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Material Selection for High Pressure (HP) Turbine Blade of Conventional Turbojet Engines

Ikpe Aniekan Essienubong[1, *], Owunna Ikechukwu[1], Patrick. O. Ebunilo[2], Ememobong Ikpe[3]

[1]Department of Mechanical Engineering, Coventry University, West Midlands, UK

[2]Department of Mechanical Engineering, University of Benin, Benin City, Nigeria

[3]Department of Instrumentation and Control, Exxon Mobil Producing Nigeria, Akwa Ibom State, Nigeria

Email address:

ikpeaniekan@gmail.com (I. A. Essienubong), ikechukwu.owunna@uniben.edu (O. Ikechukwu),
Patrick.ebunilo@uniben.edu (Patrick. O. Ebunilo), ememobong.e.ikpe@gmail.com (E. Ikpe)

[*]Corresponding author

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Abstract: Turbojet engine can be divided into three major sections including the compressor, combustion chamber and the gas turbine section. The relatively high temperature gas that passes through the high pressure turbine stages of a turbojet engine from the combustion chamber has a direct effect on the performance and efficiency of the gas turbine, which may hamper its longevity in the long run, particularly the turbine blades. The turbine blades extract energy from the high temperature gas and transfer the kinetic energy of the flowing gas to the compressor stages where it provides forward thrust and rotates the turbine shaft which drives the high pressure and low Pressure compressor fan blades. However, the ability of materials to withstand this high temperature is based on properties of such materials which can be attributed to advances in material selection, improvement techniques in terms of surface protection and cooling as well as manufacturing processes which this paper is based on. Material indices were derived for High Pressure (HP) turbine blades to determine materials that can resist yielding and creep condition when exposed to high temperature above 700°C in a turbojet engine gas turbine. Based on the material indices derived, CES software 2014 was used to generate graphs showing materials with adequate fracture toughness, fatigue strength, stiffness and yield strength property that can withstand the in-service condition of HP turbine blade. Considering all these properties in terms of relatively high temperature, Nickel based super alloys dominated the graphs but in terms of density, titanium alloys dominated as CES software gave the minimum density of nickel alloy (8150 kg/m^3) as twice that of titanium alloy (4410 kg/m^3). Although both alloys are very expensive, nickel based alloy particularly Nickel-Cr-Co-Mo Super alloy also known as Rene 41 was chosen because of its excellent corrosion property and high strength at elevated temperature (About 1000°C) which makes it suitable for conventional HP turbine blade application.

Keywords: Temperature, Failure, HP Turbine Blades, Cyclic Stresses, High Strength, Low Density, Turbojet Engine

1. Introduction

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. The concept behind turbojet engines was first demonstrated in 1937 by Frank Whittle, an RAF cadet in early 1928 and this was the first flight four years later [15, 11]. The jet engine operates by sucking in some of the air while the aircraft is in motion, compressing it to the smallest volume, mixing it with fuel and burning it inside the combustion chamber and ejecting hot gases from the hot air-fuel mixture with extremely high force to propel the aircraft. The power capacity of such engines can be expressed in pounds of thrust, a term that describes the number of pounds the engine can travel [9, 12]. A typical jet engine is accommodated by a cowling, a detachable casing that opens outwards such as a rounded automobile hood to enable assessment and maintenance of internal components of the system. Coupled to each of the engines (typical 747 has four) is a pylon, a metal arm that connects the main engine to the wing of the aircraft. Through pumps and feed tubes in the pylons, fuel is channelled from the wing tanks to the aircraft engine and the hydraulic and electrical power developed by the engine is then circulated back to the aircraft via wires contained within the pylons [8, 26]. A fan installed in front of the engine helps accelerate air flow into the first engine compartment where the low pressure compressor is located. As air is routed by the fan into the engine a metal cylinder fitted with rectangular blades widens gradually from front to rare while subjecting the incoming air to increased pressure as shown in Figure 1.

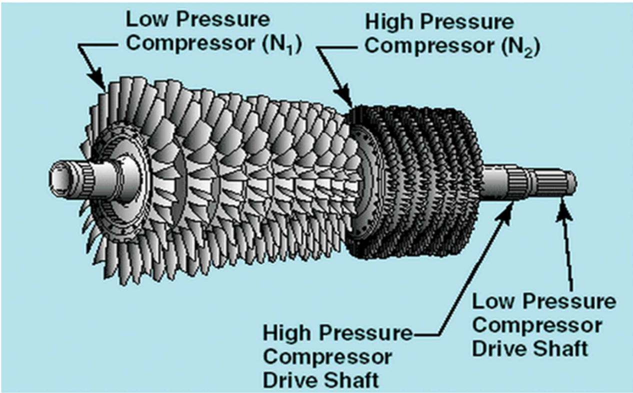

Figure 1. *Typical Compressor Blade of an Aircraft gas turbine engine [20].*

In other words, the compressor is fitted with low pressure and high pressure blades rotating alongside with the shaft like ceiling fan. Within the short space of time required for air to travel from the low pressure compressor area to the end of the high compressor area, air can possibly be compressed into a space 20 times smaller in size than the intake aperture [20, 21, 22]. Expanding as it exits the high compressor area, air enters the combustion chamber, an internal combustion engine cylinder in which air and fuel mixes together, ignites and burns. From the compressor, another high air stream barely passes through the centre of the combustion chamber without being combusted, while a third stream of air leaving the compressor is passed outside the combustor for cooling [5, 22]. During combustion and ignition of the air-fuel mixture in the combustion chamber, extremely hot volume of gases are produced, some of which exits the engine through the exhaust, while some quantity of the hot gases is routed into the turbine engine. In a typical gas turbine engine, a single turbine compartment is composed of a disc or hub with a set of turbine blades mounted on it [10, 13]. Also, the combustion chamber of a turbojet engine is annular, with an exit ring installed at the back to control exhaust gas leaving the system. The gases exiting the combustion chamber are released at a temperature of about 1700°C, while the shaft rotates at a speed in excess of 12,000 rpm [17].

In summary, air is compressed through the compressor stages of the gas turbine engine as it enters through the air inlet, temperature and pressure of the incoming air increases while the volume decreases during compressing. Temperature of the compressed air is further increased as it enters and expands in the combustion chamber (located between the turbine stages and compressor stages) where air-fuel mixture is combusted. The turbine stages extracts energy from the high temperature high pressure gases and transfers the kinetic energy of the flowing gas to the compressor stages where it provides forward thrust and rotates the turbine shaft which drives the HP and LP compressor fan blades as illustrated in Figure 2.

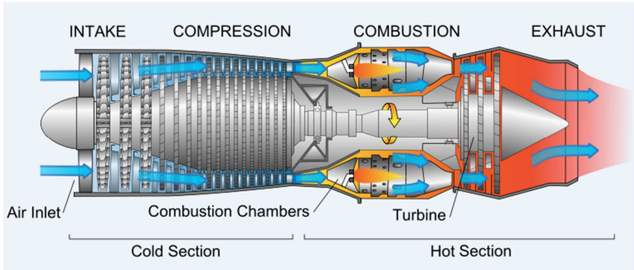

Figure 2. *Typical Gas Turbine Jet Engine [7].*

The turbine compartment is connected to the compressor compartment (either axial or centrifugal) through a shaft or spool. Turbine blades are one of the various components constituting the turbine compartment of a gas turbine system. The turbine blade functions by extracting as much energy as possible from the high temperature high pressure gases leaving the combustion chamber to power the compressor shaft, while in some systems, the turbine is also used to generate power for other components in the aircraft [5, 17]. As a result of the in-service condition of the turbine subjecting the turbine blades to intense heat, materials with relatively high strength at elevated temperature and designing of blades with labyrinthine airways and other methods such as thermal barrier coating, internal air channels, and boundary layer cooling has been adopted to ensure longevity of the turbine blade. However, fatigue failure is one of the factors that limit the longevity of the high pressure turbine blades resulting from high dynamic stresses due to vibration and resonance during operation cycle of the jet engine. To prevent the high dynamic stresses from damaging the blades, friction dampers are used in areas prone to this defect [4]. Turbine blades are exposed to very harsh operating condition inside the gas turbine such as high temperature, high stresses and high vibration effects which may potentially result in engine failure if the blades are not properly designed to resist these conditions. Moreover, the rotation cycle of turbine blades is over tens of thousands of Revolutions Per Mean (RPM) and that often subject the blade to stresses resulting from the centrifugal force and fluid forces that can result in creep, fracture or yielding failure [5]. The first stage of modern turbine engines located next to the combustion chamber operates at a temperature of about 2,500°F (1,370°C) while the early turbines operated at a temperature of about 1500°F (820°C) [13]. According to Dexclaux [8], modern military jet engines such as Snecma M 88 can operate up to a temperature level of about 2,900°F (1,590°C). In addition to the turbine engine operating temperatures, Rolls Royce [20] reported that the 5th stage of the low pressure turbine operates at a temperature of about 900°C, while the 1st stage intermediate pressure turbine operates at a temperature of about 1200°C and the 1st stage high pressure turbine operates at a temperature of about 1500°C respectively. This high temperature operation can weaken and limit the performance of the turbine blades as well as the turbine engine itself, thereby making them more prone to creep and corrosion failure, while vibrations and resonances in the turbine engine can result in fatigue failure [3]. The High Pressure (HP) turbine is exposed to the most intensed air pressure while the Low Pressure (LP) turbine is exposed to mild (cooler) lower air pressure. Difference in operating condition of the HP and LP turbine has resulted in the design of HP and LP turbine blades which are greatly different in material and cooling options despite the same thermodynamic and aerodynamic principles [10]. One of the major limitations in the early jet engines was the poor performance of materials used in the high temperature high pressure areas (combustion chamber and turbine) of the engine. Consequently, the need for better materials to eliminate these flaws necessitated further research in the field of alloys and manufacturing processes and that spurred researchers in this area of interest to unravel the challenges which lead to a long list of new materials and techniques that added more improvement to the modern turbine engines [6, 13, 17]. For example, development and adoption of super alloys in aircraft applications in 1940s and new manufacturing techniques such as vacuum induction melting (involves the application of electric currents in melting metals within a vacuum) in 1950s significantly improved the temperature performance of turbine blades. Unlike the early turbine blades, modern turbine blades are hollow with perforated holes at their leading edge as [14, 16, 18] shown in Figure 3.

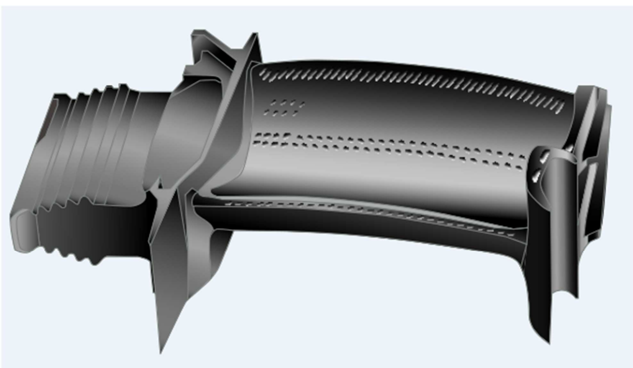

Figure 3. *Typical Modern Gas Turbine Jet Engine Blade Showing Perforations on the Leading Edge [24].*

When the turbojet engine is engaged, relatively cooled pressurised air is forced through the surface of the turbine blades and through the perforated holes, thereby, creating cooled layer of air to protect the blades from the hot gases. Moreover, hot gases from the combustor are accelerated prior to entering the turbine and this significantly lowers its maximum temperature from approximately 1800°C to approximately 1100°C while entering the first turbine stage [17, 22]. These challenges can be can be unravelled through the use of materials with relatively high melting point (temperature above the operating condition of the turbojet gas turbine engine), surface coating and cooling techniques which are the areas that will be addressed in this paper.

2. Material Characteristics Needed for the HP Turbine Blades

To withstand the aforementioned loading modes and operating conditions, the material must:

a. Be light weight so as to minimize both the power needed to drive the turbine. Therefore, total weight of the engine should have low density ρ.

2.1. Determination of Material Performance Indices

The following assumption has been made in determining the performance indices:

a. The blade is a cantilever fixed at one end and uniformly loaded as shown in Figure 4.
b. The cross sectional area of the blade is a free variable with a constant aspect ratio α

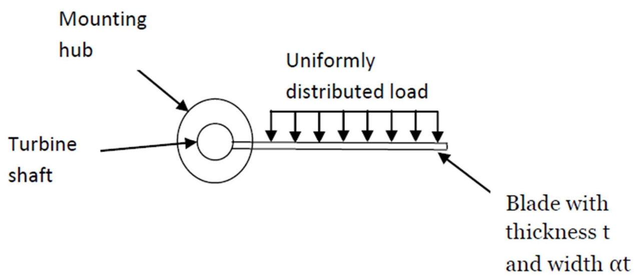

Figure 4. *HP compressor blade as a cantilever with uniform loading.*

2.2. Material Index to Resist Bending

From the generic beam bending equation

$$\frac{M}{I} = \frac{\sigma}{y} \quad (1)$$

Where:
σ is the stress at distance y from neutral axis of beam;
M is the bending moment of the blade;
y is the distance from the neutral axis;
and I is the second moment of area.
For a uniformly distributed load on a cantilever,

$$M = \frac{wl^2}{2} \quad (2)$$

Where
w is the uniformly distributed load in N/m;
l is the length of the blade in meter.

$$I = \frac{\alpha t^4}{12} \quad (3)$$

Where
t is the blade thickness;
α is the aspect ratio.
For stress (σ) = σ_{max}

$$y = \frac{t}{2} \tag{4}$$

From equation 4

$$\frac{I}{y} = \frac{\left[\frac{\alpha t^4}{12}\right]}{\left[\frac{t}{2}\right]} = \frac{\alpha t^3}{6} \tag{5}$$

Cross sectional Area,

$$A = \alpha t^2 \tag{6}$$

Substituting the value of t from equation 5 into 6 gives

$$\frac{I}{y} = \frac{\alpha}{6}\left(\sqrt{\frac{A}{\alpha}}\right)^3 = \frac{A^{\frac{3}{2}}}{6\sqrt{\alpha}} \tag{7}$$

From equation 1,

$$\frac{M}{\sigma_{max}} = \frac{wl^2}{2\sigma_{max}} = \frac{A^{\frac{3}{2}}}{6\sqrt{\alpha}} \tag{8}$$

$$A = \left(\frac{3\alpha^{\frac{1}{2}} wl^2}{\sigma_{max}}\right)^{\frac{2}{3}} \tag{9}$$

Mass of the blade, m_1 is given by

$$m_1 = Al\rho \tag{10}$$

Where
A is the cross-sectional area, and
$\rho$is the material density
Substituting equation 9 into 10,

$$m_1 = \left(\frac{3\alpha^{\frac{1}{2}} wl^2}{\sigma_{max}}\right)^{\frac{2}{3}} l\rho \tag{11}$$

For $\sigma_{max} = \sigma_y$ (yield strength)

$$m_1 = \left(3\alpha^{\frac{1}{2}} wl^2\right)^{\frac{2}{3}} l \frac{\rho}{(\sigma_y)^{\frac{2}{3}}} \tag{12}$$

Therefore, to minimize mass, the material index $\frac{(\sigma_y)^{\frac{2}{3}}}{\rho}$ should be maximized.

2.3. Material Index to Resist Fatigue

It is desired that the Fatigue strength endurance limit σ_e be as high as possible. Hence,

$$\frac{wl}{A} \leq \sigma_e \tag{13}$$

Where wl is the total load on the blade,

$$A \geq \frac{wl}{\sigma_e} \tag{14}$$

Mass, $m_2 = Al\rho$

$$m_2 = wl^2 \frac{\rho}{\sigma_e} \tag{15}$$

In order to optimize performance indices $m_2 = m_1$
To minimize mass, $\frac{\sigma_e}{\rho}$ should be maximized

2.4. Material Index to Maximize Fracture Toughness, K_{1c}

Assuming that the blade follows the equation of a centre cracked plate with a very large width [1].

$$\mathrm{K}_{1c} = \sigma(\pi c)^{0.5} \tag{16}$$

Where
K1c is the fracture toughness
σ is the applied stress and
c is a very small crack.

$$\mathrm{K}_{1c} = \frac{wl}{A}(\pi c)^{0.5} \tag{17}$$

$$A = \frac{wl}{\mathrm{K}_{1c}}(\pi c)^{0.5} \tag{18}$$

$$m_3 = Al\rho$$

$$m_3 = \frac{wl}{\mathrm{K}_{1c}}(\pi c)^{0.5} l\rho \tag{19}$$

$$m_3 = wl^2(\pi c)^{0.5}\left(\frac{\rho}{\mathrm{K}_{1c}}\right) \tag{20}$$

In order to optimize performance indices $m_3 = m_1$
To minimize mass, $\frac{\mathrm{K}_{1c}}{\rho}$ should be maximized.

2.5. Material Index to Maximize Specific Stiffness

The stiffness of a beam is given by Ashby [1].

$$S = \frac{F}{\Delta} \tag{21}$$

Where
S is the stiffness of the material
F is the load applied
Δ is the deflection.
In this case,

$$S = \frac{wl}{\left(\frac{wl^3}{8EI}\right)} = \frac{8EI}{l^2} \tag{22}$$

Where E is the Young's Modulus of the material
Substituting equation 3 into 22

$$S = \frac{8E\alpha t^4}{12l^2} \tag{23}$$

$$t = \left(\frac{12Sl^2}{8E\alpha}\right)^{\frac{1}{4}} \tag{24}$$

$$m_4 = Alp = \alpha t^2 l\rho \tag{25}$$

Substituting equation 24 into 25,

$$m_4 = \alpha\left(\frac{12Sl^2}{8E\alpha}\right)^{\frac{1}{2}} l\rho \tag{26}$$

$$m_4 = \sqrt{\alpha}\left(\frac{12S}{8}\right)^{\frac{1}{2}} l\left(\frac{\rho}{E^{\frac{1}{2}}}\right) \tag{27}$$

In order to optimize performance indices $m_4 = m_1$

Therefore to minimize mass, $\frac{E^{\frac{1}{2}}}{\rho}$ should be maximized.

2.6. Material Index to Maximize Natural Frequency

The natural frequency of a body is given by Ashby [1].

$$f = \frac{1}{2\pi}\sqrt{\left(\frac{K}{m}\right)} \tag{28}$$

Where K is the stiffness constant and is given by

$$K = \frac{AE}{l} \tag{29}$$

$$m = Al\rho \tag{30}$$

Substituting equation 29 and 30 into 28 yields

$$f = \frac{1}{2\pi}\sqrt{\left(\frac{AE}{Al^2\rho}\right)} \tag{31}$$

$$f = \frac{1}{2\pi}l\sqrt{\left(\frac{E}{\rho}\right)} \tag{32}$$

To maximize natural frequency, $\left(\frac{E}{\rho}\right)^{0.5}$ should be maximized.

The material indices to be maximized are

$\frac{E^{\frac{1}{2}}}{\rho}, \left(\frac{K_{1c}}{\rho}\right), \frac{\sigma_e}{\rho}, \frac{(\sigma_y)^{\frac{2}{3}}}{\rho}, \left(\frac{E}{\rho}\right)^{0.5}$ respectively

2.7. Selection of Material

Newer and more powerful aircraft engines are being developed in order to increase efficiencies. The air at the high pressure compression chamber is being compressed further resulting in increased temperature and pressure of the high pressure compression chamber. For engines such as the Rolls Royce Trent series and CMF's Leap X, the temperature at the end of the compression process can exceed 1300°C [21]. These temperatures are clearly above the service temperature capabilities of Titanium alloy which is about 600°C and Steel and its alloys which are about 450°C. Also, at relatively high temperature above 600°C, the protective oxide layer on the surface of Titanium alloy reacts with oxygen and carbon, thereby, causing it to be very hard and brittle in nature [19]. From the information gathered from relevant literatures, the in-service condition of turbojet engine gas turbine blades operates at relatively higher temperature than that of HP and LP compressor blades of the same engine. To meet the demands of these more powerful engines, materials which are capable of withstanding the harsh service environment are used. The desired material characteristics include;

a. The material should have high service temperature (700°C and above) in order to withstand the high temperature environment.
b. Low Coefficient of thermal expansion α in order to keep strain and thermal stresses as low as possible
c. High fracture toughness K1c to resist initiation and propagation of cracks.
d. Low density ρ so as to minimize both the power needed to drive the compressor and the total weight of the engine
e. Should have high fatigue strength σ_e in order to withstand many cycles of fluctuating loads
f. Be able to withstand tensile and bending stresses and hence should have high value of Young's Modulus and high yield strength σ_y
g. The natural frequency of the material should be high so as not to be excited into resonance.
h. The material should be erosion and corrosion resistant.

2.7.1. Material Index to Resist Bending

In order to resist bending at minimum mass, the material index $\frac{(\sigma_y)^{\frac{2}{3}}}{\rho}$ should be maximized.

2.7.2. Material Index to Resist Fatigue

To resist fatigue at minimum mass, the quantity $\frac{\sigma_e}{\rho}$ should be maximized.

2.7.3. Material Index to Maximize Fracture Toughness, K_{1c}

To maximize fracture toughness while keeping mass to its minimum, $\left(\frac{K_{1c}}{\rho}\right)$ should be maximized.

2.7.4. Material Index to Maximize Specific Stiffness

It was derived that to maximize specific stiffness, $\frac{E^{\frac{1}{2}}}{\rho}$ should be maximized.

2.7.5. Material Index to Maximize Natural Frequency

It was also derived that to maximize the natural frequency, $\left(\frac{E}{\rho}\right)^{0.5}$ should be maximized.

3. Methodology

CES Edupack software is a material selection tool that is made up of level 1, 2 and 3 and level 2 consist of more materials than level 1, while level 3 contains more materials than level 2. Based on the above material selection requirements and materials indices to be maximised, a search was carried out using CES software Level 3 to determine materials with the desired material indices. Fracture toughness is one of the most important factors that determine the longevity of HP turbine blade under such severe condition. Therefore, a minimum value of 30 MPa.m$^{0.5}$ was used to filter off materials with fracture toughness below this value as presented in Figure 5. Conventional aircraft applications require light weight materials to minimise CO_2 emissions and Fuel consumption. A graph of stiffness against young's modulus was plotted to determine a set of materials that has this properties as shown in Figure 6. Moreover, since the service temperature considered for HP turbine blades of a turbojet engine is about 700°C and above, a minimum temperature of 700°C was used as criteria to determine low cost materials in this temperature level as shown in Figure 7.

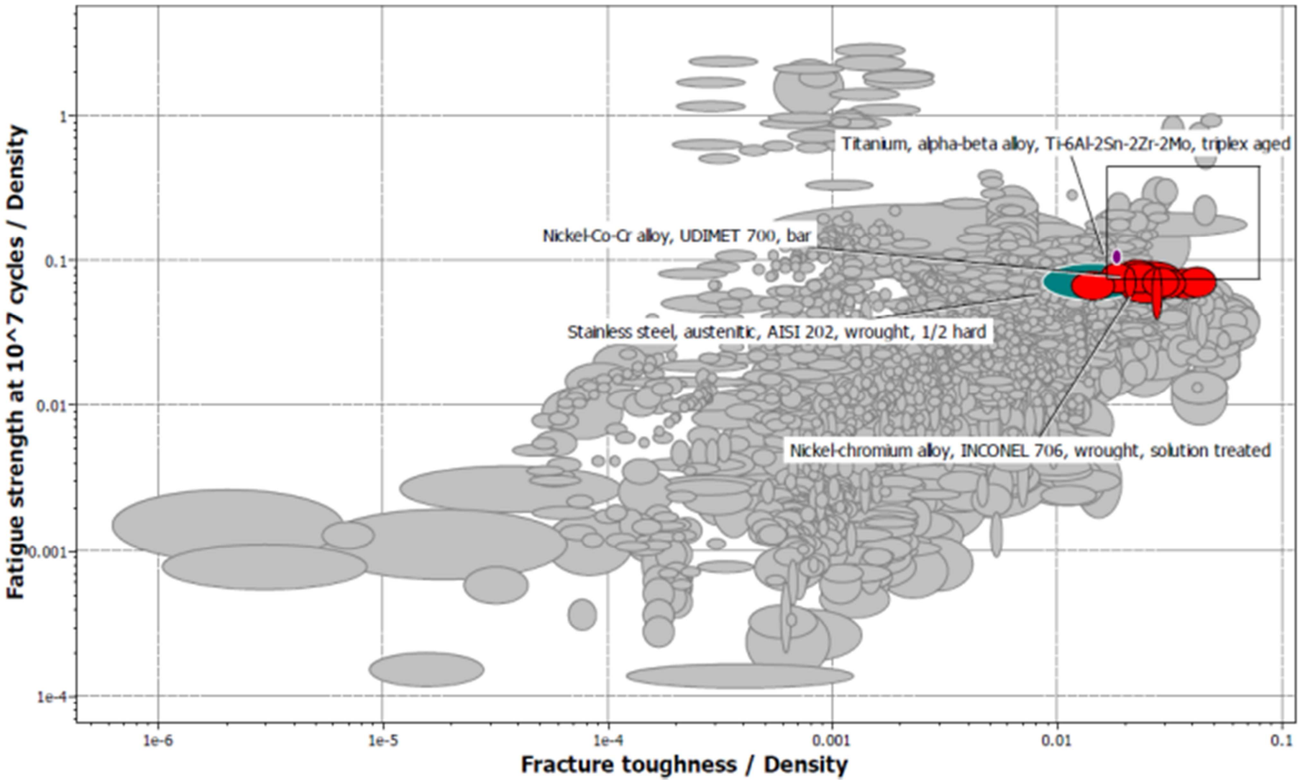

Figure 5. *Graph of fatigue strength* $\left(\frac{\sigma_e}{\rho}\right)$ *against fracture toughness* $\left(\frac{K_{1c}}{\rho}\right)$.

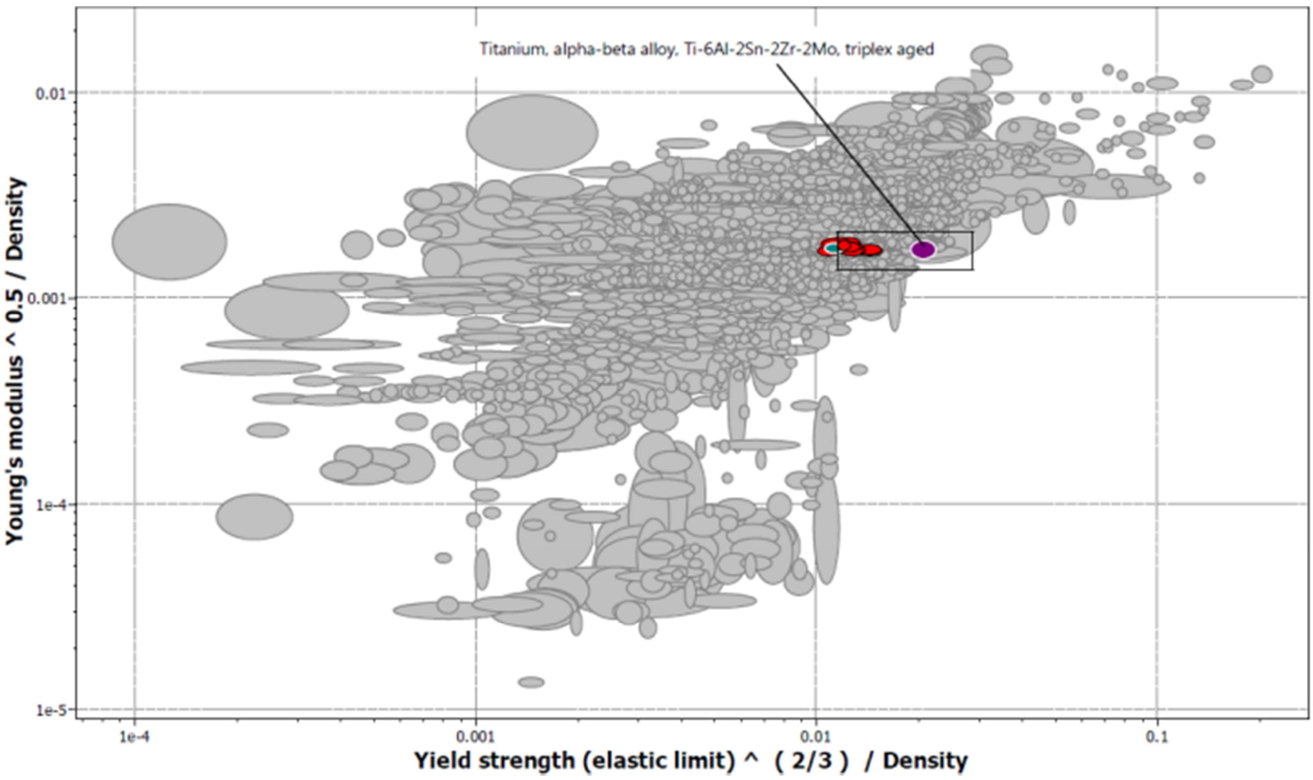

Figure 6. *Graph of Stiffness* $\left(\frac{E^{\frac{1}{2}}}{\rho}\right)$ *against yield strength* $\left(\frac{(\sigma_y)^{\frac{2}{3}}}{\rho}\right)$ *with respect to density.*

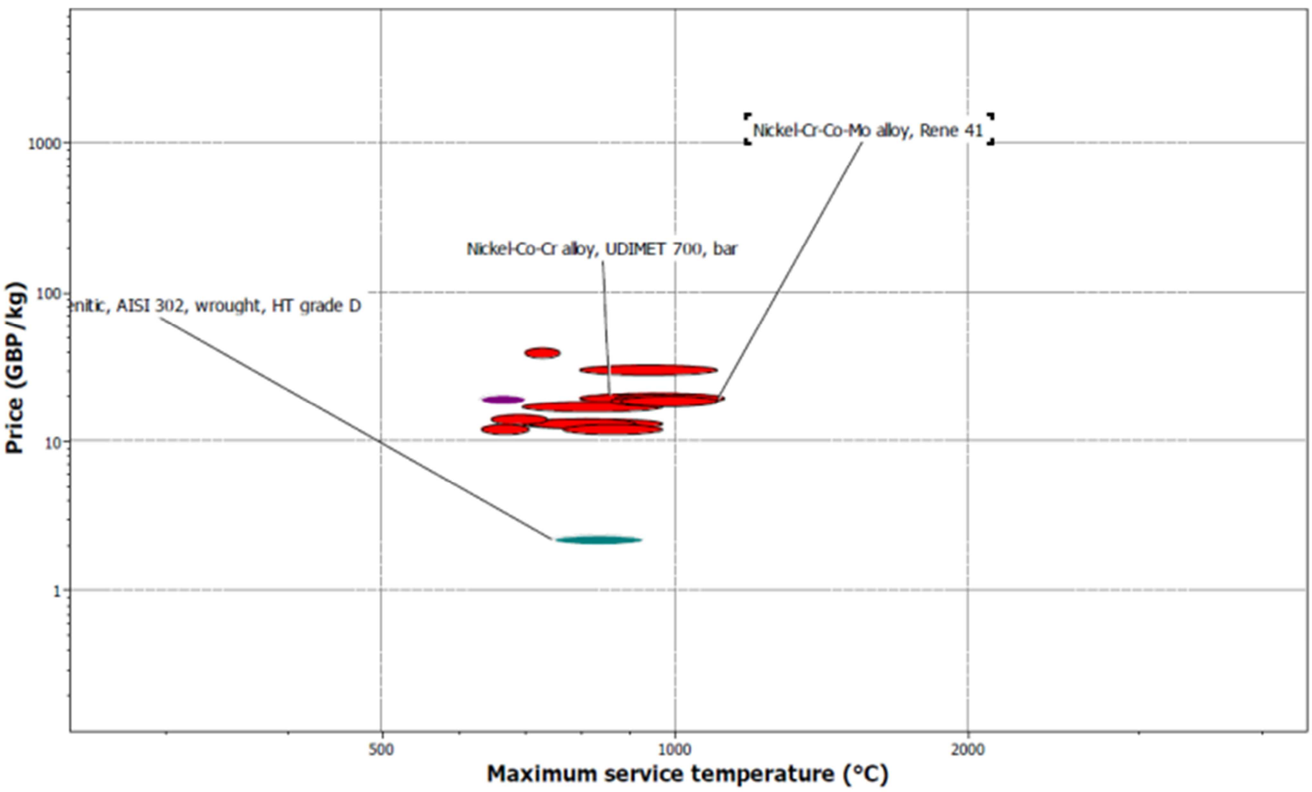

Figure 7. *Graph of price against Maximum service temperature.*

4. Result

Having carried out a search for materials that can suite the in-service condition of HP turbine blades, the following materials presented in Table 1 met the search criteria as potential materials that can be used for the aforementioned application. Each of the parameter plotted for in Figure 5, 6 and 7 was based on the high temperature and intense cyclic condition the HP turbine blades are exposed to, during operation.

Table 1. *Result of Potential Materials for HP Turbine Blades as Presented in Figure 5, 6 and 7.*

Figure 5	Figure 6	Figure 7
$\left(\frac{\sigma_e}{\rho}\right)$ Against $\left(\frac{K_{1c}}{\rho}\right)$	$\left(\frac{E^{\frac{1}{2}}}{\rho}\right)$ Against $\frac{(\sigma_y)^{\frac{2}{3}}}{\rho}$	Price Against Maximum Temperature
1. Nickel Based Alloy 2. Titanium Alloy 3. Stainless Steel	Titanium Alloy	Nickel Based Alloy Stainless Steel

5. Discussion

Figure 5 shows the result of materials obtained when specific fatigue strength is plotted against specific fracture toughness and Nickel alloys and titanium alloys were in that category while Figure 6 shows the results obtained when the specific stiffness is plotted against specific yield strength. From the Figure 6, it can be seen that a titanium alloy Ti-6Al-2Sn-2Zr-2Mo stood out because of its low density. From Figure 7, the graph of price against service temperature, the superiority of Nickel-based super alloys could be seen, and of the Nickel based super alloys that passed the selection criteria, Nickel-Cr-Co-Mo Super alloy also known as Rene 41 was chosen as the desired material for HP turbine blades, as it gave maximum working temperature at a relatively moderate cost when compared to the cost of other super alloys. From the level 3 of 2014 CES software, the operating temperature of Rene 41 is about 1000°C. According to Nathan [17], the hot gases exiting the combustion chamber immediately after combustion are at about 1700°C. This implies that the HP turbine blade is exposed to operating environment with several hundreds of degrees hotter than the melting temperature of Nickel alloy. At such condition, the material is usually subjected to cooling to prevent it from melting. This is achieved through two means such as; coating the blades with a low conductivity ceramics and designing the blades with complex internal channels through which air entering the root of the blade passes through the internal cooling channels and exits from a myriad of perforated holes on the blade surface in order to form a stream of cool air around the HP turbine blade to ensure the blade does not operate above its melting point. However, the cooling air is not entirely cooled (600°C-650°C) but it is necessary to get it extracted from the hot area of the combustor to give it the required pressure to get through the internal channels of the HP turbine blades and out of the perforated holes on the surface, thereby, keeping the blades temperature below the melting point of Nickel alloy in real case scenario. Metals are made up of several thousands of crystals [2]. Arranged structures of atoms positioned in regular lattice which forms naturally during the cooling process of metals in the molten state. The size particles of these crystals exist in the order of tens of microns located at several inclinations. At extreme temperature and under the influence of stresses and strains, the crystals can potentially slide against one another and impurities can possibly move towards the grain boundaries. This phenomenon is known as creep, a failure mode that greatly affected the early turbine blades which were manufactured from steel through forging. To reduce the effect of creep, super alloys with solid solution strengthening, grain boundary strengthening as well as thermal barrier coatings are applied in HP turbine blade design and manufacturing process. The application of protective coatings which are usually stabilized zirconium dioxide-based ceramic is to ensure the reduction of thermal damage and to control oxidation. The use of thermal protective coating limits the high temperature subjection of Nickel super alloy and this in turn reduces the creep mechanism the blade is exposed to. However, oxidation coatings minimises efficiency losses due to build-up of heat on the external part of the blade. Apart from using protective coatings, some manufacturing techniques such as hot isostatic pressing, a process used for reducing the porosity of metals can be used to improve alloys used for manufacturing HP turbine blade as well as its performance. Furthermore, the evolution of Directional Solidification (DS) and Single Crystal (SC) production techniques has played a vital role in increasing the strength of HP turbine blade alloys against the effects of creep and fatigue as DS functions by orienting grain boundaries in one direction while SC eradicates the grain boundaries [13]. Another method of improving turbine blades is through cooling which can be achieved through air or liquid. Although liquid cooling seems more attractive due to high specific heat capacity and partials of evaporative cooling, liquid cooling has some disadvantages such as corrosion, leakage, choking etc. whereas, air cooling allows the flow of air into hot areas in the turbine blades without any problem. At the early stage of jet engines, the prototypes of Frank Whittles were manufactured entirely from steel which best known for strength and surface hardness [17], but its unsuitability necessitated the search materials that can withstand high temperature and jet engine manufacturers turned to Nickel alloy which does not only exhibit high resistance against corrosion and cyclic stresses, but has an outstanding characteristics referred to as gamma prime in which Nickel combines with aluminium to maintain its strength at extreme temperature conditions. Nickel alloy is composed of austenitic-face-centred cubic (fcc) matrix phase (γ) (with each cube comprising a face with five atoms, one at the centre and one at each corner) and a variety of other phases such as gamma prime (γ'), M_6C, carbides MC and $M_{23}C_6$, of which the γ' heightens and sustains the increased strength at extreme temperature condition. The atoms present in Nickel alloys can swap in and out of the fcc lattice, but under the right composition of aluminium and Nickel, Nickel migrates to the centre of the faces and aluminium at the corners. This is called precipitate, in which half a micron (in size) of alloys are closely packed together and under this condition, a single atom of dislocation cannot easily slip or move through it, therefore the cuboids of γ' in the matrix pins every possible dislocation movement in position and causing difficulty in the metal deformation when exposed to high temperature or cyclic stresses caused by vibration of the rotating HP turbine jet. During manufacturing, Molybdenum is added to the γ' phase of Ni base alloys as strengtheners at both room and elevated temperatures [25]. Aluminium improves the protective surface oxide film, while addition of niobium, titanium and carbide formers is used to stabilize the alloys against the effect of chromium-carbide sensitization. Addition of small quantity of chromium (8%) to nickel impacts resistance to heat by creating a surface film of Cr_2O_3 as well as increasing the nickel chromium alloy sensitivity to oxidation and corrosion resistance rate. Although Rene 41 super alloy (alongside other nickel based super alloys) seems to be the conventional material for HP turbine blades, its density which is in the range of 8150-8350 kg/m^3, (about twice the density of Titanium alloys) is a major drawback for HP turbine blades. Also, the cost of super alloy and its processing is relatively high [23]. Moreover, other materials such as Ceramics and Ceramic Matrix Composites, Polymers and Polymer matrix composites etc. have been proven suitable for the manufacturing of HP turbine blades, their application in this area is very minimal and still undergoing development. Finally, the last stages of the turbine rotor blades and sometimes the stator blades (exposed to minimal or no stresses) of a jet engine are less exposed to creep effects and are not required to be manufactured with Nickel based super alloys.

6. Conclusion

This paper review focused on material selection processes for conventional material for the HP turbine blade of a turbojet engine. The primary aim was to determine materials that can withstand the service temperature of high pressure turbine blade of a turbojet engine which for decades subjected HP turbine blades of early turbojet engines to creep and cyclic stresses due to centrifugal forces acting on the blade component. The choice of material chosen satisfied the operating condition of HP turbine blades in terms of high strength at relatively high temperature.

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