Design and Analysis of a Gas Turbine Blade

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Abstract – *High pressure temperature (HPT) turbine blade is the most important component of the gas turbine and failures in this turbine blade can have dramatic effect on the safety and performance of the gas turbine. It could be concluded that the turbine blade failure might be caused by multiple failure mechanisms such as hot corrosion, erosion and fatigue. Hot corrosion could have reduced the thickness of the blade material and thus weakened the blade. This reduction of the blade thickness decreases the fatigue strength which ultimately led to the failure of the turbine blade.*

Turbine blades are subjected to very strenuous environments inside a gas turbine. They face high temperatures, high stresses, and a potentially high vibration environment. All these factors can lead to blade failure, resulting in catastrophic failure of turbine.

The external and internal surface damages include corrosion, oxidation, crack formation, erosion, foreign object damage and fretting. The internal damage of microstructure include γ' phase, CoNi3 [(Al, Ti)] phase aging (rafting), grain growth, brittle phases formation, carbides precipitation, creep and grain boundary void formation. These damages produce dimensional change which results in increase in operational stress that leads to deterioration in turbine efficiency. The deterioration of blade material is related to the high gas temperature, high steady state load levels (centrifugal load) and high thermal transient load (trips, start-ups, start downs). In this research, a review of common failures due to metallurgical defects found in gas turbine discussed is presented.

Key Words: **Turbine blade, corrosion, cooling system, overheating**

1. INTRODUCTION

A turbine blade is the individual component which makes up the turbine section of a gas turbine or steam turbine. The blades are responsible for extracting energy from the high temperature, high pressure gas produced by the combustor. The turbine blades are often the limiting component of gas turbines to survive in this difficult environment, turbine blades often use exotic materials like superalloys and many different methods of cooling that can be categorized as internal and external cooling, and thermal barrier coatings. Blade fatigue is a major source of failure in steam turbines and gas turbines. Fatigue is caused by the stress induced by vibration and resonance within the operating range of machinery. To protect blades from these high dynamic stresses, friction dampers are used.

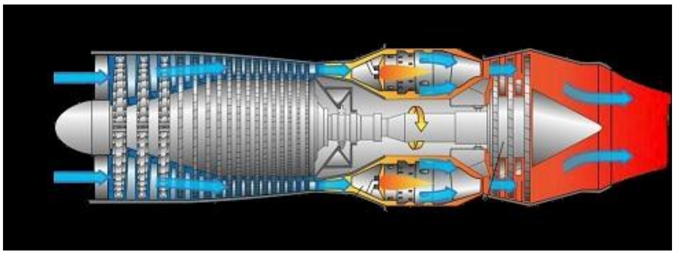

Fig 1.1: Turbojet Engine

In a gas turbine engine, a single turbine section is made up of a disk or hub that holds many turbine blades. That turbine section is connected to a compressor section via a shaft (or "spool"), and that compressor section can either be axial or centrifugal. Air is compressed, raising the pressure and temperature, through the compressor stages of the engine. The temperature is then greatly increased by combustion of fuel inside the combustor, which sits between the compressor stages and the turbine stages. The high-temperature and high- pressure exhaust gases then pass through the turbine stages. The turbine stages extract energy from this flow, lowering the pressure and temperature of the air and transfer the kinetic energy to the compressor stages along the spool. This process is very similar to how an axial compressor works, only in reverse. The number of turbine stages varies in different types of engines, with high-bypass-ratio engines tending to have the most turbine stages. The number of turbine stages can have a great effect on how the turbine blades are designed for each stage. Many gas turbine engines are twin-spool designs, meaning that there is a high-pressure spool and a low-pressure spool. Other gas turbines use three spools, adding an intermediate-pressure spool between the high- and low-pressure spool. The high-pressure turbine is exposed to the hottest, highest-pressure air, and the low-pressure turbine is subjected to cooler, lower-pressure air. The difference in conditions leads to the design of high-pressure and low-pressure turbine blades that are significantly different in material and cooling choices even though the aerodynamic and thermodynamic principles are the same.Under these severe operating conditions inside the gas and steam turbines, the blades face high temperature, high stresses, and potentially high vibrations. Steam turbine blades are critical components in power plants which convert the linear motion of high-temperature.

Fig 1.2: Gas Turbine rotor blade

Turbojet engine is the power plant of modern day's aircraft jet systems as it does not only produce the trust required by an aircraft for propulsion but also the power that enables the operation of other components in the aircraft. A typical jet engine operates with the principles of Newton's third law of motion which states that a given force exerted on a body will generate equal and opposite force of action.

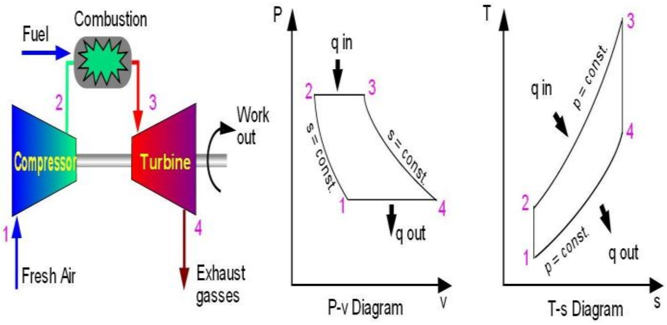

Fig 1.3: PV and TS diagrams

A study was conducted by Blotch (1982) for gas turbine failures and concluded that turbine blades and rotor component contributed to 28 percent of primary causes of gas turbine failures, whereas 18 percent is due to faults in turbine nozzles and stationary parts. Another study was done by Dundas (1994) for gas turbine losses and observed that creep, high cycle fatigue (HCF) and turbine blade cooling related failures added 62% of the total damage costs for gas turbines. In the year 1992, another study was conducted by Scientific Advisory Board (SAB) of the United States Air Force and it was concluded that high cycle fatigue (HCF) is the single biggest cause of turbine engine failures (Ritchie et al., 1999). The degree of blade material deterioration for individual blade differs and it depends on several factors such as total service time, operational conditions, manufacturing process and history of turbine. The common failures found in gas turbine blades due to metallurgical defects are discussed and illustrated.

1.1 Challenges of turbojet technology

To progress to the performance capabilities of today, two goals were (and still are) being pursued:

1. Increase the thermodynamic cycle efficiency by increasing the compressor pressure ratio.

2. Increase the ratio of power-output to engine weight by increasing the turbine inlet temperature

Trend in compressor pressure ratios

Calendar years	Compressor pressure ratio
Late 1930 to 1940	3:1 to about 6:1
Early 1950	About 10:1
1950 to 1960	20:1 to about 25:1
2000	30:1 to about 40:1

2. CAUSES OF FAILURE

2.1 Corrosion

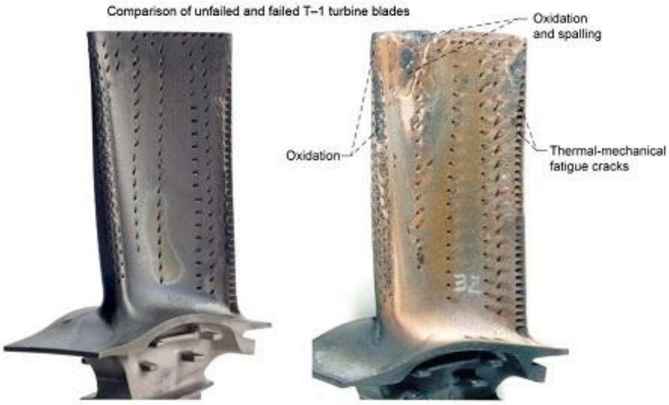

Fig 2.1: blade failure due to corrosion

Corrosion failure of machine components and structures has been studied. From the study, it is observed that maximum outages in the gas turbine power plants are due to corrosion fatigue failure of turbine blades. Many researchers have done metallographic and fractrographic analysis of failed turbine blades and concluded that chemicals compounds such as oxide, chromium sulfide and complex sulfides play significant role in reducing fatigue strength of turbine blades.

2.2 Fatigue

Fig 2.2: blade failure due to fatigue

Turbine blades are most susceptible to crack formation in regions of contact surfaces, which are exposed to both centrifugal loading and oscillatory vibrations. These mating component are failed due to fretting fatigue. Fretting fatigue results in an increase in tensile and shear stress at the contact surface, which leads to crack initiation and its propagation till failed completely. At elevated temperatures, fatigue cracks have been observed to initiate from grain boundaries, slip bands, pores, twin boundaries or due to cracking of inclusions/precipitates. The majority of turbine failure includes fatigue leading to crack initiation and propagation. The operating conditions of high rotational speed at elevated temperature, corrosion, erosion and oxidation accelerates fatigue failure.

2.3 Overheating

Fig 2.3: blade failure due to overheating

The surface degradation of turbine blade by the formation of needle separation topologically closed packed phase (TCP) occurs due to overheating. This structural instability of the alloy results in decrease in strength and ductility of alloy (Rybhino et al., 2012). Tawancy et al., (2009) have examined the first stage turbine blades and vanes and concluded that the failure of both components took place due to overheating. Overheating promoted creep, resulting in inter-granular cracking that shortened the fatigue life of blades and vanes.

2.4 Damageability of gas turbine blades

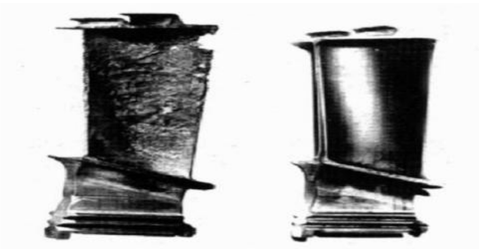

Fig 2.4: Failures to gas turbine rotor blades

a) – fatigue cracking of leading edge

b) – fatigue fracture located at the blade's locking piece p

2.5 Thermal failures

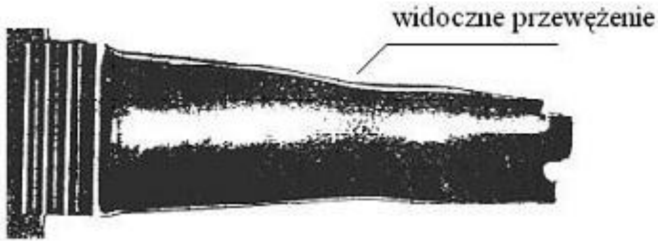

Fig 2.5: Plastic deformation of the blade (Bogdan, 2009).

Fig 2.6: Characteristic forms of failures caused by overheating of blade material

Fig 2.7: Characteristic forms of failures to gas turbine caused by long-lasting excessive temperature of exhaust gases

2.6 Chemical failure

Blade deformations in the form of dents are caused by a foreign matter ingested by the turbojet engine compressor and by particles of metal and hard carbon deposits from the combustion chamber. Such dents result in stress concentrations in blade material and prove conducive to the initiation of fatigue processes. Scratches on blade surfaces due to the foreign matter impact are also reasons for local stress concentrations and, consequently, potential corrosion centers. What results is, again, material fatigue which, together with possible corrosion, prove conducive to fatigue fracture.

Fig 2.8: Failures to turbine blades operated in the seashore environment, caused by chemical impact of exhaust gases

3. COOLING

At a constant pressure ratio, thermal efficiency of the engine increases as the turbine entry temperature (TET) increases. However, high temperatures can damage the turbine, as the blades are under large centrifugal stresses and materials are weaker at high temperature. So, turbine blade cooling is essential. Current modern turbine designs are operating with inlet temperatures higher than 1900 kelvins which is achieved by actively cooling the turbine components. Cooling of components can be achieved by air or liquid cooling. Liquid cooling seems to be more attractive because of high specific heat capacity and chances of evaporative cooling but there can be leakage, corrosion, choking and other problems. Which works against this method. On the other hand, air cooling allows the discharged air into main flow without any problem. Quantity of air required for this purpose is 1–3% of main flow and blade temperature can be reduced by 200–300 °C. There are many techniques of cooling used in gas turbine blades; convection, film, transpiration cooling, cooling effusion, pin fin cooling etc. which fall under the categories of internal and external cooling. While all methods have their differences, they all work by using cooler air (often bled from the compressor) to remove heat from the turbine blades.

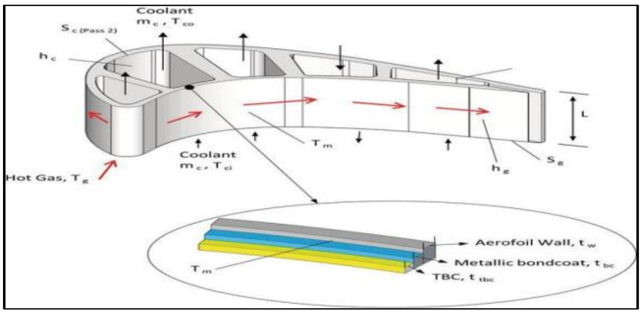

Fig 3.1: Cooling passages

3.1 Internal cooling

a) Convection cooling

It works by passing cooling air through passages internal to the blade. Heat is transferred by conduction through the blade, and then by convection into the air flowing inside of the blade. A large internal surface area is desirable for this method, so the cooling paths tend to be serpentine and full of small fins. The internal passages in the blade may be circular or elliptical in shape. Cooling is achieved by passing the air through these passages from hub towards the blade tip. This cooling air comes from an air compressor. In case of gas turbine the fluid outside is relatively hot which passes through the cooling passage and mixes with the main stream at the blade tip

b) Impingement cooling

A variation of convection cooling, impingement cooling, works by hitting the inner surface of the blade with high velocity air. This allows more heat to be transferred by convection than regular convection cooling does. Impingement cooling is used in the regions of greatest heat loads. In case of turbine blades, the leading edge has maximum temperature and thus heat load. Impingement cooling is also used in mid chord of the vane. Blades are hollow with a core. There are internal cooling passages. Cooling air enters from the leading edge region and turns towards the trailing edge.

3.2 External cooling

Film cooling (also called thin film cooling), a widely used type, allows for higher cooling effectiveness than either convection and impingement cooling. This technique consists of pumping the cooling air out of the blade through multiple small holes or slots in the structure. A thin layer (the film) of cooling air is then created on the external surface of the blade, reducing the heat transfer from main flow, whose temperature (1300 – 1800 kelvins) can exceed the melting point of the blade material (1300 – 1400 kelvins). The ability of the film cooling system to cool the surface is typically evaluated using a parameter called cooling effectiveness. Higher cooling effectiveness (with maximum value of one) indicates that the blade material temperature is closer to the coolant temperature. In locations where the blade temperature approaches the hot gas temperature, the cooling effectiveness approaches to zero. The cooling effectiveness is mainly affected by the coolant flow parameters and the injection geometry. Coolant flow parameters include the velocity, density, blowing and momentum ratios which are calculated using the coolant and mainstream flow characteristics. Injection geometry parameters consist of hole or slot geometry (i.e. cylindrical, Shaped holes or slots) and injections angle. A United States Air Force program in the early 1970s funded the development of a turbine blade that was both film and convection cooled, and that method has become common in modern turbine blades. Injecting the cooler bleed into the flow reduces turbine isentropic efficiency; the compression of the cooling air (which does not contribute power to the engine) incurs an energetic penalty; and the cooling circuit adds considerable complexity to the engine. All of these factors have to be compensated by the increase in overall performance (power and efficiency) allowed by the increase in turbine temperature. In recent years, researchers have suggested using plasma actuator for film cooling. The film cooling of turbine blades by using a dielectric barrier discharge plasma actuator was first proposed by Roy and Wang. A horseshoe-shaped plasma actuator, which is set in the vicinity of holes for gas flow, has been shown to improve the film cooling effectiveness significantly. Following the previous research, recent reports using both experimental and numerical methods demonstrated the effect of cooling enhancement by 15% using plasma actuator.

3.3 Film cooling

Film cooling involves directing bleed air from the compressor through the shaft to the turbine blades and out of holes at various locations on the blade. The coolant air creates a lower temperature film on the blade surface protecting the blade from the hot mainstream gases. Kwak and Han performed transient liquid crystal experiments using a combined multiband and narrow band technique to obtain heat transfer and film cooling effectiveness results on a scaled GE-E3 turbine blade. Their results showed high heat transfer coefficients near the leading edge and increases in blowing ratio reduced the heat transfer coefficients. Also, film cooling effectiveness was proportional to blowing ratio. In a numerical study, Acharya, et. al, focused on the effects of film cooling on a squealer tip and a flat tip turbine blade. On the flat tip, the leakage vortex was shown to be effected by the coolant injection. High effectiveness was reported in the trajectory of the injected coolant air. The squealer rim case showed decreased film cooling effectiveness as compared to the flat tip case. Ahn et al. reported measurements of tip cooling for a range of blowing ratios. A pressure sensitive paint technique was used to determine the film cooling effectiveness.

This technique does not allow the determination of heat transfer coefficients. Film cooling effectiveness was shown to be directly related to increasing BR on the blade tip. Using the same experimental technique, Gao et. al., studied the effect of off-design inlet angle conditions ranging ± 5o from the design condition. It was concluded that off design inlet angle conditions did not have a significant blade averaged effect on film cooling effectiveness. Effectiveness was found to increase up to 25% in the tip cavity for the positive angle experiment. Nasir et. al. examined heat transfer coefficients on a GE HPT blade tip model featuring tip and pressure side film cooling holes. For the case with tip and pressure side injection, high effectiveness was observed for a plane and recessed tip. Lift off was observed at BR=3 on the tip. Effectiveness was not seen in the cases with pressure side only injection. In an earlier study of a film cooled blade model, Kim and Metzger found that pressure side injection at a high blowing ratio was very effective at reducing the thermal gradient on a plane tip turbine blade. The slot-shaped coolant holes were closely spaced near the pressure side and showed broad coverage in the defined spanwise direction. Teng, examined the effect of unsteady wakes on film temperature and film cooling effectiveness on the suction side of a gas turbine blade. The wake was shown to decrease the film cooling effectiveness on the suction side. Also, the wake seemed to have a large affect on the location of boundary layer transition; while the injection did not seem to affect the transition. In cutback squealer designs, a portion of the squealer rim is removed (cutback) near the PS trailing edge to allow air within the squealer rim to exit at the trailing edge. This experiment used a pressure sensitive paint here technique to determine film cooling effectiveness. Their results show largest pressure side film cooling effectiveness at BR=1 and BR=1.5. The tip FCE was found to be largest near the trailing edge. With the cutback squealer design, PS side only film cooling showed little effect on the cavity floor. A numerical and experimental study by Wang, et. al. used CFX with validation by particle image velocimetry to investigate the tip leakage flows on a scaled-up GE-E3 blade with film cooling and a cutback squealer. The blade had several holes along the camber line and several on the pressure side. They illustrated the leakage vortex formation due to large velocity differences from the tip gap to the suction side. Their results showed that large blowing ratios resulted in a decrease in leakage mass flow rates and that the camber line holes were most effective at reducing the leakage flow rate. A numerical study was performed on the identical blade tip with matching conditions and similar blowing ratios to the present study. Heat transfer coefficients were found to be largest near the leading edge at all blowing ratios. This was found to be caused by impingement/reattachment of the leakage flow near the leading edge of the blade. Increases in film cooling effectiveness showed increased coverage with largest values near the camber line and suction side. The tip coolant air was seen to exhibit ―lift off ‖ starting at BR=2.9 and also at BR=4.7. These BR ' s exhibited large coverage with fairly high film cooling effectiveness. At BR=1 and 1.8, film cooling effectiveness was found to be high locally and immediately downstream of the tip holes. Newton, et. al., used a transient liquid crystal technique to determine heat transfer coefficients and film cooling effectiveness values on a flat turbine blade tip with ten tip holes. Their results for BR=.99 can be seen in Fig 09. Highest heat transfer was found to be at the point of flow reattachment near the leading edge. High film cooling effectiveness was found near BR=.5-.8. Local effectiveness ―streaks‖ were seen downstream of the film cooling holes. Higher heat transfer coefficients were seen surrounding the film cooling holes due to local acceleration of the leakage flow around the ― blockage‖ formed by the presence of film cooling air. Film cooling experiments on a 12x scaled flat tip turbine blade were performed in a low-speed wind tunnel cascade by Christophel, et. al. Their results showed high local ― streaking ‖ effectiveness caused by the pressure side injection and better performance with a smaller tip gap. Also, with a small tip gap, increases in blowing ratio resulted in increases in adiabatic effectiveness (film cooling effectiveness). With a larger tip gap, increases in blowing ratio resulted in decreased or constant film cooling effectiveness. Part II of this study examined the heat transfer coefficients associated with the pressure side injection. Coolant injection was found to increase heat transfer over the case studied with no coolant injection. Overall, the net heat flux reduction was found to improve with injection of coolant air

4. THERMAL BARRIER COATING

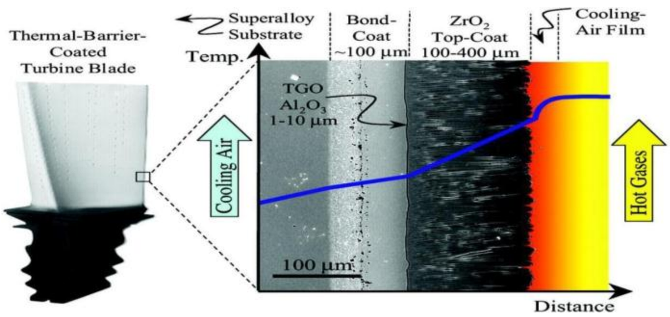

Fig 4.1: Thermal barrier coated turbine blade

Thermal barrier coatings (TBCs) are advanced materials systems usually applied to metallic surfaces operating at elevated temperatures, such as gas turbine or aero-engine parts, as a form of exhaust heat management. These 100 µm to 2 mm thick coatings of thermally insulating materials serve to insulate components from large and prolonged heat loads and can sustain an appreciable temperature difference between the load-bearing alloys and the coating surface. In doing so, these coatings can allow for higher operating temperatures while limiting the thermal exposure of structural components, extending part life by reducing oxidation and thermal fatigue. In conjunction with active film cooling, TBCs permit working fluid temperatures higher than the melting point of the metal airfoil in some turbine applications. Due to increasing demand for more efficient engines running at higher temperatures with better durability/lifetime and thinner coatings to reduce parasitic mass for rotating/moving components, there is significant motivation to develop new and advanced TBCs. The material requirements of TBCs are similar to those of heat shields, although in the latter application emissivity tends to be of greater importance.

4.1 Structure

An effective TBC needs to meet certain requirements to perform well in aggressive thermo-mechanical environments. To deal with thermal expansion stresses during heating and cooling, adequate porosity is needed, as well as appropriate matching of thermal expansion coefficients with the metal surface that the TBC is coating. Phase stability is required to prevent significant volume changes (which occur during phase changes), which would cause the coating to crack or spall. In air-breathing engines, oxidation resistance is necessary, as well as decent mechanical properties for rotating/moving parts or parts in contact. Therefore, general requirements for an effective TBC can be summarize as needing:

1) A high melting point.
2) No phase transformation between room temperature and operating temperature.
3) Low thermal conductivity.
4) Chemical inertness.
5) Similar thermal expansion match with the metallic substrate.
6) Good adherence to the substrate.
7) Low sintering rate for a porous microstructure. These requirements severely limit the number of materials that can be used, with ceramic materials usually being able to satisfy the required properties

4.2 Failure

TBCs fail through various degradation modes that include mechanical rumpling of bond coat during thermal cyclic exposure, especially, coatings in aircraft engines; accelerated oxidation, hot corrosion, molten deposit degradation. There are also issues with oxidation (areas of the TBC getting stripped off) of the TBC, which reduces the life of the metal drastically, which leads to thermal fatigue. A key feature of all TBC components is well matched thermal expansion coefficients between all layers. Thermal barrier coatings expand and contract at different rates upon heating and cooling of the environment, so materials when the different layers have poorly matched thermal expansion coefficients, a strain is introduced which can lead to cracking and ultimately failure of the coating. Cracking at the thermally-grown oxide (TGO) layer between the top-coat and bond-coat is the most common failure mode for gas turbine blade coatings. TGO growth produces a stress associated with the volume expansion which persists at all temperatures. When the system is cooled, even more mismatch is introduced from the mismatch in thermal expansion coefficients. The result is very high (2-6GPa) stresses which occur at low temperature and can produce cracking and ultimately fracture of the barrier coating. TGO formation also results in depletion of Al in the bond-coat. This can lead to the formation of undesirable phases which contribute to the mismatch stress. These processes are all accelerated by the thermal cycling which occurs in many thermal barrier coating applications

5. THERMAL ANALYSIS OF GAS TURBINE BLADE

5.1 Steady state thermal analysis

In a gas turbine blade, boundary layer develops on the blade surface and the free stream temperature are of interest. This layer acts as a buffer between the solid blade and the hot free stream, and offers resistance to heat transfer. Heat transfer occurs in this viscous layer between the blade and the fluid through both conduction and convection. After inputting the boundary conditions presented in the above Table and applying it on the gas turbine blade, the following results were obtained for IN 738 and U500 blade materials as shown in Figs. This boundary condition caused convective heat transfer to occur through one or more flat or curved faces (in contact with a fluid). Exhaust gases from the combustor are directed through the turbine in such a manner that the hottest gases impinge on turbine blades. It was observed that the maximum temperature is experienced at the leading edge of the blade, however, there was a temperature fall from the leading edge to the trailing edge of the blade. Since heat is transferred from the region of high temperature to a region of low temperature, the maximum heat flux was observed at the trailing edge. Fig.

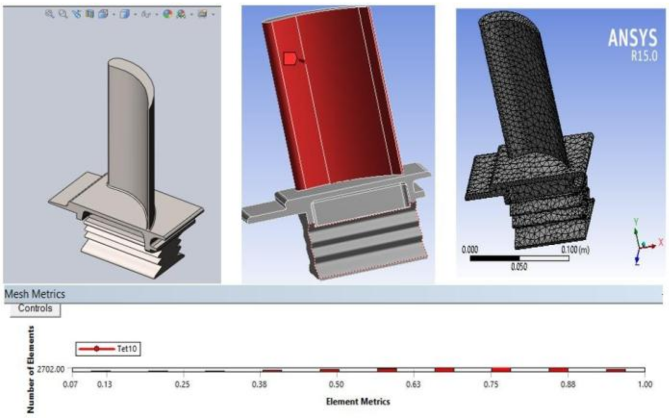

Fig 5.1: Gas Turbine Blade Geometries in SOLIDWORKS and Meshed Blade using ANSYS

The temperatures observed were below the melting temperature of the blade materials, as both IN738 and U500 turbine blade materials exhibited high temperatures of 736oC and 728oC as shown in Fig 8. Depending on the severity of heat flux in the gas turbine engine, the temperature can have significant effects on the overall turbine blade performance. The non-uniform temperature distribution from the tip to the root of the blade materials may induce thermal stresses on the turbine blade, while thermal stresses along with the mechanical stresses set up in the turbine blade during service condition may reduce the life of blade material. Figs. 5.2-5.5 represent the results obtained when static structural analysis was performed on IN 738 turbine blade material while Figs. 5.6-5.8 represent the results obtained when static structural analysis was performed on U500 turbine blade material.

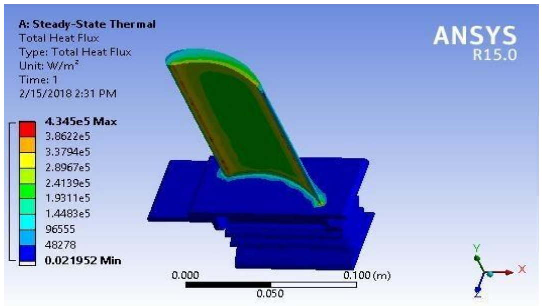

Fig 5.2: Temperature distribution on IN738 Turbine Blade

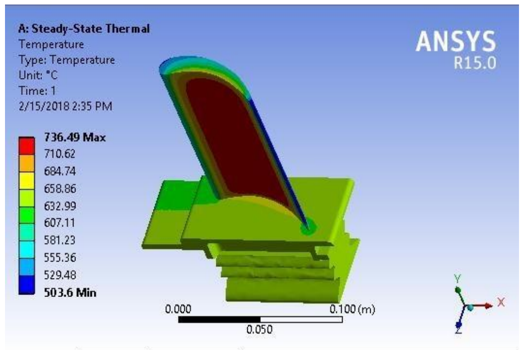

Fig 5.3: Heat Flux on IN 738 Turbine Blade

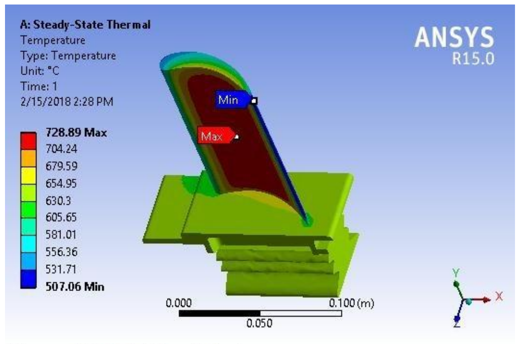

Fig 5.4: Temperature distribution on U500 Turbine Blade

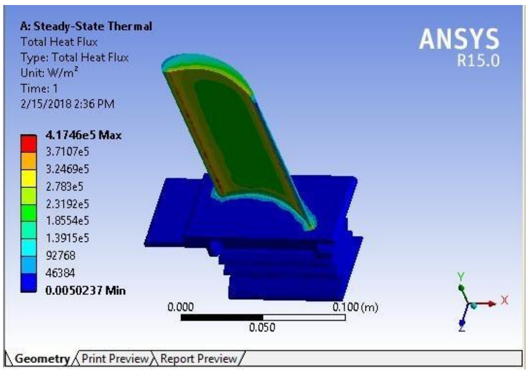

Fig 5.5: Heat Flux on U500 Turbine Blade Material

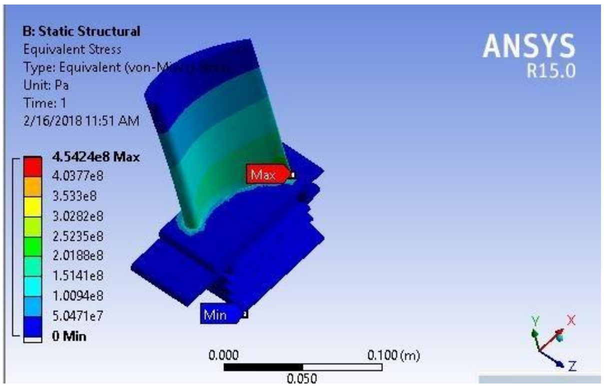

Fig 5.6: Von-mises Stress on IN738 Turbine Blade Material

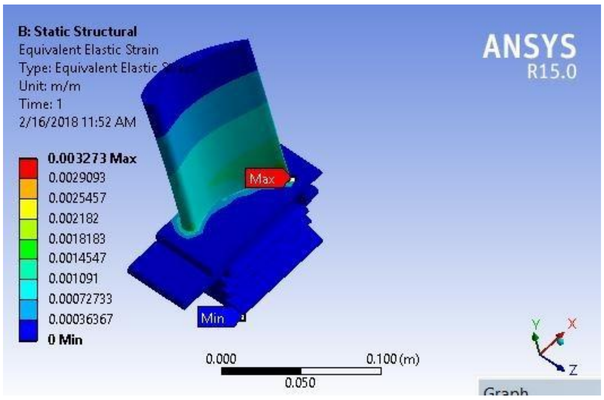

Fig 5.7: Elastic Strain on IN738 Turbine Blade Material

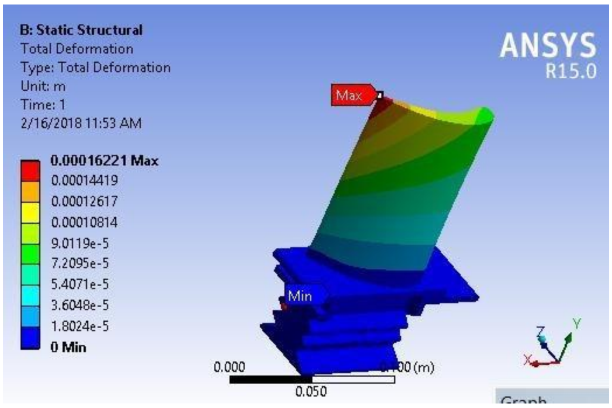

Fig 5.8: Material Total Deformation on IN738 Turbine Blade Material

CONCLUSION

In Turbine Blades are one of the most important components in the gas turbine engine. The blades are operated in harsh environmental condition at elevated temperature, high pressure and large centrifugal forces that hampers the performance and longevity of the blade material in service condition. The turbine blade material is exposed to unforeseen failure depending on the severity, and this necessitated the thermal and structural static analysis carried out in this study. From the analysis of the results, it was observed that the temperature on the turbine blades for both materials was below the melting temperature of the blade materials. Maximum temperatures were observed at the leading edge of the blade and decreased towards the trailing edge and blade root. Maximum von-mises stresses and strains were observed near the root of the turbine blade and upper surface along the blade roots. Total deformation obtained from each blade analysis were negligible, as 0.16221mm was obtained for IN 738 and 0.12125 mm obtained for U 500. This report serves as a guideline for the selection of suitable materials for minimal gas turbine blade failure and optimal working scenario.

REFERENCES

[1] Sulaiman, K.S., Rameshkumar, G.R., Vibration Diagnosis Approach for Industrial Gas Turbine and Failure Analysis, British Journal of Applied Science & Technology, 14(2), 1- 9, 2016.

[2] Kalapala P., Prasad, B.A., Anandarao, M., Material Optimization and Dynamic Approach for performance criteria in application to Gas Turbine Blade to overcome resonance; International Journal of Scientific & Engineering Research, 8(6), 189-196, 2017.

[3] Hassan M., Vibratory Analysis of Turbomachinery Blades, Rensselaer Polytechnic Institute, Hartford, Connecticut, 2008.

[4] Gharzwan M.A., High Cycle Fatigue Testing of Steel for Federal Aviation Administration (FAA), Department of Mechanical Engineering, California State University, Sacramento, (2017).

[5] Mohamad, B.A., Abdelhussien, A., Failure Analysis of Gas Turbine Blade Using Finite Element Analysis, International Journal of Mechanical Engineering and Technology (IJMET) 7(3), 299-305, 2016.

[6] Boyce, M.P., The Gas Turbine Handbook, Gulf Professional Publishing, Houston Texas, 2nd edition 2002.

[7] Ikpe A.E., Owunna I., Ebunilo, P.O., Ikpe, E., Material Selection for High Pressure (HP) Turbine Blade of Conventional Turbojet Engines, American Journal of Mechanical and Industrial Engineering 1(1), 1-9, 2016.