Analysis of a J69-T-25 engine turbine blade fracture

Myounggu Park, Young-Ha Hwang, Yun-Seung Choi, Tae-Gu Kim

Analysis of a J69-T-25 engine turbine blade fracture

Myounggu Park[a,*], Young-Ha Hwang[a], Yun-Seung Choi[b], Tae-Gu Kim[c]

[a]*Engine Division, ATRI(Aero-Tech Research Institute), ROKAF, PO Box 304-160, Kumsa dong, Dong gu, Deagu 701-799, Republic of Korea*

[b]*Samsung Techwin Co., Ltd., 28 Sungju-Dong, Changwon City, Kyungnam 641-717, Republic of Korea*

[c]*Department of Industrial Safety and Health, College of Biomedical Science and Engineering, Inje University, Gimhae, Gyeongnam 621-749, Republic of Korea*

The fracture of a turbojet engine turbine blade was investigated. Visual and surface examination showed that the turbine blade had initially cracked by a fatigue mechanism over a period of time and then failed by overload at the last moment. The fatigue crack initiated on a surface damaged by rubbing. Also the cracked turbine blade was severly damaged by hot gas flow and discolored. This heat damage made the gamma prime phase in the matrix (Ni-base superalloy) coarsen and lowered the fatigue strength of the base material assisting to the premature fatigue fracture. The decrease of the strength of the material due to degradation of gamma prime phase was verified by hardness measurements. The possible relevance of other parts attached to the engine shaft to the fracture was reviewed. From this reviewit is inferred that the root cause of cracking and excessive heat damage might be attributed to eccentricity of the shaft resulting from various reasons—shaft misalignment, uneven wear of bearing elements and mismatch in clearance, etc. However this is an assumption that should be verified by positive supporting evidence from condition monitoring of engines.

Keywords: Engine failures; Turbine blade failures; Fatigue failure; Fretting; Overheating

1. Background

Regarding the J69-T-25 turbojet engine installed on the T-37 trainer, the turbine section consists of a turbine stator and a turbine rotor. The major function of the turbine is to extract energy from the hot gas flow to drive the compressor and the accessory gearbox. If a problem arises in the turbine section it will significantly affect the whole engine function and, of course, safety of the aircraft. In this paper a sudden turbine blade fracture leading to a fatal engine failure is investigated. The ultimate goal of this analysis is to improve the safety of the aircraft and the maintenance practice.

2. Visual examination

Fig. 1(a) is the rear view of the turbine section of the J69-T-25 engine. One of the 38 blades (see the arrow) separated without any preliminary symptoms and at the same time the fragment hit the other turbine blades, stator and casing. In Fig. 1(b) is the close-up view of the blade fracture site. It reveals that the turbine stator vanes and casing as well as the rotor assembly were badly damaged during the incident.

2.1. Turbine blade

Fig. 2 is a magnified view of the fracture surface of the turbine blade root disassembled from the turbine rotor. Macroscopically there are two distinctive regions on the fracture surface. The region I is called the 'thumb nail' shaped region which is partially discolored. This region has a smooth and flat surface. The region II has a fresher and rougher fracture surface than that of region I. From the surface shape and condition it is thought that the crack origin is at the corner of the fracture region ['O' in Fig. 2(b)]. Fig. 3 indicates the front side of the turbine blade root. It is evident from the rubbing marks that the turbine blade was touched by other objects during revolution and that surface damage might act as a stress raiser for crack initiation. Fig. 4 shows the discolored pattern of the turbine blade. Especially, the discolored pattern on the root area reveals that hot gas flow was directed onto the turbine blade root area which was not a normal situation.

2.2. Hub

Fig 5.(a) is the hub disassembled from the turbine rotor assembly. The dovetail slots in the hub are severely discolored due to hot gas flow [Fig. 5(b)]. This means that the heated gas stream was not following the normal route, which might explain the heat damage on the slots. On the front side of the dovetail slot in the hub there was a rubbing mark and this observation is in agreement with the turbine blade root having a wear mark on the front side. Therefore, it is obvious that the turbine rotor assembly was touched by an object when it revolved at high speed (see Fig. 6).

3. Fractographic analysis

Careful observation using SEM (scanning electron microscopy) showed that most of the fracture surface was covered by corrosion products. So a partial crack which was not much affected by heated gas was chosen for fractographic analysis. This crack was found during FPI (fluorescent-penetration inspection) of the turbine blades. Fig. 7 is the fluorescent indication of a crack on the turbine blade root. Fig. 8 shows the optically observed partial crack. The crack was opened with care after cutting one side. Close observation of the fracture surface revealed striations. From these results it was confirmed that the fracture mode was fatigue [see Figs. 9(a) and (b) and 10(a) and (b)]. At the crack origin fatigue striations were not observed (covered by corrosion product). In the middle and edge regions of the fracture surface, the striation marks are evident and well observable.

4. Metallography

4.1. Microstructural analysis

There was no pertinent specification about the base material of the turbine blade. To identify the material, energy-dispersive X-ray analysis was done and the results indicated that the base material was a

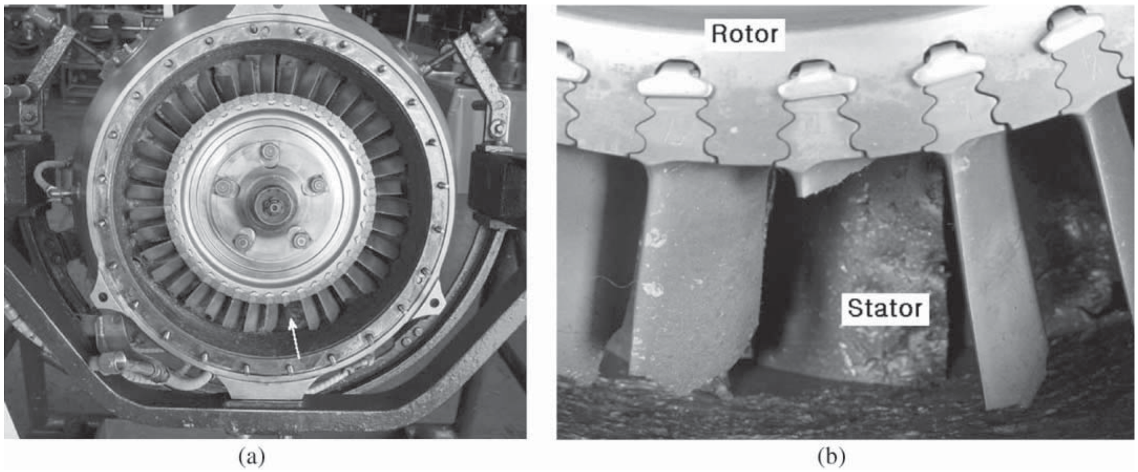

Fig. 1. The rear view of the turbine section of the J69-T-25 turbojet engine.

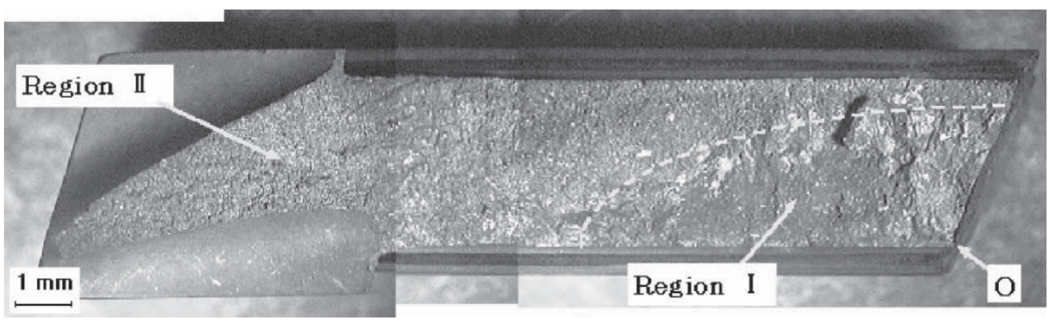

Fig. 2. The cracked and partially discolored turbine blade root (6×).

Ni-base superalloy. From the EDX spectrum (Fig. 11) the presence of Al and Ti which produce the gamma prime phase was confirmed. Longitudinal and transverse cross sections were prepared and examined revealing typical cast structure of Ni-base superalloy [Fig. 12(a) and (b)]. No signs of any microstructural anomalies were noticed.

4.2. Cracking path

To analyze the cracking path a sample which included flat and rough surfaces was prepared. The sample also has a discolored region on its surface. From the microstructure, it is confirmed that the crack growth mode changed from transdentritic to interdendritic. Transdentritic crack growth indicates that cracking occurred by a fatigue mechanism and interdendritic crack growth by overload. Usually this cracking mode transition phenomenon is one of the typical features of casting fracture by a fatigue mechanism. Fig. 13(a) and (b) shows the cracking path from the origin of the crack to the end of the crack area.

4.3. Precipitate (γ' phase) analysis

It is well known fact that nickel-base superalloy is used at temperatures of 540 °C (1000 F) and above [1]. The common strengthening mechanism of this alloy is precipitation hardening through gamma prime (γ') precipitation in the matrix. The γ' is an intermetallic compound of nominal composition [$Ni_3(Al,Ti)$] which

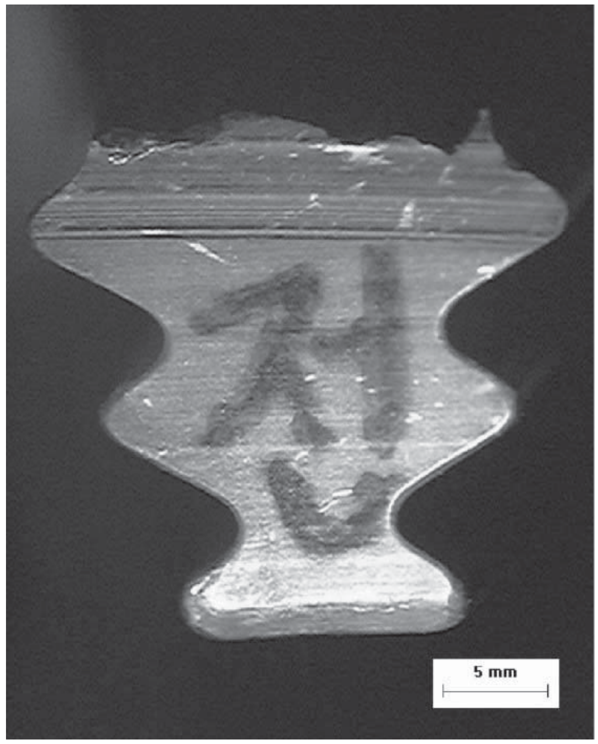

Fig. 3. The front side of the turbine blade root indicating the rubbing mark (3×).

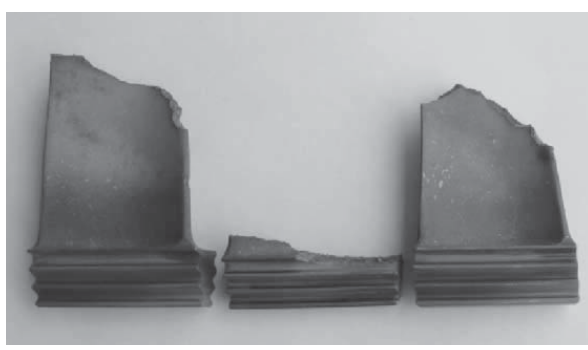

Fig. 4. Abnormal discoloration of the turbine blade root caused by heat damage.

is stable over a relatively narrow range of composition. It is precipitated as spheroidal particles in the early Ni-base alloys, which tended to have a low volume fraction of particles. Later, cuboidal precipitates were noted in alloys with higher aluminium and titanium contents. The examination of the gamma prime of the fractured turbine blade was done using FE–SEM (field emission scanning electron microscope). Fig. 14(a) is the morphology of γ' which is directly affected by heated gas and its shape is not in order. However, as shown in Fig. 14(b) the morphology of γ' which is not touched by hot gas is square and in good order. So it can be guessed that the subsurface region affected by hot gas began to coarsen. Thus, though the fatigue crack initiated by surface damage, the heat damage might have had a contributing effect on the fracture of the turbine blade.

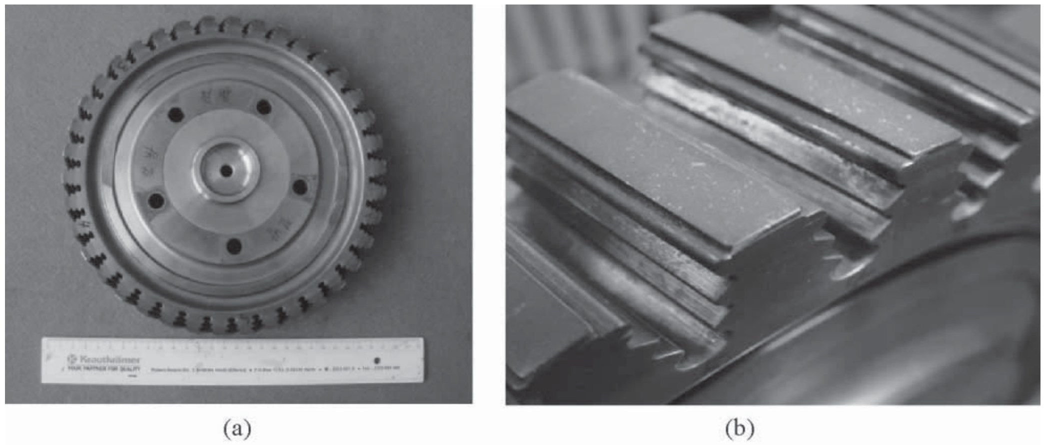

Fig. 5. The hub disassembled from the turbine rotor assembly; (a) the overall view of the hub and (b) the discolored dovetail slots in the hub.

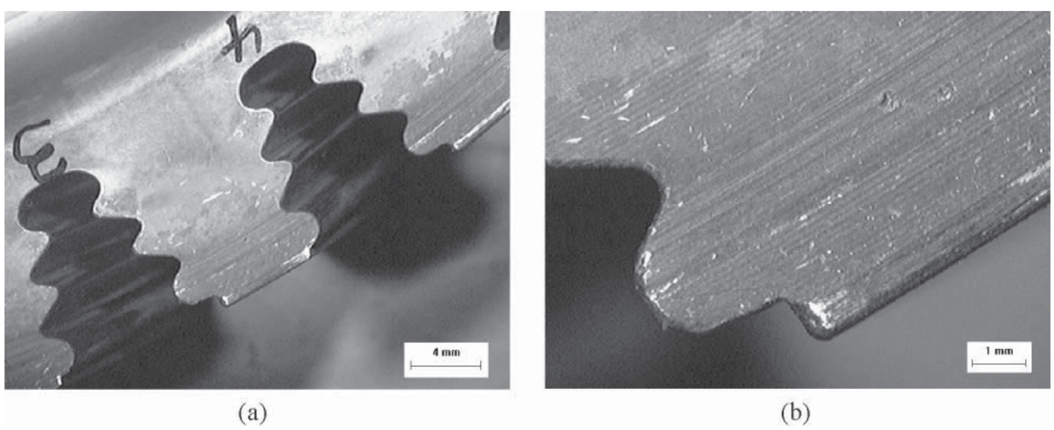

Fig. 6. (a) Wear marks on the front side of the dovetail in the hub and (b) enlarged view of Fig. 6(a).

Fig. 7. Indication of crack revealed during the FPI (fluorescent-penetration inspection).

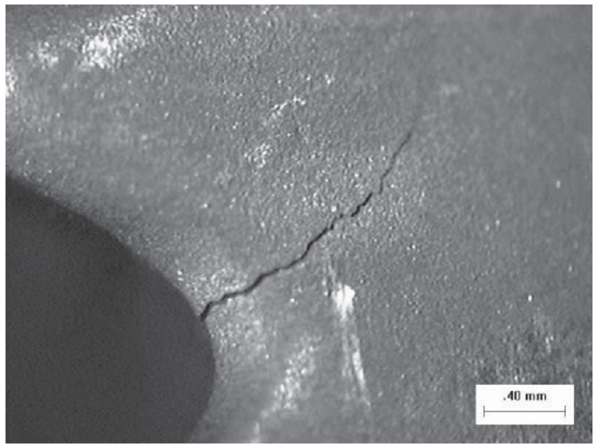

Fig. 8. Optically observed partial crack (40x).

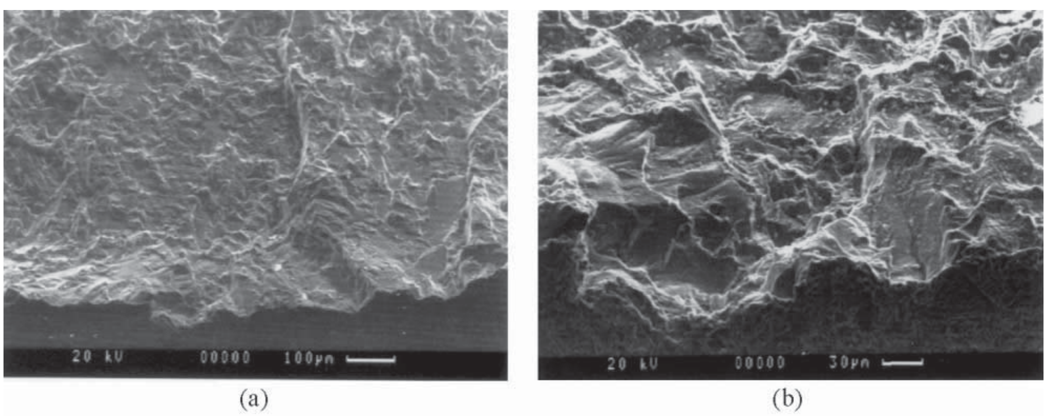

Fig. 9. Crack origin sites; (b) (300 x) is a magnified view of (a) (100 x).

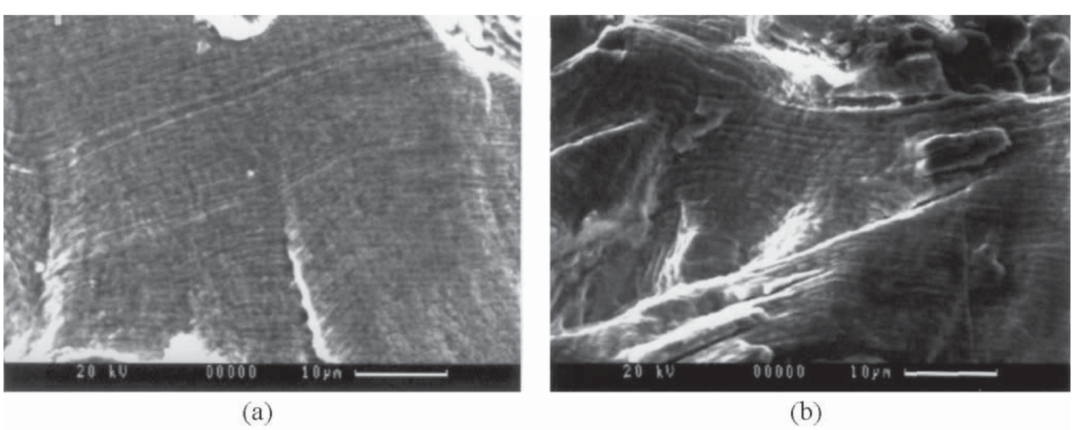

Fig. 10. Fatigue striations; (a) middle region of the fracture surface and (b) edge region of the fracture surface, both 5000 x.

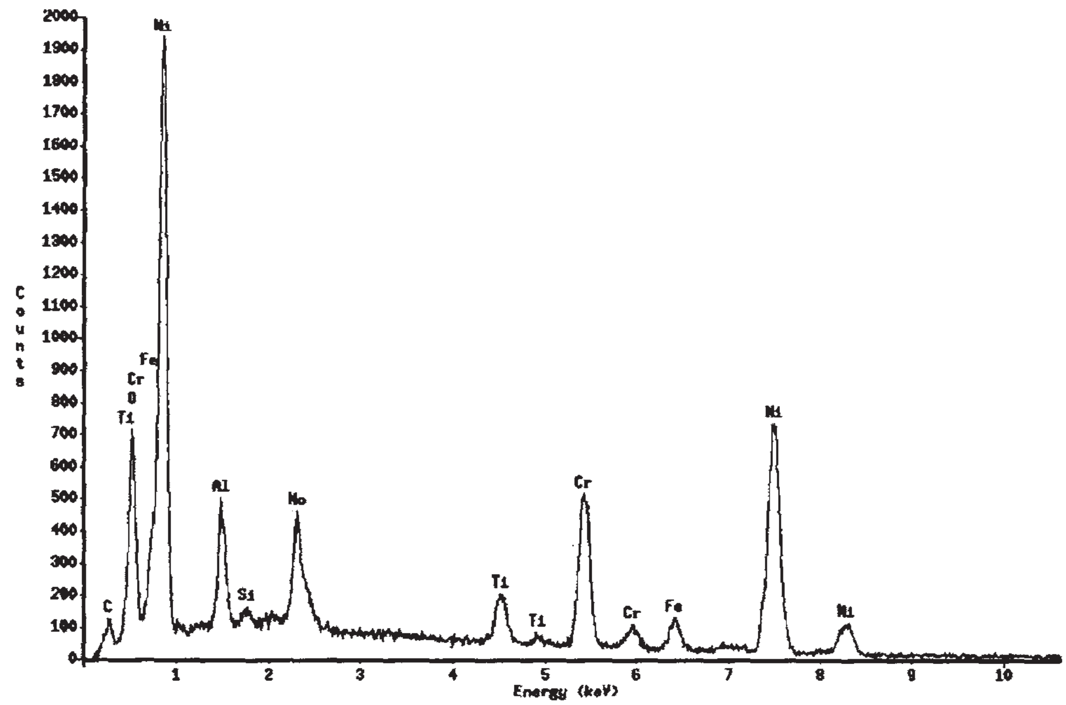

Fig. 11. EDX spectrum of turbine blade.

5. Hardness test

Microhardness measurements were taken on the longitudinal cross section including the fracture surface at a load of 1 kg. The measured microhardness values were converted into Vicker's hardness number and the results are shown in Fig. 15. The subsurface which showed degraded gamma prime had the lowest value and the hardness number increased gradually away from the subsurface. This tendency correlates with γ' phase examination results.

6. Discussion

In the previous sections it was shown that the fractured turbine blade had score marks on the surface and was excessively heated. The surface damage supplied the fatigue crack initiation sites and the abnormally directed hot gas degraded the gamma prime phase which was the main strengthening element of the matrix. Then what was the root cause of these two phenomena? To solve this problem, we consulted other case histories about the same type of engine. From the data, it is known that the compressor rotor and nearby parts—compressor cover and radial diffuser vane which have wide enough clearance—had collided. In the same context it can be postulated that the blade had also been rubbing against other parts internally. The relative positions of the parts are indicated in Fig. 16. The possible sources of this kind of failure can be shaft misalignment, uneven wear of bearing elements and mismatch in clearance, etc. If this happens internal collision takes place among parts and though the hot gas stream is normal, the turbine blade root region can be directly impinged by heated gas flow due to shaft eccentricity. However, this is an assumption and there is no positive supporting evidence of this scenario. Thus, condition monitoring of the engine is indispensible for the better understanding of these phenomena.

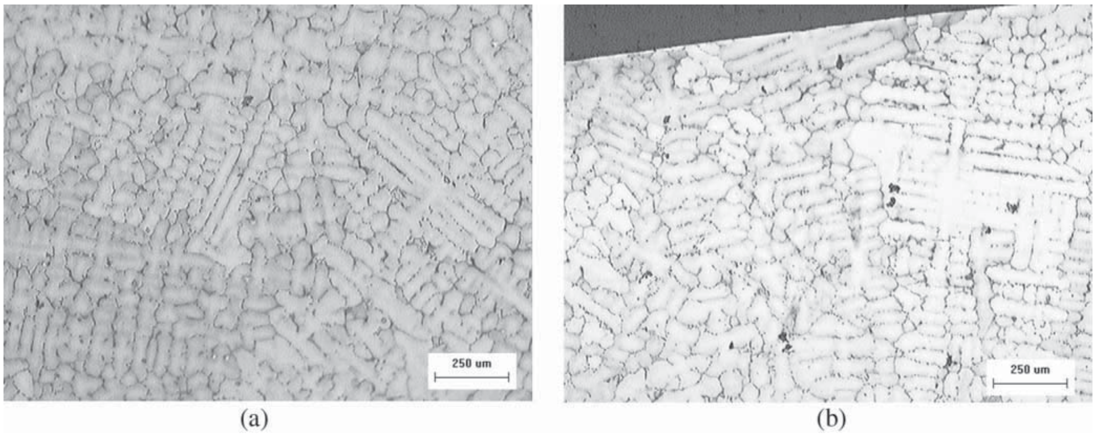

Fig. 12. Etched microstructure of the fractured turbine blade; (a) longitudinal cross section and (b) transverse cross section, both 100 x.

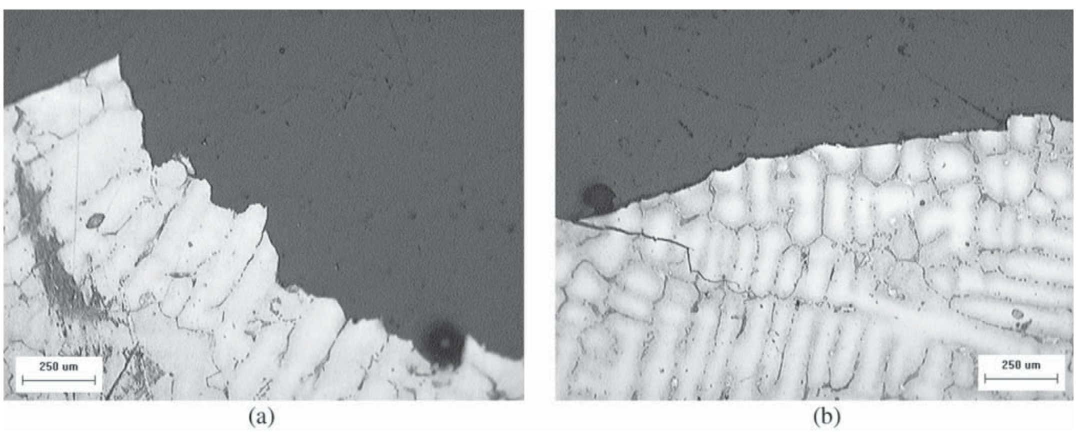

Fig. 13. Cracking path transition; (a) transdendritic; fracture mode is fatigue and (b) interdendritic; fracture mode is overload, both 200x.

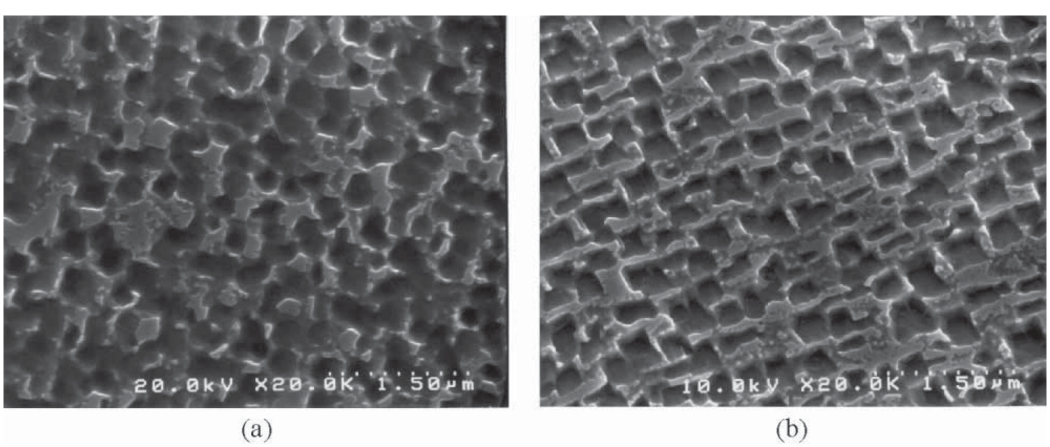

Fig. 14. The morphology of γ' phase; (a) the subsurface region of the fracture surface affected by heated gas flow and (b) the region without heated gas flow, both 20, 000x.

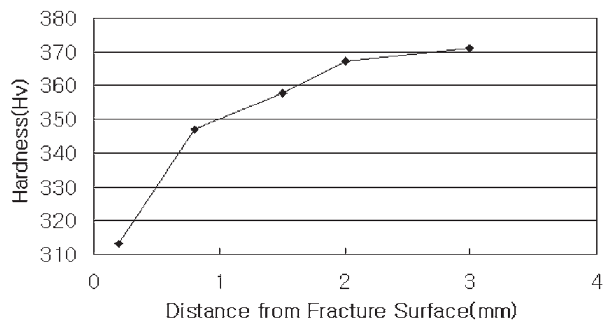

Fig. 15. Hardness test results of the fractured turbine blade.

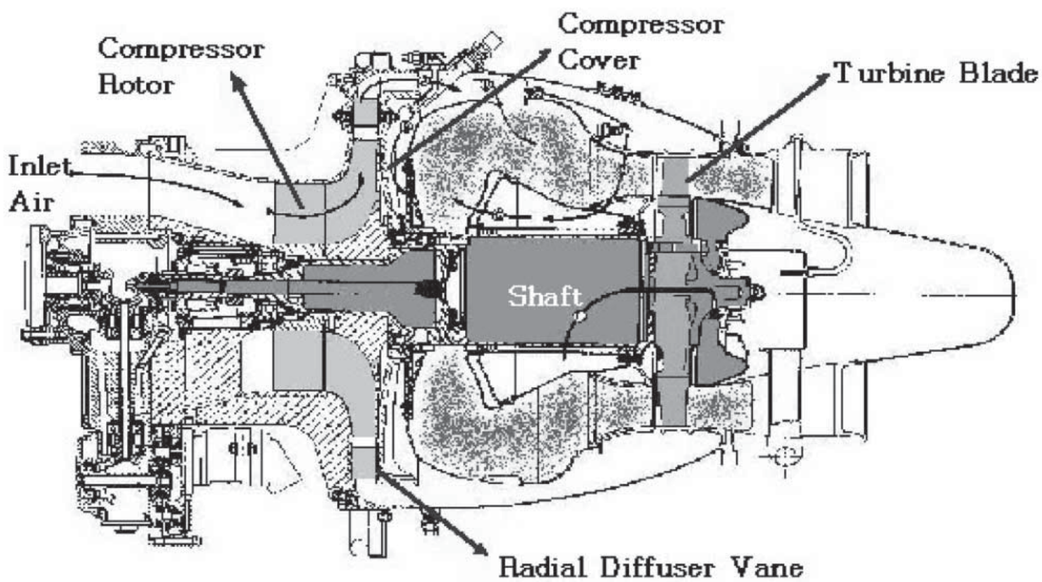

Fig. 16. The cross sectional view of the J69-T-25 turbojet engine.

7. Conclusion

From a consideration of the results, the premature fatigue fracture of the turbine blade can be attributed to the surface damage on the front side and the abnormally induced heat damage. The root cause of these two phenomena is now under systematic study. During depot and field maintenance, the condition of the shaft and its relative parts is thoroughly monitored. The results of the engine condition monitoring will be ready to feedback into improvement of engine integrity and aircraft safety.

Reference

[1] AMS specialty handbook—heat-resistant materials. ASM International, 1997. p. 221–54.