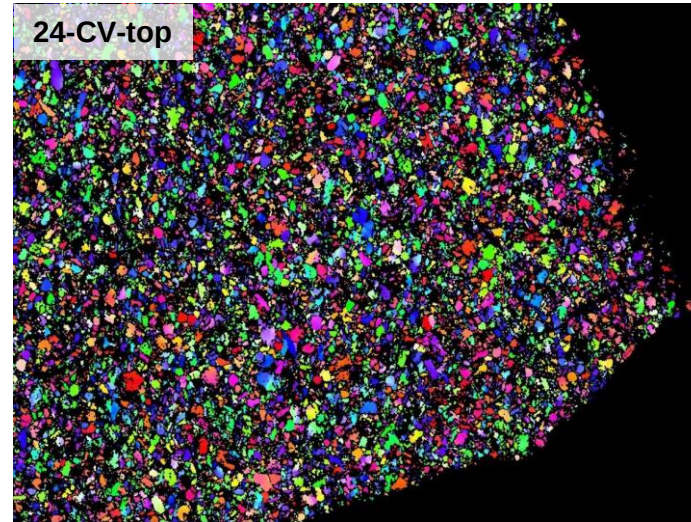
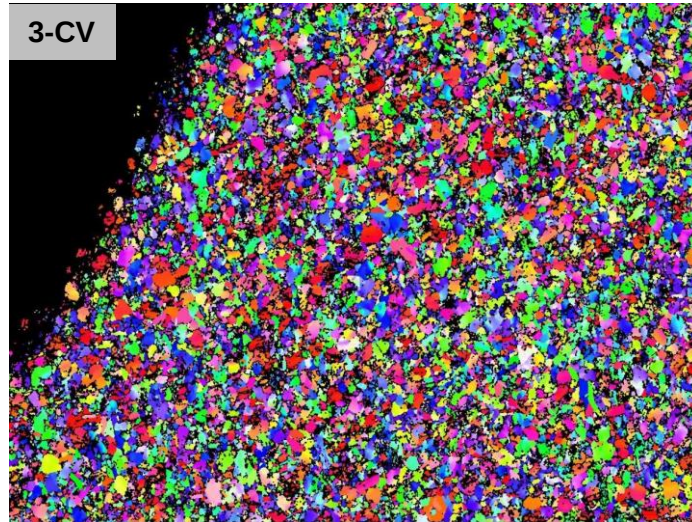
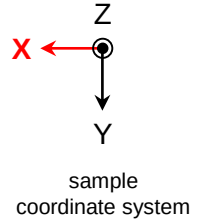
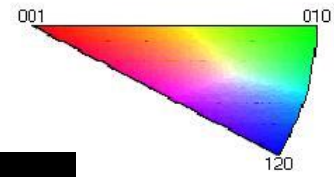


2mil

Appendix C7: Views of the band contrast maps for the different areas analyzed on the 2 blades.

IPF X Maps

• COLORED ORIENTATION || X AXIS

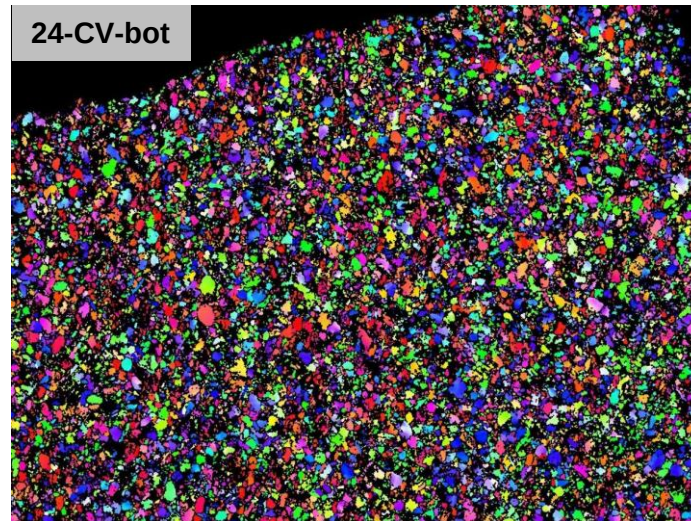
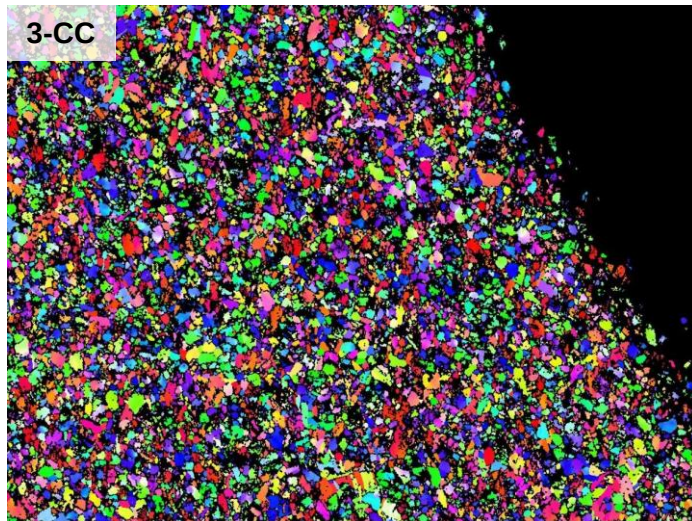
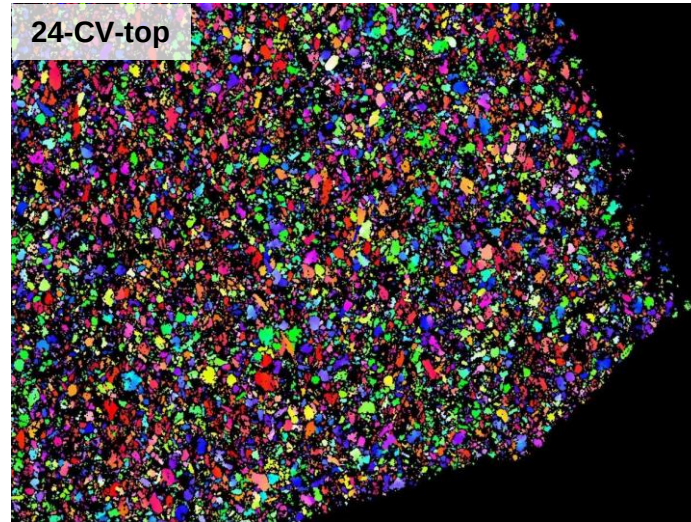
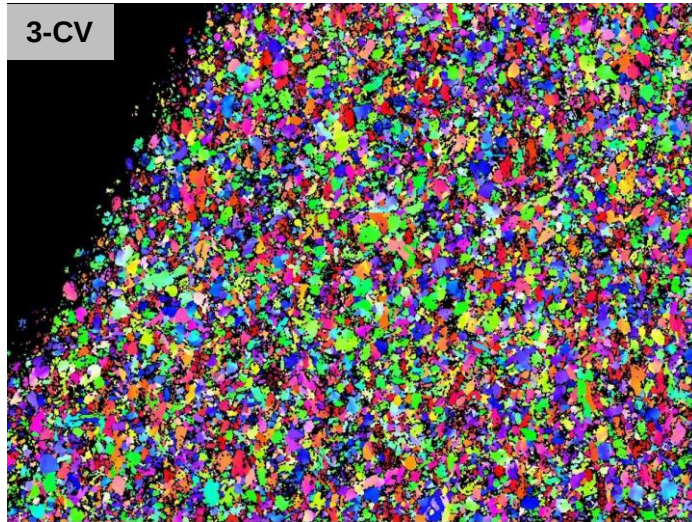
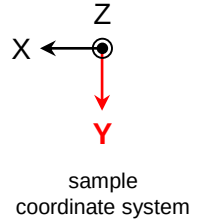
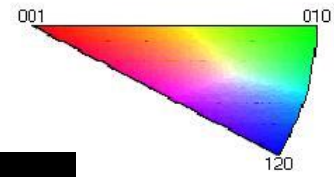


2mil

Appendix C8: EBSD data for the areas analyzed in the two blades in the X-orientation.

IPF Y Maps

• COLORED ORIENTATION || Y AXIS

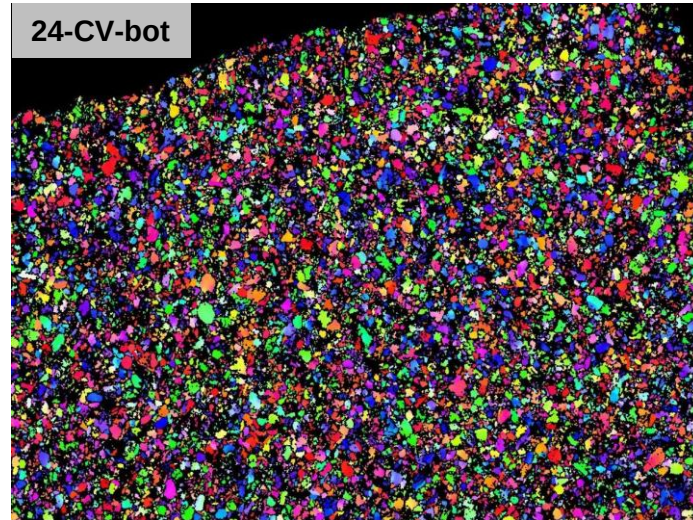
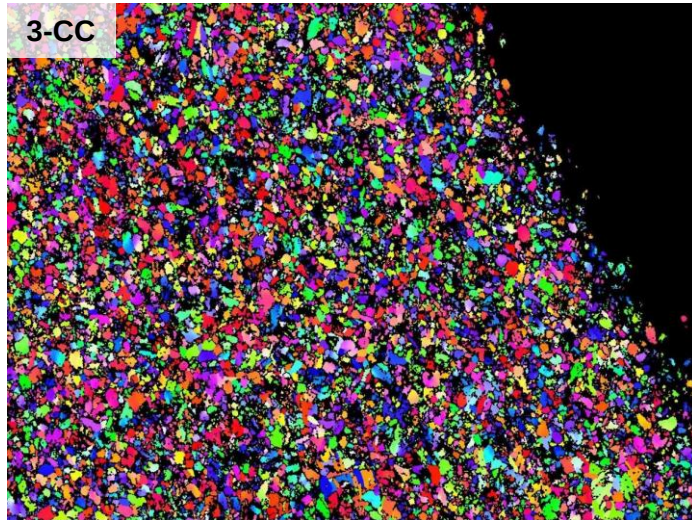
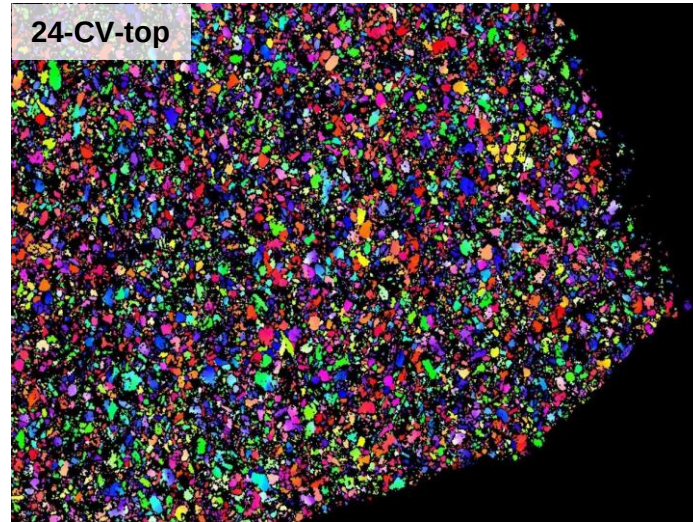
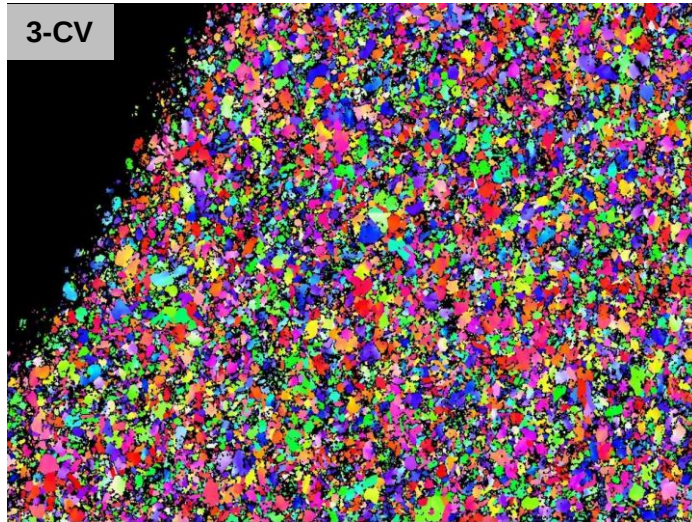
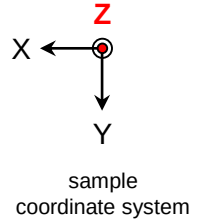
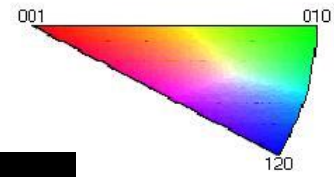


2mil

Appendix C9: EBSD data for the areas analyzed in the two blades in the Y-orientation.

IPF Z Maps

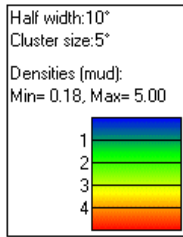
• COLORED ORIENTATION || Z AXIS



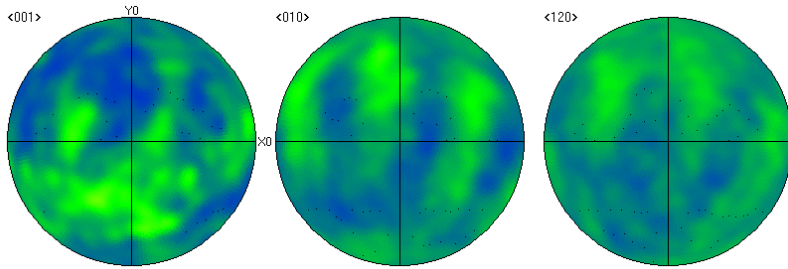
 2mil

Appendix C10: EBSD data for the areas analyzed in the two blades in the Z-orientation.

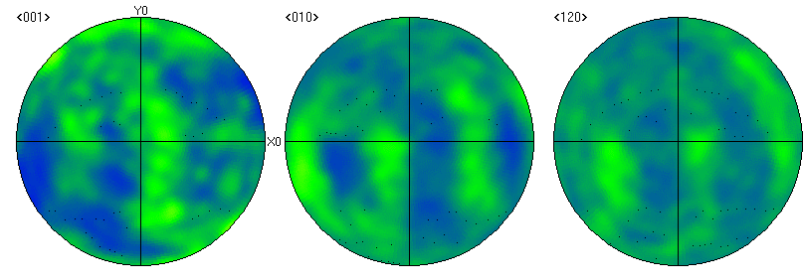
Pole Figures



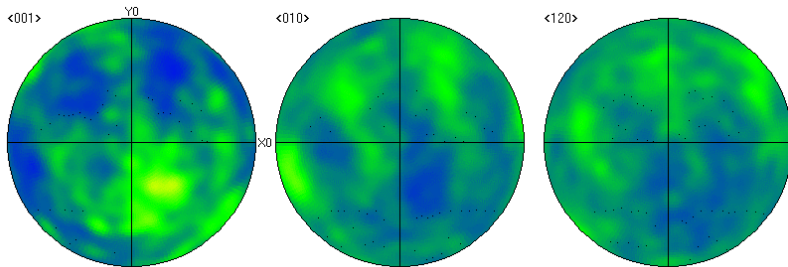
3-CV



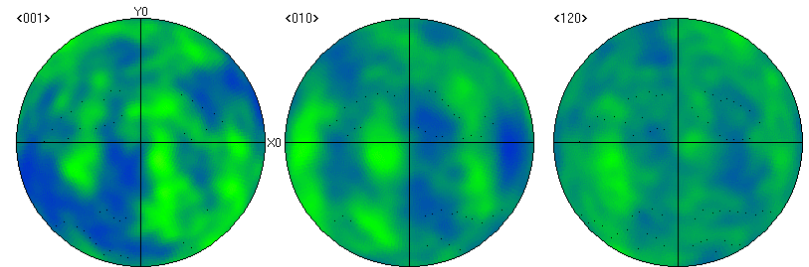
24-CV-top



3-CC



24-CV-bot



Appendix C11: EBSD pole figures looking for areas of preferential grain alignment.