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LIFING REQUIREMENT BASED TURBINE AIRFOIL MASS ESTIMATION METHOD IN CONCEPTUAL AERO-ENGINE DESIGN

Stefan Bretschneider*
Hermann Klingels
Fabian Donus
MTU Aero Engines
Dachauer Str. 665
89655 München
Germany

Meinrad Weisser
Institute of Aircraft Propulsion Systems
Universität Stuttgart
70569 Stuttgart
Germany

ABSTRACT

The presented paper describes a method developed by MTU Aero Engines to estimate the mass of turbine blades during multi-disciplinary conceptual design studies based on a prescribed airfoil lifetime [1, 2]. For a given material, the target lifetime can be translated into a maximally allowable material temperature and stress level. While the latter has to be maintained by an appropriate mechanical design of the turbine blades, the material temperature needs to be established by sufficient cooling air. The predominant life-limiting effects are taken into account to determine the allowable temperatures and stresses as an accumulation of the varying operating condition over a flight cycle. The applicable stress levels are then used to calculate the necessary radial area distribution of the airfoil and by this a prediction of its mass is possible. Furthermore, the methodology estimates the required amount of cooling air per airfoil cascade from the computed material temperatures. Example calculations are presented and discussed which illustrate design trends and the benefits which are gained from the proposed method.

NOMENCLATURE

A	Area
a	Alternating
cor	Corrosion
cr	Creep
cyc	Cycle
D	Damage
F	Force
HCF	High Cycle Fatigue
i	Index counter
$incr$	Increment
j	Stage counter
k	Constant
L	Lifetime
M	Heat transfer parameter
m	Mass
$\dot{m}$	Mass flow
max	Maximum
MOPEDS	Modular Performance and Engine Design System
MDP	Mechanical Design Point
MidCL	Mid Climb
MTO	Maximum Take-Off
N	Number
$OTDF$	Overall Temperature Distribution Factor
OP	Operating point
p_3	Compressor exit pressure
r	Radius
r_H	Airfoil hub radius
R	Rotor
R_j	Degree of reaction
$RTDF$	Radial Temperature Distribution Factor
S	Stator

*Address all correspondence to this author.

T	Temperature
t	Time
T_{CA}	Cooling air temperature
T_G	Gas temperature
T_M	Material temperature
TOC	Top of Climb
tot	Total
Δ	Delta
ε_{CA}	Cooling effectiveness
η_{CA}	Cooling efficiency
σ	Stress
Ψ_j	Stage loading
ρ	Density
ω	Rotational speed

INTRODUCTION

Within the conceptual design phase of an aero-engine each concept under investigation is typically represented initially as a performance cycle. Other preliminary design disciplines are subsequently added in order to include all important inter-disciplinary dependencies; these disciplines include topics from aerodynamics, weights and dimensions, mechanics, noise, and cost. The multi-disciplinary interdependencies are then taken into account in order to find the best possible engine concept at a given technology level. In this context many conceptual design tools feature methods to estimate component masses and dimensions based on the automated knowledge-based preliminary design of their simplified parts. Some well-known published examples are NASA's WATE (Weight Analysis of Turbine Engines) code [3], GENERAL ELECTRIC's CPD (Computerized Preliminary Design Code) [4], ROLLS-ROYCE's GENESIS [5], MTU AERO ENGINES's MOPEDS (Modular Performance and Engine Design System) [6], and PMDO-LITE (Preliminary Multi-Disciplinary Optimization) [7] from PRATT&WHITNEY CANADA. DONUS [8] proved that the accuracy of such simplified methods is generally very high. However, the author also concluded that a good mass estimation especially of the turbine airfoils is difficult to achieve with current methods. Even though the airfoil mass makes-up only a small fraction of the component mass, it highly influences the mass of the turbine disks, which typically contributes the largest weight portion to the component, as illustrated by BRETSCHNEIDER [9].

Within some preliminary design methods, such as described by KURZKE [10], the material temperatures which are necessary for the design of the airfoils need to be obtained from the mean gas temperatures. This is done by estimating an amount of cooling air from empirical correlations. The obtained material temperature is then used to get an allowable stress level from a material database. With some simplified beam theory methods [11] a first estimate of the airfoil's radial area distribution and its mass is made possible. Even though this procedure is a good starting point, it has some disadvantages: If the material temperature used for the airfoil design is determined from the mean gas temperatures and the cooling air assumptions only, all uncertainties of the prediction are directly transfered to the allowable design stress and the estimated mass. Turbine blades are life limited parts and thus the consequence is that the lifetime of the turbine alters with the calculated material temperature over a concept study with varying cycle temperatures. In reality, a certain lifetime is actually a design target, due to its strong influence on the projected maintenance costs. From the mechanical point of view the resulting airfoil mass is thus a result of three major interdependent contributors as illustrated in Fig. (1): Target lifetime, cooling air and design criteria. The following discussions will show that the material selection is the true technology limiter which drives the allowable design stress, temperature, airfoil mass and also the necessary amount of cooling air as a function of the target lifetime only. With this, the airfoil design criteria can be gained directly from the material characteristics and the chosen design mission. The necessary amount of cooling air then becomes a side product, which is obtained from the difference between the metal and gas temperatures and the applied cooling technology. If used in conceptual design, this guarantees that all turbines investigated during a concept study are compared on the basis of a common target lifetime.

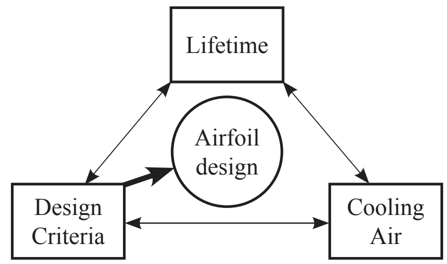

FIGURE 1. AIRFOIL DESIGN CONTRIBUTORS [2]

The authors want to point out, that the intent of the proposed method is not to deliver an aerodynamic design of the airfoil. The method was developed to enable a more realistic mass estimation of the turbine airfoils at a time when no aerodynamic profiling is performed yet.

DESIGN CRITERIA FROM LIFING ASSESSMENT

Cumulative Damage over the Flight Cycle

During one flight cycle all turbine parts are subjected to a variable load profile at which temperatures and rotational speeds are changing dependent on the performance of the engine. At all phases during each flight, the life-limited parts are incrementally damaged dependent on the current operating condition. For this reason the turbine airfoils need to be designed for a specific surveillance time over which these cyclic loads can occur before the part's lifetime is consumed and it needs to be replaced. The latter is typically given as the number of flight cycles that the turbine part has to survive. The damage D_{OP} caused at each operating point is quantified as the ratio of duration t_{OP} and expected lifetime L_{OP} at the occurring stress level

$$D_{OP} = \frac{t_{OP}}{L_{OP}} . \tag{1}$$

This formulation was first used by MINER [12] for the design of parts under cyclic loads, but it can be applied to include other types of damages into the damage calculation too. In order to reduce the number of necessary operating points in the preliminary design calculation to a reasonable amount, the target mission is approximated by characteristic sections with constant thermal and mechanical loads. Some of the mission points which are typically taken into consideration are Take-off (TO), Mid-Climb (MidCL), Top-of-Climb (TOC), Cruise, and Approach as shown in Fig. (2). The overall damage of an airfoil during one flight cycle needs to be determined from the sum of the individual damages caused at all operating points over the target mission by assuming MINER's theory of linear damage accumulation

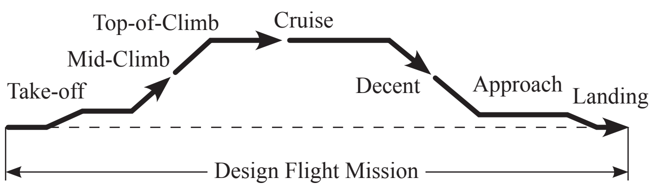

FIGURE 2. TYPICAL DESIGN FLIGHT MISSION

$$D_{cyc} = D_{TO} + D_{MidCl} + D_{TOC} + D_{Cruise} + \dots , \tag{2}$$

$$D_{cyc} = \frac{t_{TO}}{L_{TO}} + \frac{t_{MidCl}}{L_{MidCl}} + \frac{t_{TOC}}{L_{TOC}} + \frac{t_{Cruise}}{L_{Cruise}} + \dots . \tag{3}$$

The formulation allows to freely extend the amount of operating points dependent on the available engine performance data and the desired accuracy. According to MINER it is necessary to determine the maximally allowable cumulative damage D_{max} from experiments. However, an often applied simplification is to assume that the maximum lifetime is reached when the cumulative damage is approximately unity $D_{max} \approx 1$ (ROBINSON's rule). With this assumption the number of flight cycles N_{cyc} can be obtained from Eqn. (2) until when the full part life has been consumed.

$$N_{cyc} = \frac{D_{max}}{D_{cyc}} \approx \frac{1}{D_{cyc}} . \tag{4}$$

With this, the maximally achievable lifetime L_{tot} with respect to the targeted flight mission is obtained from the product of the number of flight cycles N_{cyc} and their duration t_{cyc} and thus approximatly equal to

$$L_{tot} \approx t_{cyc} \cdot \left[\frac{t_{TO}}{L_{TO}} + \frac{t_{MidCl}}{L_{MidCl}} + \frac{t_{TOC}}{L_{TOC}} + \frac{t_{Cruise}}{L_{Cruise}} + \dots \right]^{-1} . \tag{5}$$

where t_{cyc} is the overall mission time calculated as the sum of the durations of all flight sections. From this discussion it becomes clear that the design of all life-limited parts will be strongly influenced by the desired lifetime, by the targeted flight mission and by the stress and temperature levels coming from the selected performance cycle. With the above simplifications and Eqn. (1) to (5) it is possible to translate the defined design lifetime, into an equivalent corrosion and creep lifetime, L_{cor} and L_{cr}, at one mechanical design point (MDP) only. The latter is not part of the target mission, it is an artificial condition that consists of the worst case requirements of all operating points and includes additional margins, such as the highest temperatures and largest rotational speeds.

Relevant Damage Types

Hot Gas Corrosion The oxygen in the turbine gas flow reacts with the airfoil material due to the high temperatures causing hot gas corrosion. This reaction creates metal oxides at the surface of the airfoils. The oxidation products are brittle and cracked so that the erosion of the airfoil is even accelerated and severe damage is caused. In addition to oxidation, the airfoil surface is attacked by pollutants in the main gas flow. Typical contamination involves salt or sulphur compounds. Especially the latter leads to the generation of metal sulphides of the base material which are characterized by low melting points. The sulphide melts and the airfoil is eroded by the high flow velocities [13]. Corrosion resistance can be improved by coating of the base material. A typical treatment to enhance corrosion resistance is the diffusion of chromium or aluminium into the base material. This treatment creates metal sulphides at the airfoil surface with very high melting temperatures [14] and enhances the base material. Another alternative is vapour coating of corrosion resistant materials. Especially ceramics are often applied to turbine airfoils. Besides their favorable corrosion properties ceramic coatings have a low thermal conductivity. This advantage can either be used to further increase the cycle temperatures or to reduce the necessary amount of cooling air while maintaining constant airfoil life. However, the lifetime characteristics are dependent on the selected base material and coating. Therefore material data for every material pairing needs to be available. In general, materials show a reduction of corrosion lifetime L_{cor} with an increase of material temperature T_M as schematically depicted in Fig. (3). Plotted over a logarithmic scale a linear reduction of lifetime over temperature is typical. Once a certain temperature limit T_{lim} is exceeded, a rapid decrease of lifetime is observed. Dependent on the applied combination of airfoil material and coating this effect is either caused by the the evaporation or melting of the latter. For simplicity it is assumed in the following that maximum life is reached and failure occurs as soon as the coating is worn-out at one single spot at the surface of a turbine airfoil. Therefore, it is necessary to limit the maximally occurring material temperatures to a level that the desired lifetime can be reached. Corrosion lifetime L_{cor} for a given flight cycle is calculated from the duration t_i and the lifetime $L_{cor,i}$ at all N operating points by using MINER's method of linear damage accumulation

$$L_{cor} = t_{cyc} \cdot \left[\sum_{i=1}^{N} \frac{t_i}{L_{cor}\left(\hat{T}_{M,i}\right)} \right]^{-1} . \tag{6}$$

In order to achieve a unique solution it is necessary to develop a functional dependency between the peak material temperatures of all operating points. For this purpose it is assumed that the achievable peak cooling effectiveness $\hat{\varepsilon}_{CA}$ is identical at all operating points. With this, the cooling air temperature $T_{CA,i}$ and the peak gas temperature $\hat{T}_{G,i}$ of two operating points are related by

$$\hat{\varepsilon}_{CA} = \frac{\hat{T}_{G,i} - \hat{T}_{M,i}}{\hat{T}_{G,i} - T_{CA,i}} = \frac{\hat{T}_{G,i+1} - \hat{T}_{M,i+1}}{\hat{T}_{G,i+1} - T_{CA,i+1}} . \tag{7}$$

The gas and cooling air temperatures at all operating points are obtained from performance synthesis in advance. However, synthesis can only predict mean gas temperatures $\bar{T}_G$ but no peak material temperatures. For this reason an estimation is described in a later section which allows to relate both temperatures. With Eqn. (6) and Eqn. (7) a relationship between the desired corrosion lifetime L_{ox} and the peak gas temperatures over the target mission is obtained. This is used to calculate the maximally allowable airfoil peak temperature at all operating points so that the lifing requirement is fulfilled.

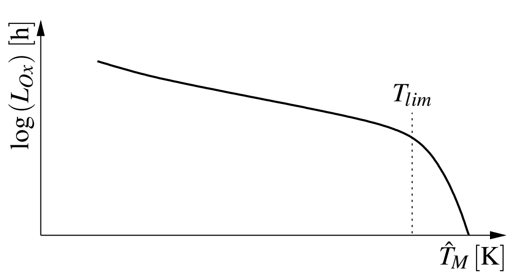

FIGURE 3. CORROSION LIFETIME CHARACTERISTICS [2]

Creep Damage If materials are subjected to heat for a longer period they start to slowly deform under the influence of stresses. This occurs even if the stress levels are way below the yield strength of the material. For this reason the creep properties of airfoil materials are of special importance to the design of turbines [15], because they are not only subjected to high temperatures but also to severe mechanical loads caused by the centrifugal forces. Creep material data is typically available in form of LARSON-MILLER curves. However, in case of this work MTU's material database was used to deliver the relationship between tolerable creep stress σ_{cr}, mean material temperature $\bar{T}_M$ and creep lifetime L_{cr}. The necessary mean material temperature is obtained from an empirically known temperature difference ΔT_M^{MDP} between the maximally tolerable peak temperature - which is defined during the corrosion life assessment - and the mean temperature inside of the airfoil at the same radial position at MDP. This difference is best formulated as a function of the mean cooling effectiveness

$$\Delta T_M^{MDP} = k\,[K] \cdot \bar{\varepsilon}_{CA} \tag{8}$$

whereby the parameter k has to be empirically determined from existing reference turbine designs. The mean cooling effectiveness $\bar{\varepsilon}_{CA}$ is determined by

$$\bar{\varepsilon}_{CA} = \frac{\hat{T}_G^{MDP} - \bar{T}_M^{MDP}}{\hat{T}_G^{MDP} - T_{CA}^{MDP}} = \frac{\hat{T}_G^{MDP} - \left(\hat{T}_M^{MDP} - \Delta T_M^{MDP}\right)}{\hat{T}_G^{MDP} - T_{CA}^{MDP}} . \tag{9}$$

The temperature difference between the peak and the mean material temperature ΔT_M^{MDP} is directly obtained from Eqn. (8) and (9).

$$\Delta T_M^{MDP} = \frac{k\left(\hat{T}_G^{MDP} - \hat{T}_M^{MDP}\right)}{\hat{T}_G^{MDP} - T_{CA}^{MDP} - k} . \tag{10}$$

This formulation has the advantage that ΔT_M^{MDP} is nullified in case of an uncooled airfoil, where $\hat{T}_G^{MDP} = \hat{T}_M^{MDP}$. The scaling of the peak material temperature between the individual mission points is again based on the assumption of a constant mean cooling effectiveness $\bar{\varepsilon}_{CA}$ over the operating range [see Eqn. (7)]

$$\bar{\varepsilon}_{CA} = \frac{\hat{T}_{G,i} - \left(\hat{T}_{M,i} - \Delta T_{M,i}\right)}{\hat{T}_{G,i} - T_{CA,i}} = \frac{\hat{T}_G^{MDP} - \left(\hat{T}_M^{MDP} - \Delta T_M^{MDP}\right)}{\hat{T}_G^{MDP} - T_{CA}^{MDP}} . \tag{11}$$

With this, the temperature differences $\Delta T_{M,i}$ and the mean material temperatures $\bar{T}_{M,i}$ at all mission points can be determined. Again, the damage caused by creep is accumulated with ROBINSON's rule. The mean temperature at each operating point is calculated from $\bar{T}_M = \hat{T}_M - \Delta T_M$. Once the material temperature is set, the creep lifetime at each operating point is only a function of the occurring stresses $\sigma_{cr,i}$

$$L_{cr} = t_{cyc} \cdot \left[\sum_{i=1}^{N} \frac{t_i}{L_{cr}(\sigma_{cr,i})} \right]^{-1} . \quad (12)$$

In order to solve Eqn. (12) it is necessary to define a relationship between the occurring stresses at the different operating points. Because the loading of the rotor blades is governed by the centrifugal forces, it can be assumed that the stresses in the rotor blades scale proportional with the square of the rotational speeds n^2. With this, the stress levels of two operating points can be related

$$\frac{\sigma_{cr,i}}{\sigma_{cr,i+1}} = \left[\frac{n_i}{n_{i+1}} \right]^2 . \quad (13)$$

Because stator vanes are only subjected to aerodynamic forces, it is assumed that the stresses within their airfoils scale proportional to the pressure at the exit of the combustor p_3

$$\frac{\sigma_i}{\sigma_{i+1}} = \frac{p_{3,i}}{p_{3,i+1}} . \quad (14)$$

It was already mentioned that performance data and rotational speeds are available prior to the execution of this method. The maximally tolerable stresses at all operating points can be calculated from Eqn. (12) to Eqn. (14) in an iterative process. With this, the design stress at the position of the peak material temperature is obtained from the lifing requirement.

HCF - High Cycle Fatigue High cyclic, stochastic loads are generated within the airfoils by aerodynamic and acoustic oscillations during the operation of the turbine. Even though their amplitude is small, such loads lead to structural damages after only a limited amount of cycles. This damage type is often referred to as high cycle fatigue (HCF). Unlike to what was described in the previous sections, turbine blades are designed as HCF endurable. This eases the design calculations because no damage accumulation needs to be considered. Within the context of the this method the alternating stresses caused from HCF are approximated as sinusoidal oscillations. Furthermore, the maximal tolerable material temperature $\hat{T}_M$ is already known from what was calculated in respect to corrosion. Typically materials can survive an unlimited amount of load alternations below a critical stress level. The corresponding stress amplitude, called fatigue strength, is dependent on the oscillation's mean stress σ_m and the material's temperature. The fatigue strength is typically depicted in HAIGH diagrams. However, a HAIGH diagram is only valid for a single material temperature, consequently a set of diagrams is necessary to assess HCF at different temperatures. The expected HCF stress amplitude σ_a is defined from experience with existing designs. With this the maximally acceptable mean stress σ_m can be obtained by interpolation between this set of HAIGH diagrams in every operating point at the defined material temperature $\hat{T}_M$. The maximally occurring airfoil stress needs to be restricted to this value.

TEMPERATURES AND COOLING FLOWS

Estimation of Cooling Flows From the previous discussions it becomes clear, that the maximally tolerable material temperature and stress level is not to be exceeded if the lifetime requirement needs to be fulfilled. In case the peak gas temperature $\hat{T}_G$ exceeds the calculated limit, enough cooling air is necessary to establish the desired material temperatures. Hence, it is suggesting to include a methodology which also allows to estimate the necessary amount of cooling flow from the lifing requirement. A method based on the work of AINLEY [16] and the simplifications introduced by HALLS [17] is implemented to estimate the cooling flow from the difference between gas stream and material temperature. Only the peak gas temperature at MDP is relevant for the definition of the cooling flow, because this is also the operating point where the highest thermal loads occur. The cooling flow is adjusted so that the maximally tolerable mean material $\bar{T}_M$ is met at the hottest section of the airfoil. It can be shown that together with the mean cooling effectiveness $\bar{\varepsilon}$ at MDP from Eqn. (9) a simple formulation of the heat transfer parameter M in the airfoil is obtained

$$M = \frac{1}{\eta_{CA}} \cdot \frac{\bar{\varepsilon}_{CA}^{MDP}}{1 - \bar{\varepsilon}_{CA}^{MDP}} , \quad (15)$$

which is directly related to the necessary amount of cooling air; see GRIEB [18]. The cooling efficiency η_{CA} is a descriptor of the applied cooling technology. Based on the experience from existing engines a linear correlation is often used to describe the ratio of cooling air $\dot{m}_{CA}$ and compressor inlet flow $\dot{m}_{25}$ as a function of the heat transfer parameter

$$\left(\frac{\dot{m}_{CA}}{\dot{m}_{25}} \right)_{blade} = k_1 + k_2 \cdot M = k_1 + k_2 \cdot \frac{1}{\eta_{CA}} \cdot \frac{\bar{\varepsilon}_{CA}^{MDP}}{1 - \bar{\varepsilon}_{CA}^{MDP}} . \quad (16)$$

Some authors have published values for the parameters k_1 and k_2: GRIEB quantified $k_1 \approx 0.007 \ldots 0.018$ und $k_2 = 0.017$, HORLOCK et al. [19] suggest $k_1 = 0$ and $k_2 = 0.035$, YOUNG und WILCOCK [20] give $k_1 = 0$ und $k_2 = 0.045$, and HOLLAND and THAKE [21] have used $k_1 = 0$ and $k_2 = 0.025$. The variety of figures illustrates the dependency on technology level and cooling technology. A further refinement of the method is possible if different values for rotors and stators are used. Additional cooling air flows are used for platform cooling or as sealing air. In general, the required cooling air needs to be taken into account for each airfoil cascade, because the flow is mixed into the main gas stream and thus lowers the temperature of the succeeding blade row. This has the additional advantage that also the distribution of the cooling flows over the turbine component is obtained.

Gas Temperature Distribution It was already shown, that the determination of the peak gas temperature $\hat{T}_G^{MDP}$ is significant for the estimation of the cooling flows. Furthermore, a relationship between the peak and mean gas temperatures is required in all operating points to cumulate the incremental damages of corrosion and creep. In general, the difference between peak and mean temperature originates from the radial temperature profile which is generated over the the cascade as it is schematically shown in Fig. (4) for a generic mid stage turbine cascade. However, only the mean gas temperatures can be predicted by performance synthesis. In order to also obtain the peak gas temperatures, an approach based on the work of GAUNTNER [22] was chosen: If a recovery factor of unity is assumed,

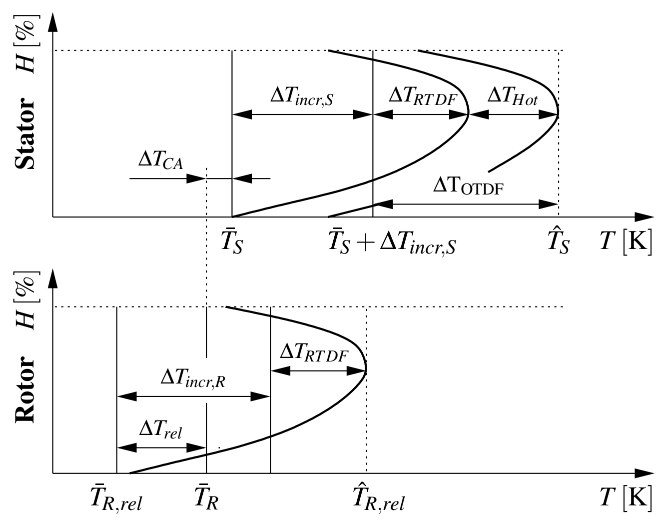

FIGURE 4. ESTIMATION OF PEAK GAS TEMPERATURE [2]

the peak gas temperature of the j-th stage's stator vane $\hat{T}_{G,S,j}$ can be obtained by adding the increments $\Delta T_{incr,S,j}$ und $\Delta T_{OTDF,j}$ to the mean gas temperature at the entry of the component $\bar{T}_{G,j}$

$$\hat{T}_{G,S,j} = \bar{T}_{S,j} + \Delta T_{incr,S,n} + \Delta T_{OTDF,j} \,. \tag{17}$$

Behind the stator the mean gas temperature $\bar{T}_{G,j}$ is slightly reduced by ΔT_{CA}, because the stator's cooling air is mixed into the main stream. However, an analogue expression can be used to derive the peak gas temperature of each rotor blade by

$$\hat{T}_{G,R,j} = \bar{T}_{R,j} - \Delta T_{G,rel,j} + \Delta T_{incr,R,j} + \Delta T_{RTDF,j} \,. \tag{18}$$

The difference between peak and mean gas temperature is described by ΔT_{RTDF}. In addition to that, a variation of the circumferential temperature exists in the stator cascades which is caused by the positioning of the burners in the combustor. The thermal worst case for the design of the stator vanes occurs if the airfoil is located in a combustor hot spot. Therefore, the gas temperature is additionally incremented by ΔT_{Hot}. Both increments ΔT_{Hot} and ΔT_{RTDF} are combined and written as ΔT_{OTDF}. Both increments ΔT_{RTDF} and ΔT_{OTDF} are described as a fraction of the temperature increase over the combustor [23]. With this, the increments of the j-th stage can be written as

$$\Delta T_{OTDF,j} = OTDF_j \cdot (T_4 - T_{31}) \,, \tag{19}$$
$$\Delta T_{RTDF,j} = RTDF_j \cdot (T_4 - T_{31}) \,, \tag{20}$$

whereby the circumferential temperature profile is not relevant to the rotor blades, because they only face an averaged temperature peak because of their rotation. The two coefficients $OTDF_j$ and $RTDF_j$ are technology descriptors which are dependent on the design of the combustor. While GRIEB suggests values of $OTDF \approx 0.30 \ldots 0.35$ and $RTDF \approx 0.08 \ldots 0.10$, YOUNG and WILCOCK [20] have published $OTDF \approx 0.10$ and $RTDF \approx 0.05$ for the first stage of an high pressure turbine. Values of low pressure turbines are smaller, because the temperature profiles are diminished by preceding cascades. Furthermore, a temperature increment $\Delta T_{incr,S,j}$ is added to the mean temperature of each cascade which emulates effects such as uncertainties, deterioration, production scatter and random temperature peaks. Typically the increments are known from experience at the inlet and outlet of the component. In order to obtain the increment of each stage from these values a linear distribution proportional to the relative enthalpy drop over each stage is assumed. Because it is the temperature in the rotating system which is relevant to the rotor blades, the increment of the rotor $\Delta T_{incr,R,j}$ needs to be accounted in the rotating system. A good estimate is to multiply the stators's increment by the ratio of the mean temperatures of both systems. For the same reason an additional difference $\Delta T_{G,rel,j}$ needs to be applied to the mean temperature of the rotor. With some degree of simplification, this can be expressed as a function of the stage's temperature drop ΔT_j, its aerodynamic loading factor Ψ_j and its kinematic degree of reaction R_j [18]

$$\Delta T_{G,rel,j} = \left[\frac{1}{2} - \frac{1}{2\Psi_j}(2R_j - 1)\right] \Delta T_j \,. \tag{21}$$

The temperature of the cooling air is known from its off-take position in the compressor. However, increments $T_{incr,CA}$ may also be applied to the cooling air temperature because of the above mentioned effects. In addition to that, the cooling air temperature can be altered by pumping effects or the application of pre-swirl nozzles.

ESTIMATION OF AIRFOIL MASS

In order to estimate the weight of the turbine airfoils, it is required to perform a preliminary mechanical design based on physical design laws and loads. In general, turbine blades are subjected to a variety of loads which have to be considered during their design. Typical airfoil loads are centrifugal stresses, bending moments, torsional moments and cyclic loads from aerodynamic or acoustic oscillations. However, at the time when conceptual preliminary design studies are performed, the detailed aerodynamic shaping of the turbine airfoils is not known yet. Typically, the airfoil is only represented by an aspect and taper ratio within an annulus. Some published methods employ flat plates [24] or prismatic airfoils from standard profiles [9] to generate a blade mass. Even though it becomes obvious that twisting moments and bending moments cannot be included, it is possible to assess the loads from centrifugal forces and from oscillations by simplified correlations. The schematic of a turbine rotor blade is shown in Fig. (5). The stress distribution in the airfoil is dependent on the centrifugal load which is caused by the blade's own mass [11]. An incrementally thick slice of the airfoil of mass dm causes a centrifugal load dF which is dependent on the radial position r, the rotational speed ω, and the cross sectional area $A(r)$ of the slice itself

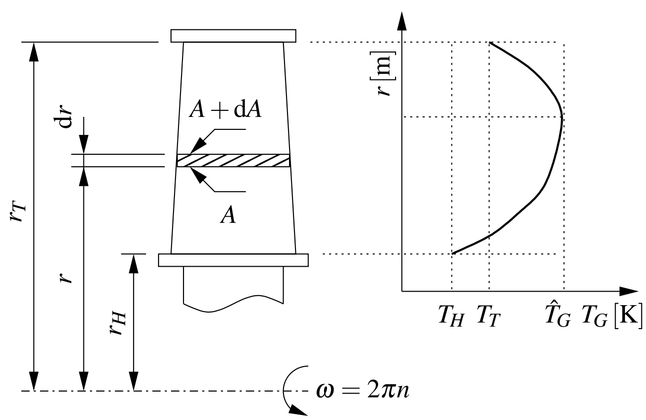

FIGURE 5. TURBINE AIRFOIL DEFINITIONS [2]

$$\mathrm{d}F = r\omega^2\,\mathrm{d}m = \rho r \omega^2 A(r)\,\mathrm{d}r. \qquad (22)$$

If Eqn. (22) is integrated starting from the airfoil's tip at r_T, a functional dependency between the centrifugal force F, the radial stress σ and the airfoil's cross sectional area at every radial position between r and r_T is obtained

$$F(r) = \rho\omega^2 \int_r^{r_T} A(r) r\,\mathrm{d}r \quad \text{and} \quad \sigma(r) = \frac{\rho\omega^2}{A(r)} \int_r^{r_T} A(r) r\,\mathrm{d}r. \qquad (23)$$

If a radial area distribution can be found which satisfies the allowable stress criteria at all radial positions the determination of the airfoil mass becomes possible too. The airfoil's tip area A_T can either be defined from aerodynamic profiling, or in case of a shrouded rotor, it needs to be large enough to carry the additional load of the shroud. From here, a certain increase of the airfoil cross section is typically necessary just because of manufacturing reasons. This generates a linear increase of airfoil area and centrifugal stress. Once the allowable limit is reached, a larger growth rate is required to not further increase the stress level. In case a constant maximum stress is used for the design, an analytical formulation is of the area increase is possible, such as given by KAWAIKE [25]. However, using only the stress limit which was obtained from the hottest cross section, would create a very conservative and too heavy a design. With the knowledge of the temperature profile a local design stress criteria from HCF resistance and creep life is obtained at every radial position. Whereby it is the smaller stress of both which is used to calculate the area increase if the gain from evaluating the manufacturing taper was not sufficient enough to not exceed the allowable limit. This procedure has the advantage that a smaller growth rate of the cross sections is acceptable, because the lower temperatures increase the acceptable stress levels. An example of a possible radial distribution of stress and area is shown in Fig. (6) showing a taper-driven section at the top and local design limits from

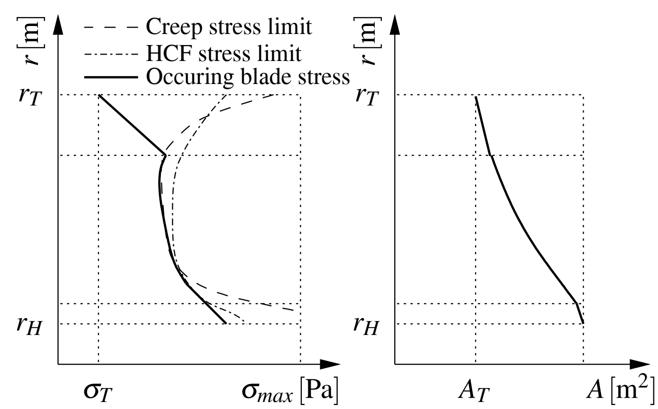

FIGURE 6. STRESS AND AREA DISTRIBUTION [2]

HCF and creep. At this stage, corrosion is not an issue because the airfoil's peak temperature was already selected to suit the appropriate lifetime. The final mass m of the airfoil can be obtained by the integration of the computed radial area distribution

SOME RESULTS AND EXAMPLES

Airfoil Redesign In order to evaluate the quality of the described methodology, it was applied to conceptually redesign three rotor blades from existing low pressure turbines. The first example is characterized by high temperature levels, hollow profiles and a cooled trailing edge. It is thus dominated by design stresses from corrosion and creep. The other two examples are uncooled and have lower material temperatures, thus they also reflect design stresses defined by HCF. The calculated radial area distributions as well as the comparison to the existing airfoil is presented in Fig. (7). The upper radial sections of all three airfoils is mainly influenced by the chosen taper ratio and thus precisely met. Once the design stress limit is reached, the radial area increase follows the stress criterion which is either defined by creep or HCF. It can be seen, that the redesign of the first example blade matches its original very well at all radial positions. This is also true for the other two examples except for the area distribution in the lower section between 0% and 40% of the radial height of the second example blade. Here the areas are noticably overpredicted. The same applies to the lower section of the third example but with a smaller deviation to the original. These differences originate from the idealized representation of the radial temperature profiles. The latter directly affects the local design criterion. Furthermore, any error made effects all subsequent lower radial positions. The mass and center of gravity of each of the examples was computed and the difference between real blade and conceptual design was evaluated as shown in Table (1). Even though the calculated radial area distributions deviated slightly from their corresponding originals, the results show that the developed methodology is capable of delivering a very high result quality during conceptual design studies. While the predicted mass only deviated within a range of 11%, the calculated centers of gravity where as close as 1.2%.

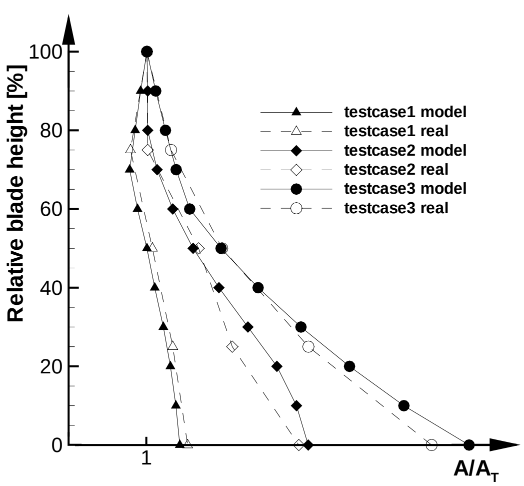

FIGURE 7. AIRFOIL RADIAL AREA DISTRIBUTION

TABLE 1. DIFFERENCE IN MASS AND CENTER OF GRAVITY

Testcase	No 1	No 2	No 3
Mass	2.3%	10.8%	5.0%
Center of Gravity	0.2%	0.3%	1.2%

Effect of Lifetime on Mass In order to illustrate the effect of a varying lifetime requirement on the results of the conceptual airfoil design an example study was performed. An uncooled example blade was selected as a reference design and the target lifetime was varied. In this case, the airfoil's material temperature is constant over the study and the required lifetime is only dependent on the creep characteristic and not on corrosion. Otherwise, cooling would be necessary to alter the materials peak temperature and with this the corrosion lifetime. It was further assumed that the airfoil's tip area is defined from aerodynamic aspects only and thus it is also kept constant. From the LARSON-MILLER characteristics it becomes clear that the tolerable design stress needs to be reduced if the lifetime target increases at a constant temperature. The radial stress and area distributions as a function of the relative target lifetime are shown in Fig. (8). The computed airfoil mass is shown over the required lifetime in Fig. (9). Within this study, airfoil mass is increased by almost 20% if the lifetime target is doubled. The increase of the target lifetime leads to the expected reduction of the tolerable design stress and thus an increase in area and mass. However, if the lifetime requirement becomes very small, the area distribution and thus also the mass is not further reduced. Here, the area distribution is only defined by the production taper - which is of course a rather theoretical limit. The tolerable stresses defined by the creep characteristic have become so large that they are no longer relevant. The results of this study are, of course, dependent on the chosen airfoil material and temperature. Still, the example illustrates the desired effect of the lifetime on the predicted airfoil mass.

Effect of Lifetime on Cooling Air The relationship between target lifetime, tolerable material temperature at MDP and

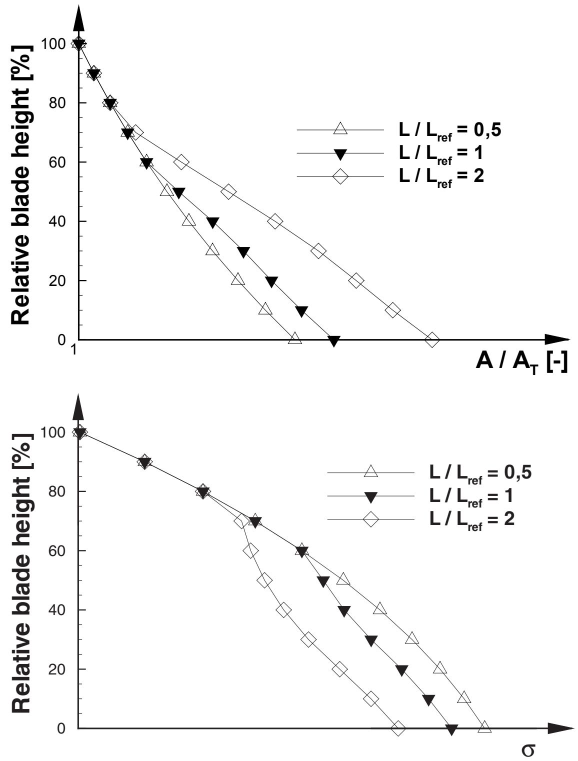

FIGURE 8. RADIAL DISTRIBUTIONS VS. LIFETIME [2]

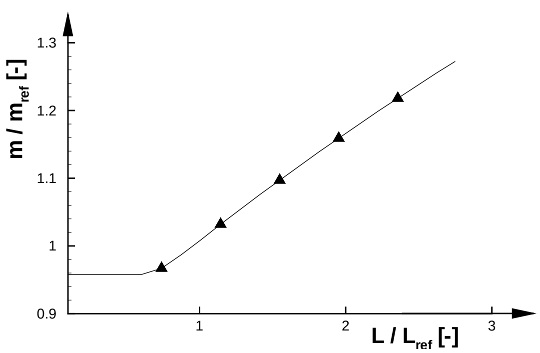

FIGURE 9. AIRFOIL MASS VS. LIFETIME [2]

cooling air was described in the previous sections. A study with varying target lifetimes was performed to illustrate this dependency and the value of the developed methodology. The calculated amount of cooling air of the first cascades of a low pressure turbine is displayed as a function of lifetime is depicted in Fig. (10). The tolerable airfoil temperature at MDP is shown in Fig. (11), whereby the material temperatures of the three cascades are normalized with the temperature of the uncooled airfoil. It can be seen, that an increase in target lifetime requires a lower material temperature, because the corrosion and creep life

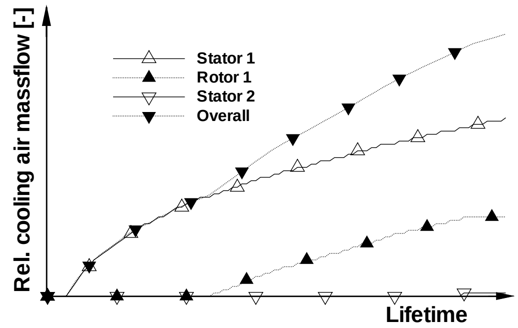

FIGURE 10. COOLING AIR REQUIREMENT VS. LIFETIME [2]

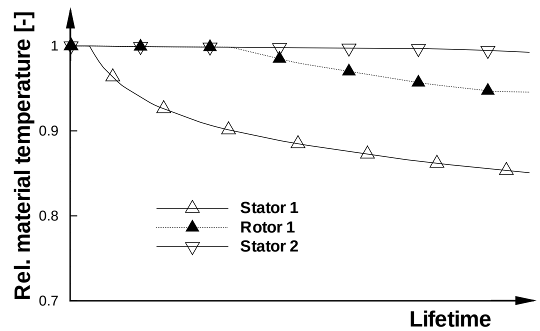

FIGURE 11. TEMPERATURE VS. LIFETIME [2]

of each cascade is directly related to its temperature. This can only be achieved by the use of additional cooling air if a constant cooling technology level is assumed. The cooling air itself is mixed into the main stream after each cascade. This decreases the gas temperature of the succeeding cascade and helps to fulfill the lifing requirement of this airfoil. However, if the target lifetime is further increased, cooling is also required here and a discontinuity in the cooling flow characteristic occurs.

SUMMARY AND CONCLUSION

The presented method estimates the mass of turbine blades during multi-disciplinary conceptual design studies on the basis of a common target lifetime. The advantage of the method is the inclusion of multiple operating points over a flight mission and lifetime dependent material data. If both are known, the target lifetime can be translated into a maximally allowable material temperature and stress level. While the stress level can be maintained by an appropriate mechanical design of the turbine blades, the material temperature needs to be established by a sufficient amount of cooling air. The necessary performance data is available during cycle design from synthesis calculations. The redesign of three existing low pressure turbine airfoils showed that the airfoil mass can be estimated within a precision of 11%. This is an encouraging result, especially if all uncertainties and the number of undefined details is taken into consideration, which still exist at the conceptual design phase. Even more important, all relevant dependencies between cycle design, cooling air requirements and mechanical design based on material properties are included into conceptual design studies. Even though results were only presented for low pressure turbine blades without film cooling, it is expected that the proposed methodology can also be applied to high pressure turbine blades within the conceptual design phase, if the necessary knowledge base is available.

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