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TECHNICAL ARTICLE—PEER-REVIEWED

# Life Consumption Assessment of a Large Jet Engine

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**Abstract** This paper deals with the performance study and life consumption assessment of a long range civil aircraft engine with respect to take-off variables like airport altitude, ambient temperature, thrust levels, and flight duration. The component selected for lifing analysis is the first stage high pressure turbine blade/disk of a 430 kN thrust engine. Performance tools like TURBOMATCH and HERMES are validated against the published data, with deviation less than 4%. Based on failure modes like low cycle fatigue, creep, and oxidation, the integrated lifing tool is used to estimate the severity indices at various off-design conditions. It is observed that creep is the main criteria in blade failure due to extreme high temperature gas coming from combustor, whereas for disk it is fatigue due to repeated stresses with varying shaft speeds. With increase in airport altitude, the blade and disk severity increases with respect to that at sea level conditions. As the ambient temperature goes up, blade severity also increases as well but the change in disk severity is not as much due to less change in spool speed. Operating with reduced thrust, both the blade and disk severity decreases with deratings. As flight duration increases with reference to a standard mission, both the component damages go up but the blade severity is more pronounced than the disk due to exposure to high temperature for a long time. As a whole, the total damage severity of disk and blade assembly increases with rise in altitude, outside air temperature, and mission duration and decreases with engine deratings.

**Keywords** Thermal fatigue · Low cycle fatigue · Hot corrosion · Oxidation

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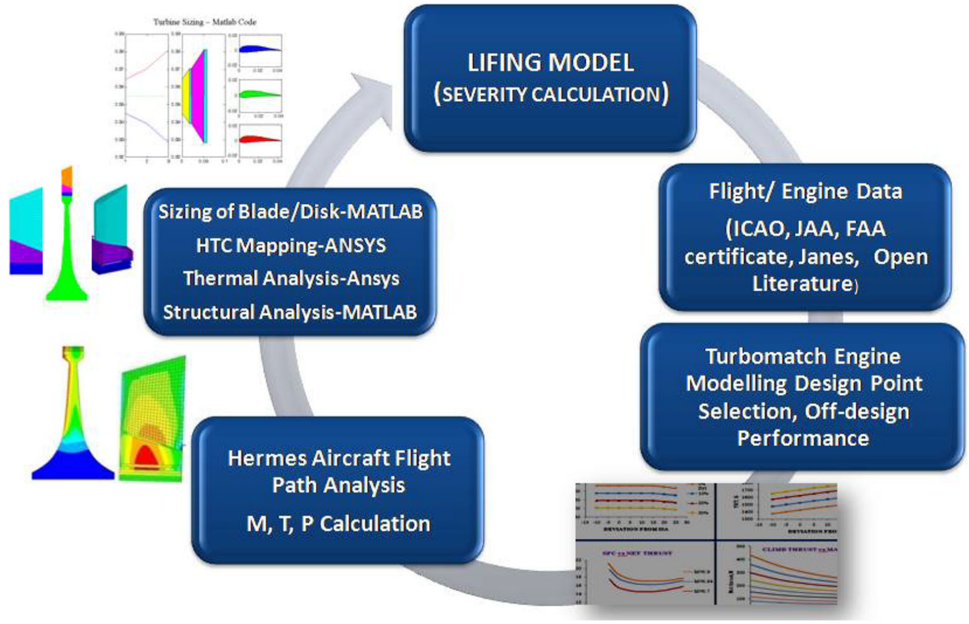
## Introduction

Engine operating and maintenance cost is a major part of airline expenses and any reduction will benefit the passengers directly. Take-off is one of the critical segments of a flight profile where the aircraft with maximum takeoff weight (MTOW) has to be accelerated in a limited length of runway. The type of aircraft and airport from which it can be operated economically is decided by the thrust rating of the engine. For long range flights, the fuel load as well as the payload increases demanding higher take-off thrust of the engine. Therefore, there are always trade-offs between available thrust, runway length, MTOW, and range to be covered. To meet the high demand for thrust, the engine has to operate at maximum loading condition, i.e., at maximum pressure and temperature. This leads to the deterioration of hot end components resulting in premature withdrawals for maintenance, repair, or even replacement. To overcome this situation, the concept of de-rating has been developed to minimize the maintenance cost. De-rating the power setting of the engine has significant consequences on operating cost of the airliner and its effect on engine performance and life consumption patterns has been studied in this paper.

This study uses an integrated lifing tool as shown in Fig. 1 that represents the relationship between the operational usage of the aero engine and component condition [1]. With the use of simulation packages, the model is able to determine the condition of engine components and life consumption analysis

**Fig. 1** An integrated lifing tool




with reference to some standard case in various off-design situations. The total damage severity of disks and blades in the high pressure (HP) rotor assembly is estimated and found to increase with rise in altitude, outside air temperature, and mission duration and to decrease with engine de-rating.

## Methodology

For this study a high bypass turbofan engine used in a long range civil aircraft has been selected. Engine performance data like maximum take-off thrust, exhaust gas temperature (EGT), fuel flow, engine pressure ratio (EPR), bypass ratio, speed, etc. are collected from various test certificates of ICAO, FAA, JAA. With the help of simulation tool TURBOMATCH, a performance model is created with suitable design point selection and run for various off-design situations to get engine ratings and parameters at various locations. Similarly an aircraft model is selected that suits the above engine configuration and then payload and range is calculated with aircraft flight path analysis tool HERMES and verified with the actual payload-range diagram values. Analysis is carried out for different take-off conditions. Parameters obtained at various locations of the engine model are used as the inputs to the lifing model [2]. The lifing model is based on various modes of failure of gas turbine components such as creep, low cycle fatigue, oxidation, etc. and calculates the cycle or hours to failure depending upon the mode of failure for different flight segments starting from take-off to landing. All segment failure/damage index values are added together to find the severity index for each component such as HP turbine disc or blades. Assuming the reference case severity index as 100%, all other case severity index values are calculated and compared.

The HP turbine disc and blades are considered here for analysis. The HP turbine operates in very harsh conditions downstream of the combustion chamber, where flame temperature in excess of 2200 K prevails. Several cooling channels are provided to keep the temperature of blade and disk under acceptable limits.

### Engine Model

From test certificates and various treatises, engine parameters such as mass flow, bypass ratio, fuel flow, HP and LP spool rotational speeds, fuel flow, etc. are collected and a TURBOMATCH engine model is created [3]. It is capable of running steady state, off-design, and transient simulations for aero engine configuration. A precise gas turbine configuration similar to the high bypass turbofan engine is created with its various components. With the basic assumption of one dimensional flow modeling, several simulations are run to match the data of maximum gross thrust, fuel flow, specific fuel consumption (SFC), and EGT with that generated by TURBOMATCH within an acceptable deviation for selection of a design point. Once the design point is fixed, various off-design points are run to check the stability of the model and subsequently engine ratings are established.

### Aircraft Model

An aircraft geometric configuration is obtained from Jane's All The World's Aircraft [4]. With the help of engine design point parameters, aircraft flight path analysis is carried out using simulation software HERMES. The stability of the aircraft model is determined comparing the generated payload-range diagram with the original one. A variety of off-design simulations related to take-off conditions, such as change in ambient temperature, runway height above sea level, runway length, and flight duration are run to get the gas parameters such as mass flow, temperature and pressure, fuel flow/fuel–air ratio, etc. at the required locations.

### Sizing Model

Knowing the compressor and turbine characteristics from TURBOMATCH, basic geometry of HP turbine disk and blade and number of blades are estimated [5]. Various assumptions like free vortex flow, constant angular velocity, choked nozzle for maximum mass flow, constant mean diameter, 50% reaction at blade mid height, straight sided annulus wall, etc. are made to do a preliminary design of a turbine. Out of various blade profiles used for aero engines, NACA A3K7 is selected for the turbine blade where as for the disk shape is assumed to be hyperbolic. Rene40 is taken as blade material and Inco 718 for the disk.

### Heat Transfer Mapping

The heat transfer effects from the hot gas stream to the blade are estimated using suitable correlations for Reynolds and Nusselt numbers [6]. The shape of blade leading edge is modeled as a circular cylinder and the trailing edge as flat plate. Cooling effectiveness with film cooling on blade profile, thermal barrier coating, and combustor pattern factors have been taken into account for the above calculation.

### Lifing Model

The lifing model is based on various failure mechanisms such as creep (Larson–Millar approach), low cycle fatigue (Basquin/Coffin–Manson relation), and oxidation (Arrhenius equation) [7–10]. Based on these approaches, the Palmgren–Miner law is used to calculate the time/cycles to failure, not considering the way failure is reached. These models are representative for the safe life philosophy, aiming to retire a component before a crack originates.

The steady state damage in terms of hours and fatigue in terms of cycles are obtained for various locations from the lifing model. All the results are converted to nondimensional numbers with appropriate calculation. For a certain case the maximum amount of damage for a single component is taken as the reference (i.e., 100%) and accordingly other case damages are calculated, as severity index. For various off-design conditions, these damage indices are compared and analyzed to draw some conclusions.

**Table 1** Engine specification at take-off

| Parameters | |
|---|---|
| Max take-off thrust | 430 kN |
| Overall pressure ratio | 40.5 |
| Bypass ratio | 8.3 |
| Air flow rate | 1480 kg/s |
| HP spool rpm | 9500 |
| LP spool rpm | 2300 |

## Engine Design Point and TURBOMATCH Modeling

The engine specifications at take-off conditions are presented in Table 1. This has been considered as design point simulation in TURBOMATCH. Various bleed location as shown in Fig. 2 are included in the simulation for better accuracy of the model and also for the stability for different off-design conditions.

### Off-Design Performance

With the engine geometry fixed by the design point calculation, the performance at any off-design condition can be estimated from the component characteristics and engine performance maps [11–13]. When calculating the off-design performance, it is very important to select an engine parameter as a key. This key will determine the matching conditions of the engine. Suitable parameters can be the rotational speed, turbine entry temperature (TET), EGT, etc. As engines are designed for a limiting value of TET, it is used as a key in this exercise. The compressor characteristics are generated using TURBOMATCH. ISA sea level ambient conditions are taken as the reference state. Effect of ambient temperature deviated from IAS condition on thrust and TET is shown in Figs. 3 and 4.

The compressor map for the last three stages of the HP compressor (HPC) is shown in Fig. 3. Varying the combustor outlet temperature (COT) or TET from 1000 to 1775 K, various operating points are located on speed lines. As seen in Fig. 3, the fan operates at a wide range of mass flow to accommodate the smooth engine operation at various off-design conditions, where as HPC operates with minimal operating range.

A similar trend is observed with the HPC, the running line becomes steep and approaches the surge line. Assuming altitude and shaft speed constant, the variation of net thrust and SFC with flight speed is influenced by momentum drag, ram compression, and ram temperature rise. At low flight

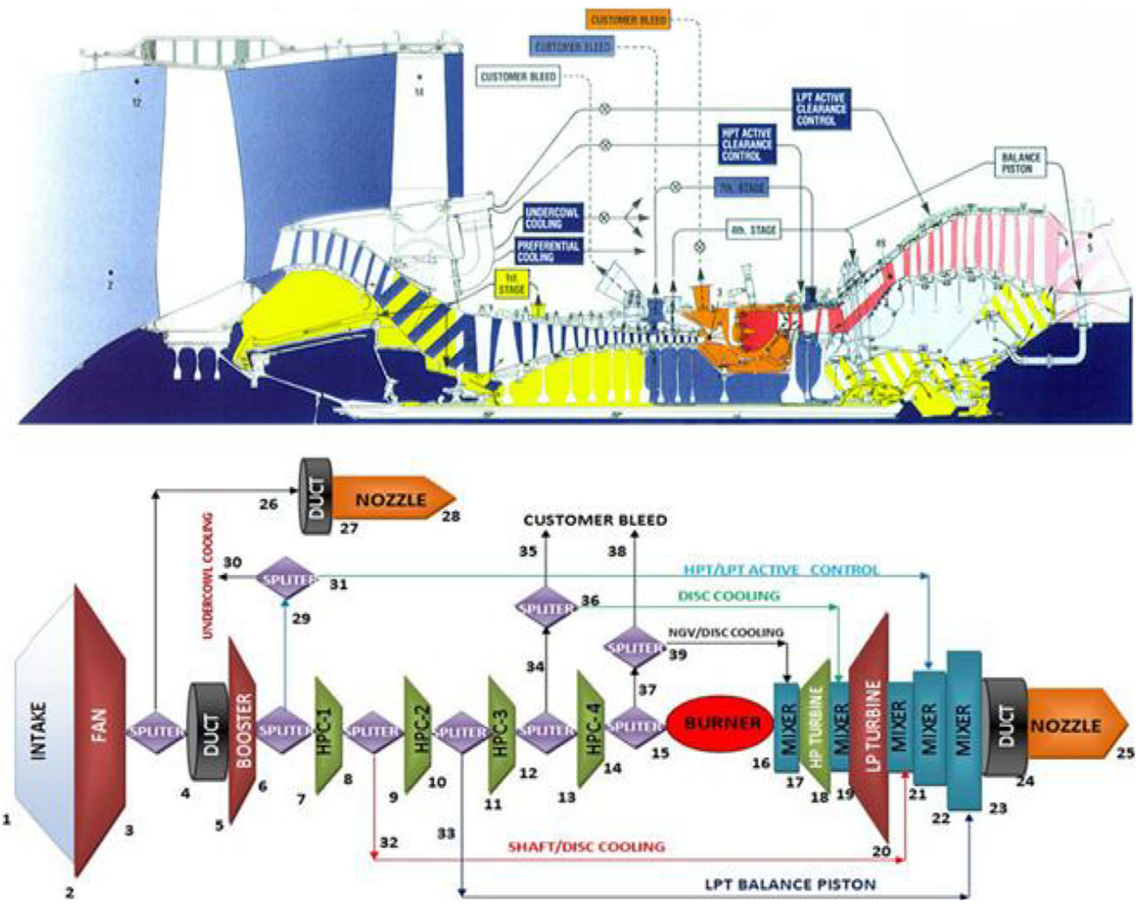



**Fig. 2** Engine mass flow diagram and TURBOMATCH model

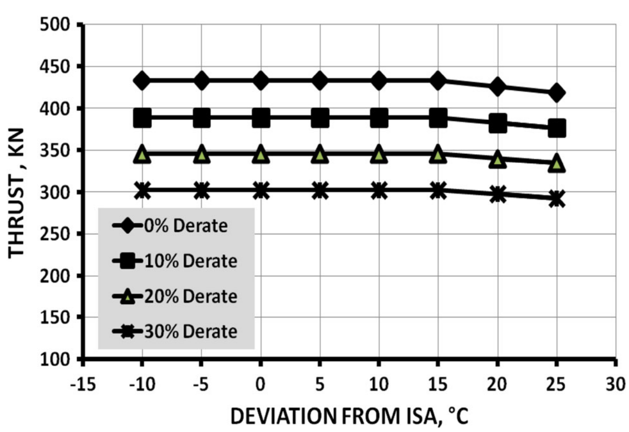



**Fig. 3** Effect of ambient temperature and derating on the Flat rating thrust of the engine

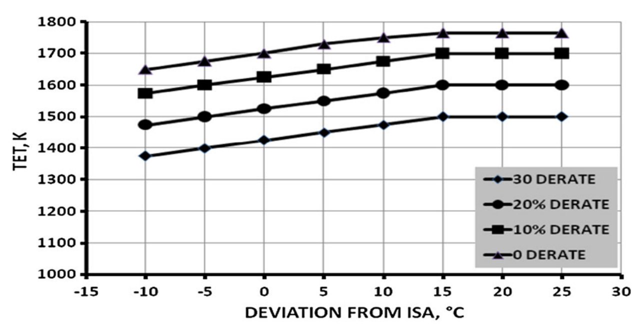



**Fig. 4** Effect of ambient temperature and derating on TET of the engine

velocity ($M < 0.3$), momentum drag predominates and thrust falls because the ram compression and temperature rise is very small. In the case of high bypass turbofan, the momentum drag effect is more pronounced due to lower jet velocity. From Fig. 5 it is seen that at $M = 0$ to $M = 0.2$, the compressor running lines are parallel to each other with pressure ratio being constant. As Mach number increases ram compression starts to influence, increasing mass flow ratio and nozzle pressure ratio, and the non-dimensional mass flow decreases for a fixed value of TET and the running line tends toward the surge line. The effect of flight Mach number on net thrust and fuel flow at various altitudes is shown in Figs. 6 and 7 where TET is kept constant. For a speed of $M = 0.8$ to $M = 0.85$, the engine thrust remains constant for a cruising altitude of 10–13 km and as the fuel flow increases with flight speed, 0.8 Mach is the ideal aircraft speed for cruising.

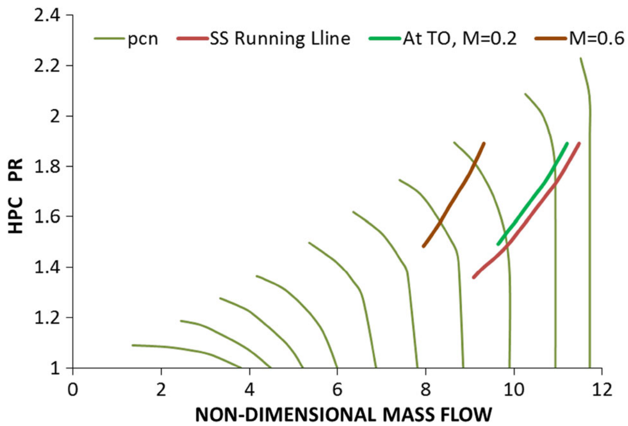



**Fig. 5** Last stage HPC characteristics and running lines at sea level condition

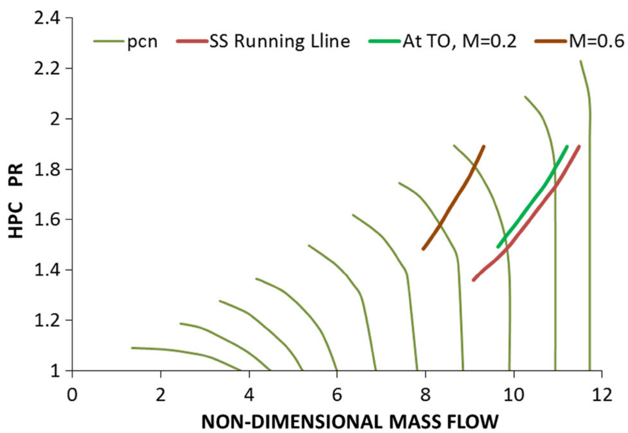



**Fig. 6** Effect of flight Mach number and altitude on net thrust

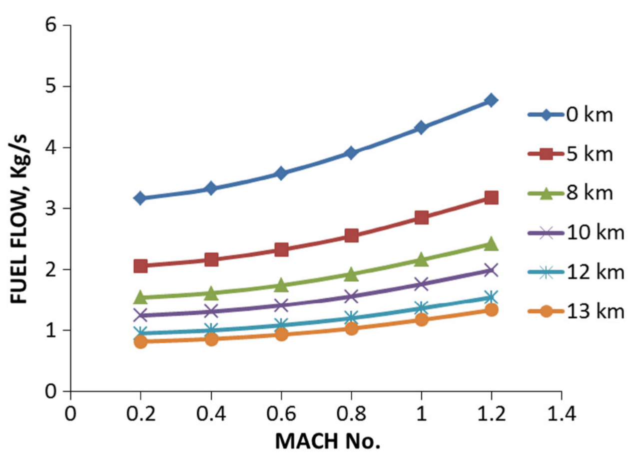



**Fig. 7** Effect of flight Mach number and altitude on engine fuel flow

## Aircraft Performance Model Structure

All the aircraft flight path simulations are carried out using simulation model HERMES [14]. The model predicts aerodynamic characteristics of a given aircraft and calculates the overall performance of the aircraft/engine combination to monitor various performance parameters during the flight, i.e., take-off, climb, cruise, descent, and landing, etc.

The HEREMS model consists of six different modules such as input data, mission profile module, atmospheric module, engine data, aerodynamic module, and aircraft performance module. The modules were developed separately and then integrated into the main model that computes the aerodynamic characteristics of the aircraft, fuel, distance and time for the climb, cruise, hold and descent segments of the mission. Allowances were made for the remaining segments that include startup, taxi, take-off, landing, and taxi-out. The module also computes the take-off distance. The computation occurs sequentially using publicly available data on the general configuration of the aircraft, MTOW, payload-range diagram, and/or mission profile.

For a given range the computation of the cruise fuel and distance can only be performed if the time from the top of climb (TOC) to the top of descent (TOD) is known or vice versa. At the start of the calculation neither the distance nor the time are known due to an uncertainty of the climb and descent distance/time. For this reason an estimate, based on flying Mach number and assuming that the distance from TOC to the TOD is the distance between the departure and the destination airport, is used to initialize the solution. The final result is compared to the specified mission and an iterative procedure is performed to determine the correct time spend at cruise.

In this program MTOW, payload and fuel load, etc. are the inputs. Although the range of the aircraft is also specified this is an initial trial value used to increase the accuracy of the model and to initialize the solution. In practice, the range is an unknown that is computed by an iterative procedure. Each iterative cycle determines the fuel required for a trial distance. Convergence is achieved when the total fuel load is consumed. In the model this is attained by subtracting the calculated total fuel (mission fuel plus reserves) and the payload from the MTOW, to give a trial OEW value. This value is then compared to the actual OEW of the aircraft and the difference is used to adjust the time spent at cruise and the distance covered, i.e., range. The computation is repeated until convergence within 0.1% of the OEW is achieved.

### Aircraft Type

For the purpose of study, one baseline aircraft suitable to the representative high bypass turbofan engine was selected and modeled. The aircraft is capable of flying from medium range to long range flights. The input data such as shape, geometry, load capacity, etc. are collected from

**Table 2** Aircraft specification

| Parameters | |
|---|---|
| MTOW | 300,000 (kg) |
| Max range | 14,500 (km) |
| No of engines | 2 |
| Max payload | 68,000 (kg) |
| Max fuel load | 177,000 (kg) |
| Wing area | 428 ($m^2$) |
| Cruising altitude | 10,668 (m) |
| Cruising Mach number | 0.84 |

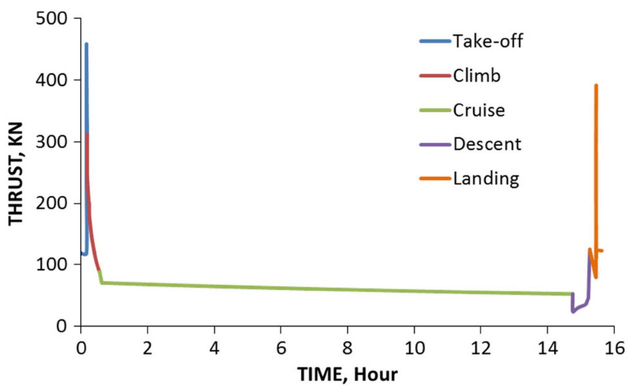



**Fig. 8** Thrust variation throughout whole Flight Profile

public domain for the purpose of simulation. The specification of the selected aircraft is given in Table 2.

## Flight Path Analysis Results and Discussion

### *Thrust Variation in Different Segments of Flight*

The thrust is a function of atmospheric properties like temperature, pressure, density of air, engine parameters like bypass ratio, bleed percentage, combustion temperature and aircraft geometry, speed, and payload. Its value varies from maximum during take-off to minimum in landing. The simulation results for a 14,000 km, 16 h flight are shown in Fig. 8. During taxi-out, the engine starts with minimum sustainable COT of 1050 K and goes to maximum take-off thrust with COT = 1785 K. As the aircraft starts rolling during take-off, the thrust decreases due to inlet momentum drag. The speed increases during climb and the momentum drag effect further reduces the net thrust from 312 kN at initiation of climb (i.e., 500 m above ground) to 90 kN at the TOC 11,500 m cruise altitude with cruising speed of 0.84 Mach.

During cruise, aircraft speed is kept constant at 0.84 Mach, so the momentum drag is constant throughout this phase. As the fuel mass is consumed with respect to length of flight, the aircraft weight reduces and the thrust requirement also decreases. This effect is clearly visible in Fig. 8 where the cruise thrust decreases from 90 to 60 kN at the end of this segment. For descent to lower altitude, the pilot reduces the thrust to minimum level so that the aircraft comes down with gravity. To keep the airplane at a particular descent altitude, the thrust level has to increase to balance the aircraft's weight as per stability equation. Then the landing and approach phase starts with a decrease in thrust and then on ground it again increases for thrust reversing as a braking action.

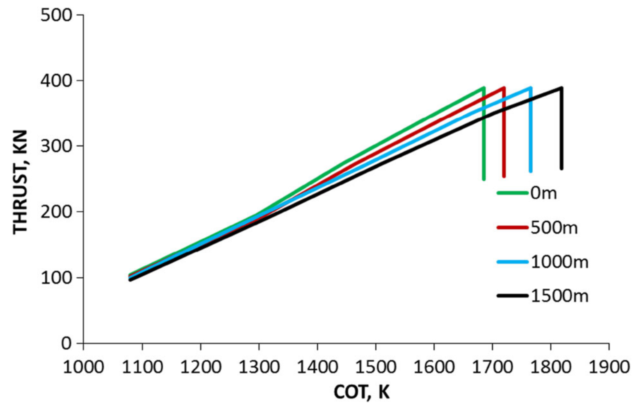



**Fig. 9** COT variation with airport altitude for same take-off thrust

### *Take-Off Thrust with Respect to Airport Altitude*

The ambient condition and runway altitude has a great impact on thrust generation. If COT is maintained constant in all airports from zero altitude to 1500 m altitude, the net thrust produced by the engine decreases due to lower density of incoming air. It can be observed that there is a thrust reduction of 50 kN, operating between airports of sea level to 1500 m altitude, with constant value of COT = 1670 K. But for various reasons such as clearing obstacles or carrying full payload, etc. the take-off thrust is always maintained constant with increase in fuel flow, i.e., COT. This has a severe impact on gas turbine hot end components, increasing the thermal fatigue. Considering 10% derate, i.e., the take-off thrust requirement is 387 kN/engine, there will be an increase in COT from 1685 to 1818 K (130 K in 1500 m) as shown in Fig. 9. The amount of fuel also increases from 357 kg at sea level to 390 kg at 1500 m height.

### *Take-Off Thrust Variation with Outside Air Temperature (Oat)*

With increase in air temperature air density decreases and compression becomes more and more difficult. As a result the EPR falls, as does the thrust. To maintain the same

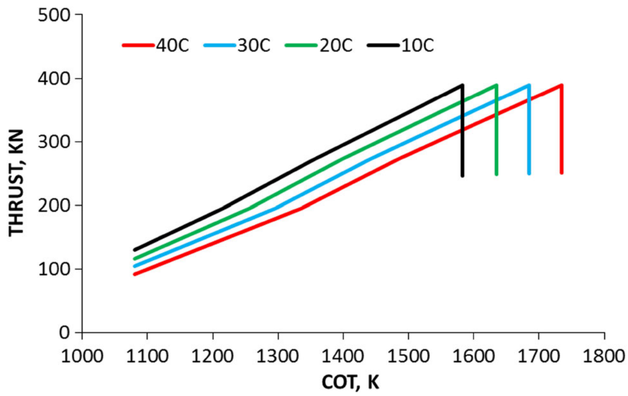



**Fig. 10** COT variation with OAT for same TO thrust

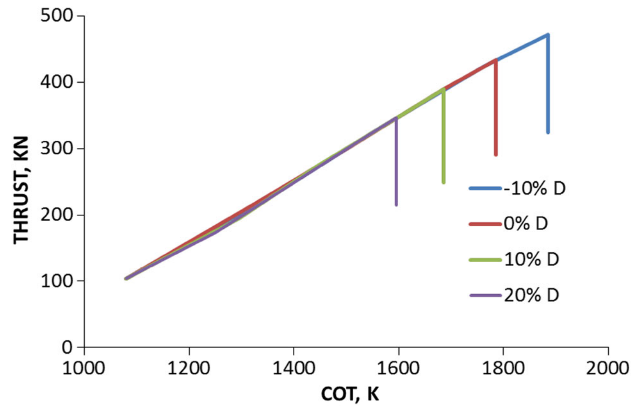



**Fig. 11** Effect of derating on thrust

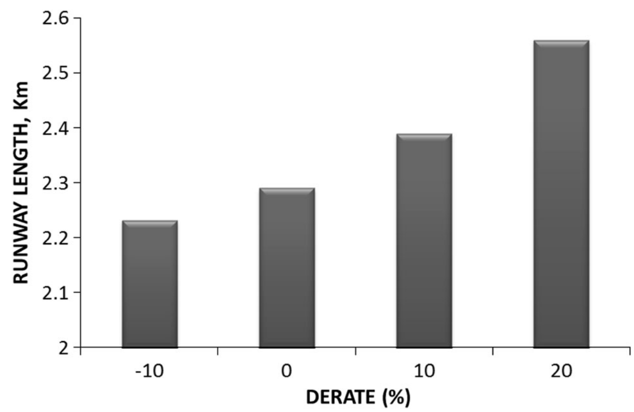



**Fig. 12** Change in runway length with derating

amount of take-off thrust, fuel flow must be increased, as a result COT increases. It is obvious from Fig. 10 that COT increases from 1583 K with OAT = 10 °C to 1735 K with OAT = 40 °C to maintain the same take-off thrust of 387 kN. The fuel flow also increases from 343 to 365 kg.

*Effect of Derate*

Derating of engines is similar to having a less powerful engine on the aircraft. While operating at derating condition, the reduced thrust level is a new maximum and is not intended to be exceeded. Derating may be used only if the expected take-off weight is low enough to permit the use of reduced thrust equal to or below the level of derate. Using a thrust level for take-off which is lower than the maximum allowable, the engine's internal operating pressures and temperatures maintain a lower value which results in reduced stress and wear on the engine, reduced cost of parts, and maintenance and hence increased engine life. However, before derating the engine, it is important to study the runway length, altitude, slope, wind condition, flap setting, etc.

When airport altitude, temperature and payload, fuel load, etc. are kept constant, the COT decreases from with 1885 to 1595 K for −10% derate to 20% derate as shown in Fig. 11. With derate an aircraft with fixed payload requires more runway length to attain the required rotational speed for take-off. The runway length increases by about 330 m with an increase in derating from −10 to 20% as shown in Fig. 12.

## Modes of Failure

There is a continuous demand for higher thrust and lower fuel consumption of aero engines and consequently each stage in the engine is required to work with higher loads without compromising safety. Therefore, aero gas turbine components are designed to operate in aggressive environments where thermal gradients and mechanical loads are high. High reliability of aero engines is therefore of paramount importance. Several lifing procedures have been developed to decide the life cycle of components in order to increase reliability and service life, and reduce maintenance costs. However, there are still a significant number of failures affecting critical components during normal operation and the rejection rate during overhaul for contention of incipient failure mechanisms is fairly high.

Cyclic loading of metallic materials at high temperature is known to cause a complex evolution of damage which can hardly be described in a unique, simple, and straightforward manner [15]. This statement holds true even more correctly in the case that a component is subjected to thermo-mechanical fatigue (TMF) loading conditions which often result in service from a combination of thermal transients during startup and shutdown with mechanical strain cycles [16]. Such complex cycling may lead to component failures resulting from oxidation, fatigue, and creep. The individual extent of these damage constituents and their mutual and synergistic interactions depend very strongly on the specific material considered and the loading conditions applied.

### Creep

Creep is the progressive deformation of a material at constant temperature under the action of constant stress. It is based on the mobility of dislocations and discontinuities in the material, caused by the operating stresses and temperatures: at high temperatures, this process is highlighted, thus aggravating performance and strength. Creep requires four parameters for its description: time, temperature, and stress and strain [17].

The prediction of rupture times at various combinations of stress and temperature usually involves some measure of extrapolation from short term creep tests to long component life times. The simplest way would be to determine the creep rate at short time and assume that this same rate will apply throughout the component life. The most popular of the time–temperature parameters is known as the Larson–Miller parameter which is expressed in the form

$$P = mH/R = (T/1000)(C + \log_{10} t_{\mathrm{f}})$$

From which the time to rupture or failure is calculated as

$$t_{\mathrm{f}} = 10^{(1000P/T)\times C}$$

where $P$ is the Larson–Miller parameter; $H$ is the activation energy for creep; $R$ is the universal gas constant; $T$ is the absolute temperature, K; $m$, $C$ are constants (usually, 20), $t_{\mathrm{f}}$ is the time to rupture/failure, hour.

The procedure for calculating creep life either for a disk or blade is as follows:

- The flight envelope is split into several segments (take-off, climb, cruise, and descent), each one characterized by a time length, and a well-defined operating condition. The Larson–Miller parameter ($P$) for the disk/blade is calculated from the material chart against the resulting stress. The stress is calculated for each flying segment using Von-Mises theorem.
- The temperature ($T$) distribution is obtained from the ANSYS thermal model.
- The cumulative creep life is estimated as Palmgren–Miner law, $\sum \frac{t_i}{t_{\mathrm{f}_i}} = 1$, where $t_i$ is the time duration in $i$th segment of flight and $t_{\mathrm{f}_i}$ is time to creep failure considering the stress and temperature levels of $i$th segment.

### Oxidation

Oxidation is the most important high temperature corrosion reaction. Due to high power demand, the TET is increasing to accommodate more thrust with high EPR. Accordingly the turbine blades are subjected to elevated temperatures in various segments of flight profile.

Prediction of metal temperature is based on measurements, the oxidation depth, which was assumed to be parabolic with time, and confirming to Arrhenius equation between time and temperature, is [18]

$$\left[\log C - \log\left(d^2/t\right)\right][T + 460] = Q/R$$

where log $C$ is the constant (=$A$, suppose); $d$ is the depth of oxidation front, in.; $t$ is the time, hours; $T$ is the temperature on blade surface, °F; $Q$ is the activation energy, ft-lb/lb; $R$ is the gas constant, ft/universal gas constant.

So, time to failure is estimated as $t_{\mathrm{f}} = \frac{d^2}{10^{A-\frac{Q/R}{T+460}}}$

Then as per Miner law, $\sum \frac{t_i}{t_{\mathrm{f}_i}} = 1$ cumulative damage is calculated for the whole mission.

### Low Cycle Fatigue

In aero gas turbine engines, hot end components such as combustion chamber liners, turbine blades, disks, and nozzle guide vanes are designed to operate in high temperature environments with high thermal gradients. During repeated start-ups and shut-downs, these components experience cyclic strains, which are induced both thermally by rapid gas temperature changes, and mechanically by centrifugal force and pressure difference. Under these circumstances, TMF, usually characterized by a LCF, occurs leading to initiation of crack and subsequent crack propagation [19]. Generally life prediction for turbine blades is based solely on crack initiation while for vanes and combustor liners, which have much greater damage tolerance, crack growth life may be used as well as initiation life.

Under the influence of high temperature, the total strain of the component is the summation of elastic strain and plastic strain which are expressed as Baquin equation and Coffin and Manson equation, respectively, as

$$\varepsilon_{\mathrm{e}} = \frac{\sigma_2}{E} = \frac{\sigma_{\mathrm{f}}'}{E}(2N_{\mathrm{f}})^b$$

$$\varepsilon_{\mathrm{p}} = \varepsilon_{\mathrm{f}}'(2N_{\mathrm{f}})^c$$

So as per Neuber rule,

$$\varepsilon_{\mathrm{f}} = \varepsilon_{\mathrm{e}} + \varepsilon_{\mathrm{p}}$$

$$\varepsilon_{\mathrm{f}} = \frac{\sigma_{\mathrm{f}}'}{E}(2N_{\mathrm{f}})^b + \varepsilon_{\mathrm{f}}'(2N_{\mathrm{f}})^c$$

where, $\varepsilon_{\mathrm{e}}$, $\varepsilon_{\mathrm{p}}$ are the elastic and plastic component of the cyclic strain amplitude; $\sigma_{\mathrm{a}}$ is the cyclic stress amplitude, $\sigma_{\mathrm{f}}'$ is the regression intercept called the fatigue strength coefficient; $N_{\mathrm{f}}$ is the number of cycles to failure; $b$ is the regression slope called the fatigue strength exponent; $c$ is the regression slope called the fatigue ductility exponent,

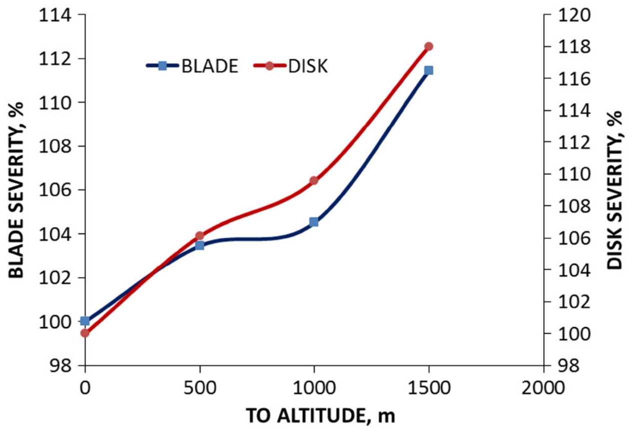



**Fig. 13** Effect of take-off altitude on severity indices of HP turbine blade and disk

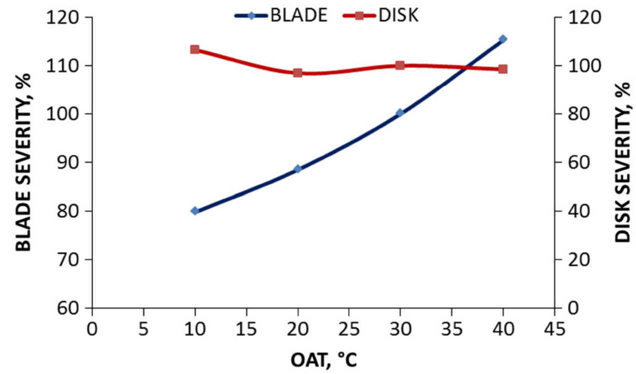



**Fig. 14** Effect of ambient temperature on severity indices of HP turbine blade and disk

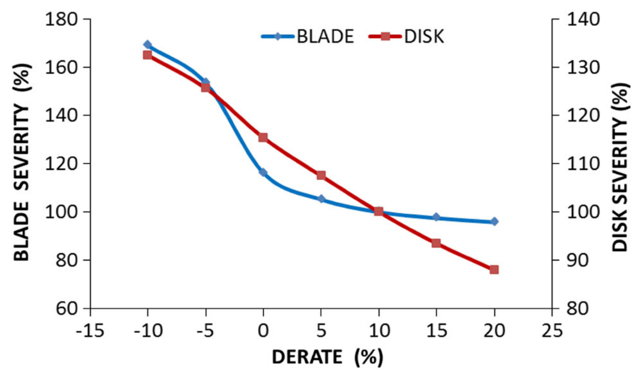



**Fig. 15** Effect of engine derating on severity indices of HP turbine blade and disk

$\varepsilon_f'$ is the regression intercept called the fatigue ductility coefficient.

Then as per Miner law, $\sum \frac{n_i}{N_{f_i}} = 1$, total number of cumulative cycles are calculated, where $n_i$ is the number of cycles in which blade/disk is subjected to stress level $\sigma_i$ and $N_i$ is the number of cycles to failure due to LCF.

## Life Prediction

TMF life prediction models take into account the interaction between cyclic damage (fatigue) and steady state damage (creep and oxidation) at varying temperatures [19, 20]. A disk burst is the worst catastrophic failure possible in engine. Disk design is governed by LCF crack initiation due to engine start and stop cycles [15]. Different models like damage-based criteria, stress-based criteria, strain-based criteria, and energy-based criteria are used by different researchers. The approach taken in this paper is based on damage summation, which calculates different damage index or severity index for different modes of failure and then added to predict the component life.

The linear damage summation model which is also called the linear life fraction or linear cumulative damage, is the simplest expression for steady state cyclic life prediction.

$$D_{\text{total}} = D_{\text{cyclic}} + D_{\text{steady state}} = D_{\text{fatigue}} + D_{\text{creep}} + D_{\text{oxidation}}$$

Individual damage index is calculated from the lifing model either in terms of hours or cycles to failure for various case studies and flight segments. For comparison, these values are converted to non-dimensional numbers known as damage/severity index and are expressed as,

$$\lambda = \frac{1}{\text{LCF}} + \frac{\text{Time of flight}}{\text{Creep damage}} + \frac{\text{Time of flight}}{\text{Oxidation damage}}$$

The damage index value of an ideal case is taken as 100% for both HP turbine blade and disk and with respect to this the severity indices are calculated for variations in take-off conditions, i.e., change in airport altitude, OAT, deratings, and also flight hours as shown in Figs. 13, 14, 15, and 16.

The severity index of both blade and disk increases when altitude of the airport or runway is increased as shown in Fig. 13. Keeping the thrust demand constant for a fixed take-off distance, the engine control system allows more fuel flow which increases the TET and the stress level on blades and disk. When the ambient temperature or OAT increases, the severity index of blades increases due to rise in air temperature of the core flow and in turn TET. However, the severity index of a disk is unaffected by OAT and remains almost 100% as shown in Fig. 14.

Engine derating affects the severity index of blades and disks alike as shown in Fig. 15. The flight duration strongly affects both the blades and disk as shown in Fig. 16. Operating at max power setting for considerable time

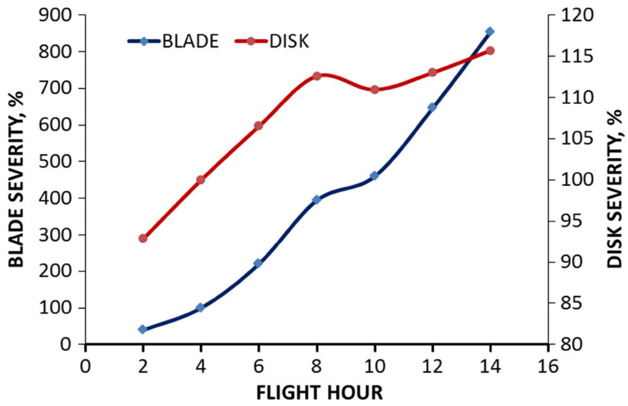



**Fig. 16** Effect of flight duration on severity indices of HP turbine blade and disk

contributes to creep and TMF increasing the severity index and possibly leading to premature failure.

## Conclusions

This analytical study was aimed to minimize the maintenance cost by assessing the life consumption of gas turbine components, especially for turbine blades and disks in different situations. An integrated lifing tool used to calculate the life consumption pattern of these components in terms of total severity that accounts for fatigue, creep, and oxidation was found very effective. The following inferences can be drawn from this study:

- In the flat rating analysis, when the engine operates at 40 °C OAT, the maximum take-off thrust goes down by 4%.
- The engine is found to experience a 30% increase in fuel flow during take-off before surge and 20% reduction in fuel flow before combustion flame extinction during slam acceleration and deceleration.
- COT or TET goes up by 133 °C for a change of airport altitude from 0 to 1500 m and by 152 °C while operating the aircraft in 0–40 °C ambient air temperature.
- Runwaylength increases by 300 m with 20% derate for same amount of payload, but TET goes down by 120 °C.
- Low cycle fatigue and TMF are controlled by shaft speed and have a dominant effect on disk failure, accounting for more than 98% of total damage. Governed by stress, temperature, and time creep on the other hand, accounts for 85% of failure mode in blades.
- Take-off airport altitude has a significant effect on the performance and life consumption pattern of engine components. With increase in altitude, though the climbing distance is reduced, the total severity is increased by 4% with 500 m and 15% with 1500 m altitude variation. Here the blade is affected less (15%) than the disk (18%) because of higher values of centrifugal stress in comparison to thermal stress.
- A 30 °C rise in OAT increases the blade severity by 20% with reference to an ISA normal day but a disk only varies by 2%.
- For HP turbine disk, LCF is driven by shaft speed and temperature but shaft speed has a major influence only at lower temperatures. As temperature increases its effect comes down and also the severity. On an average the severity index of HP turbine rotor assembly goes up by 13% with ambient temperature variation from 10 to 40 °C.
- The total severity of blades deceases as derate percentage increases. The blade severity index drops by 70% while operating between −10 and 0% derate, but only drops 20% from 0 to 20% derate due to reduction in TET.
- For a disk, the total severity index drops by 25% for a 0–20% derate. As it is not affected by the direct gas temperature, the relationship is linear with derate. On an average the total assembly severity index drops by 45% for 0 to −10% derate and 15% for 0–20% derate.
- Operating the engine with 10% take-off derate, the life of critical components like turbine blades and disks increases by 15% compared to full rating.
- The component severity index increases with trip length but the blade severity is much more pronounced than the disk due to more exposure to high temperature compared to a standard mission. As a result, the total severity index of the HP turbine rotor assembly is increased five times in a 14 h flight compared to a normal 4 h flight.

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