Algorithmic Aero Engine Life Usage Monitoring Based on Reference Analysis of Design Mission

MANFRED KOEHL

1 ABSTRACT

Fracture critical areas in an aero engine are usually located in fast rotating components which are exposed to high temperatures. Therefore, a direct local measurement of damage related parameters like strains and temperatures is not feasible. Instead, only a few remotely measured quantities, e.g. engine intake conditions, rotational speeds, and, possibly, gas path temperatures and pressures, are accessible as input for an on-board life usage monitoring system. This shortcoming has been compensated by sophisticated mathematical models which calculate the local physical parameters needed for assessing the life consumption of each critical area.

The models for temperature and stress calculation consist of algorithms containing parameters which have to be adjusted individually for each considered engine area. This is achieved by an optimization with respect to a reference analysis performed for the purpose of structural life prediction which is part of the engine development process. The reference analysis is usually a finite element calculation applied to a representative flight profile (the design mission). The parameters of the algorithm are determined in such a way that the maximum deviation between the results of the algorithm and those of the reference analysis is minimized over the whole design mission. The algorithms are suitable for use in real time. They allow for fast transition of the input signals, measured under real aircraft and engine manoeuvring, to the calculated temperatures and stresses.

From the temperature and stress histories of a flight, the cyclic damage is determined by application of the crack initiation or crack propagation data. Life consumption results are available immediately after the end of a flight. They are measured in units of consumed reference cycles which are defined by the most damaging cycle of the design mission. The remaining operational life of each critical area is obtained by a comparison with the respective number of reference cycles of the released service life.

MTU Aero Engines GmbH, Dept. TPM,
P.O. Box 50 06 40, 80976 Muenchen, Germany.

This method is implemented in the on-board life usage monitoring system OLMOS of the German Airforce Tornado fleet which has been in use since 1987. It has been improved and extended since then by periodical updates according to the experiences made and to the newest developments like the extension from crack initiation to crack propagation monitoring. The current presentation reports some experience gained with this monitoring system and outlines the mathematical models used.

2 INTRODUCTION

In-service monitoring of fracture critical aero engine components requires efficient algorithms with a simple structure. The algorithms must be able to quickly process the incoming signals. A data sampling frequency of about 2 Hz is necessary to catch all load cycles including their peak values. Therefore, the calculation of temperatures and stresses has to be performed within every half second.

The free parameters of the algorithms are fitted optimally with respect to finite element calculations performed for a representative flight mission profile. A good quality of the algorithms is only achievable if the structure of the algorithms - although simple for a quick evaluation - well reflects the essential physical behaviour of the considered system. Under this assumption it can be expected that the algorithms are able to model the physical behaviour also for arbitrary flown missions within sufficient accuracy.

Beyond their application for on-board life usage monitoring in real time, the algorithms also serve for the on ground evaluation of recorded real flight data. In the design stage of an engine the prediction of the usable life time of newly developed engine components is of great interest. Normally, only the design mission is calculated by the finite element method. Real missions can be different according to the individual tasks of the aircraft. So the design mission is not representative in many cases. Here, the monitoring algorithms can be applied. This is done at MTU by the use of a mission analysis program based on the algorithms. With this program it is possible to analyse a great number of recorded flights in a short time.

3 DATA BASE FROM FINITE ELEMENT REFERENCE ANALYSES

For the engine life prediction which is part of the entire engine development process, finite element structure calculations are carried out to determine the temperature and stress histories for a predefined design mission. This mission is characterised by given rotational speeds and engine intake conditions as functions of the time. The mission is assumed to be representative for a typical real flight mission with respect to the life consumption of the engine components. The subsequent damage calculation reveals the structure locations with the highest life consumption, and is a basis for the declaration of those critical areas of the structure which are to be monitored. Although only the stresses in these areas need to be calculated by the monitoring system, the temperatures have to be calculated also in some additional structure points. These points are selected in such a way that a good approximation of the thermal stresses at the critical areas by the monitoring algorithms is possible, and they must be dense enough to allow for an adequate modelling of the heat conduction.

The stress and temperature histories in the selected points of the structure, obtained by the FE design analysis, serve as a data base for the tuning of the monitoring algorithm parameters.

4 GAS PATH TEMPERATURES

Some representative gas path temperatures are needed as heat sources for the calculation of the metal temperature development in the structure. They are either measured, if possible, or modelled, based on the performance design calculations. In many cases it was found out that a model of the form of a power function

$$T_{gas}^{*} = a + b \cdot (N_{rot}^{*})^{c} \qquad (1)$$

is sufficient, where T_{gas}^{*} and N_{rot}^{*} are the thermodynamically reduced gas temperature and rotational speed (normalized to ISA standard conditions). The parameters *a, b* and *c* are fitted according to the individual gas temperature from the design performance calculation.

For each temperature point one gas temperature is used for the calculation. If for example the temperature points of a compressor constisting of several stages with a long extension in axial direction are considered, and the compressor inlet and outlet temperatures $T_{gas,1}$ resp. $T_{gas,2}$ are available, a linear combination of these two temperatures of the form

$$T_{gas} = (1-c) \cdot T_{gas,1} + c \cdot T_{gas,2} \qquad (2)$$

can serve for different points with individual c-values according to the point position.

5 METAL TEMPERATURES

The metal temperature development requires a dynamical model. The model used in OLMOS is a discrete dynamical system and can be thought as derived from Fourier's heat conduction equation by discretisation with respect to the time and to the spatial coordinates. The temperatures in each considered point are calculated successively in small time steps (for example steps of 0.5s), starting with an initial temperature distribution. The temperature at time step *i* follows from its value at the previous time step *i-1* in the following way.

To every temperature point a leading temperature T_{lead} is assigned which depends on the selected gas temperature T_{gas}:

$$T_{lead}^{(i)} = T_{lead}^{(i-1)} + a \cdot (b \cdot T_{gas}^{(i)} - T_{lead}^{(i-1)}) \qquad (3)$$

This leading temperature T_{lead} and the temperatures of (normally three or four) neighbouring points T_j determine the increment of the metal temperature of the considered point:

$$T^{(i)} = T^{(i-1)} + c \cdot (T_{lead}^{(i)} - T^{(i-1)}) + \sum d_j \cdot (T_j^{(i-1)} - T^{(i-1)}) \qquad (4)$$

The second term on the right hand side represents the heat transfer from gas, the third term the heat conduction in the metal.

The parameters a, c, d_j are functions of the rotational speed N_{rot} , for example represented as power functions of the form $e + f \cdot (N_{rot} / N_{rot,ref})^g$ with a given reference speed $N_{rot,ref}$ and constants e, f and g. The values of these constants are determined by using the results of the design mission FE-analysis in such a way that the maximum temperature error between the model-calculation and the FE-calculation is minimized, taken over the whole design mission.

Instead of the dependency on the rotational speed N_{rot}, a more general ‘engine operating parameter’ could be introduced. For example, in [1] a combination of speed, pressure and gas temperature is used.

The parameter b is also dependent on the rotational speed. It is calculated for some (normally about five) selected speed values under the condition that the steady state metal temperatures for these speeds - which are known from FE-calculations - are met. The b-values for other speeds are linearly interpolated.

Note that the heat transfer from the gas to the metal point is represented only from the more or less remote gas path temperature T_{gas} which is delayed in time twice successively, by the effect of the parameters a and c, and which is distorted by the parameter b. Nevertheless, experience shows that these parameters are able to reproduce well the effect of the heat transfer. This capability was obviously gained by the optimization with respect to the design FE-analysis where the boundary conditions are accurately modelled.

Considering the stability of the temperature algorithms, it is important to avoid uncontrolled growth or oscillations of the temperature solutions. A stability analysis showed that by imposing of certain limits on the parameters these effects can be excluded.

6 STRESSES

The stresses for each critical area are modelled as the sum of assembly-, centrifugal-, and thermal-stresses:

$$\sigma = c_0 + c_{cf} \cdot N_{rot}^2 + \sum c_j \cdot T_j \qquad (5)$$

The thermal stresses are approximated as a linear combination of the temperatures in some representative points. The constants c_0 , c_{cf} and c_j are optimized by minimizing the maximum stress error over the whole design mission with respect to the FE-calculation.

7 LIFE CONSUMPTION ASSESSMENT

The two considered lifing concepts, the safe crack initiation concept and the safe crack propagation concept, are explained in detail in [2] and [3].

In life usage monitoring the released life of an engine component is formulated in terms of the maximum allowed number of reference cycles. Therefore, it is convenient to define the relative cyclic damage of an arbitrarily given stress cycle by the ratio of the damage of the cycle to the damage due to one reference cycle : D_{cycle} *: = (damage of cycle) / (damage of reference cycle).* Thus, the relative cyclic damage is expressed in units of *one reference cycle.* Application of this definition leads to different expressions for the crack initiation and for the crack propagation regime.

If N_{cycle} denotes the total number of a given stress cycle leading to crack initiation, a measure for the damage due to one cycle is introduced by its inverse $1/N_{cycle}$. In particular, the damage of one reference cycle is denoted by $1/N_{ref}$. The relative cyclic damage for the crack initiation phase therefore is $D_{cycle} = N_{ref} / N_{cycle}$. For the crack propagation phase, the damage of an arbitrary cycle can be quantified by the crack propagation rate $(da/dN)_{cycle}$, the damage of the reference cycle by $(da/dN)_{ref}$, yielding the relative cyclic damage $D_{cycle} = (da/dN)_{cycle} / (da/dN)_{ref}$.

Under the concept of crack initiation life, the damage accumulation process is assumed to be a linear process. The damage increments are independent from the current state of accumulated damage and the Miner’s Rule is used, i.e. the relative cyclic damage values of each extracted cycle are added up.

The damage accumulation process in the crack propagation regime - in contrast - is a non-linear one. The damage increment depends additionally on the currently accumulated stage of damage. The physical representation of the accumulated damage is the crack depth a, but in the terminology of life usage the number of consumed reference cycles N is used. The current stage of damage determines on one hand the value of the geometry function $g(a)$ and the term $K_{add}(a)$ representing the residual stress (see [3]), and on the other hand the crack growth increment of the reference cycle $(da/dN)_{ref}$ which is used as reference for the relative cyclic damage.

If the concept of safe crack propagation life is used to extend the life beyond the safe crack initiation life, then it is necessary that the algorithm switches from one procedure to the other controlled by the current stage of cumulated damage. To distinguish between both procedures, it is checked if the already consumed number of cycles is below or above the released number of cycles to crack initiation. If it is below, then the crack initiation damage process applies, otherwise the crack propagation process.

The life usage monitoring process for aero engines is strongly related to the history of an engine run [4]. Each engine run is treated separately and at its end all cycles are closed and the damage accounts are updated. Since an engine run can be considered as short compared to the life of fracture critical parts and the cracks at critical areas grow slowly, we can assume that the current state of damage is nearly constant during one engine run. Therefore it is justified to determine the values of the crack propagation damage dependent parameters only at the beginning of each engine run and keep them constant for all cycles of this run. According to [3], these parameters, the geometry function $g(a)$, the term $K_{add}(a)$**,** and the reference cycle crack propagation increment $(da/dN)_{ref}$, are given as functions of the accumulated number of cycles. Usually a representation in form of tables is used where the current values are obtained by linear interpolation.

The procedure of the flight damage calculation (more precise: of the damage calculation for an entire engine run) for crack initiation and propagation is sketched in Figure 1.

Cycle Extraction ($\sigma_{min}, \sigma_{max}, T$)	
crack initiation ⇗	⇘ crack propagation
equivalent **0 - σ_{max} cycle**	**ΔK, R**
⇓	⇓
	crack propagation law
S-N diagram ⇒ N_{cycle}	⇒ $(da/dN)_{cycle}$
⇓	⇓
$D_{flight} = \sum_{flight\ cycles} \frac{N_{ref}}{N_{cycle}}$	$D_{flight} = \sum_{flight\ cycles} \frac{(da/dN)_{cycle}}{(da/dN)_{ref}}$

FIGURE 1: FLIGHT DAMAGE CALCULATION SCHEME

For crack initiation, the extracted stress cycles are converted into equivalent zero-max-cycles, and their damage $1/N_{cycle}$ is determined by use of the S-N diagram. The total damage of the flight expressed in multiples of reference cycles is obtained by summation over the relative cyclic damage values $D_{cycle} = N_{ref} / N_{cycle}$ of each cycle.

For crack propagation, all extracted stress cycles are converted into stress intensity factor cycles using

$$K(a) \quad = \quad g(a) \cdot \sigma_{critical\ area} + \quad K_{add}(a) \tag{6}$$

with the geometry function $g(a)$ and the term $K_{add}(a)$. The corresponding crack propagation increments $(da/dN)_{cycle}$ are calculated by evaluation of the crack propagation law according to stress intensity factor range ΔK , R-ratio and temperature. Dividing the respective crack propagation increments $(da/dN)_{cycle}$ by the reference cycle crack propagation $(da/dN)_{ref}$ yields the damage increment $D_{cycle} = (da/dN)_{cycle} / (da/dN)_{ref}$ of the considered cycle in terms of multiples of the reference cycle. Damage increments are accumulated over the whole engine run.

In the final calculation phase - i.e. after the engine has been switched off - the damage accumulated over this engine run is added to the damage accounts.

8 SOME EXPERIENCE GAINED

In the German OLMOS system of the TORNADO engine RB199 the concepts of crack initiation and crack propagation life are combined, where the safe crack propagation life is used to extend the initially released crack initiation life. The experience made with the crack propagation algorithms for the rim slot fillet of a turbine disc, reported in [3], is summarised here. A main frame computer simulation was performed for a number of recorded real flight missions. It was found out that the cyclic damage of a flight is higher for the crack propagation phase than for the crack initiation phase. This can be explained because of the different effect of the subcycles. The subcycle damage in the crack propagation regime is higher due to different slopes of SN-curve and crack propagation law, and more damaging subcycles exist in the crack propagation regime since the crack propagation threshold is relatively lower than the endurance limit in the crack initiation phase. The ratio of the crack propagation damage to the crack initiation damage per flight for the analysed missions showed a scatter between 1 and 3.5 . The average damage per flight time for the crack propagation turned out to be greater by a factor between 1.5 and 2.0 compared with the crack initiation. Calculations showed that the service period in terms of engine flight time can be increased by about 40%, if in addition to the crack initiation regime a safe percentage (up to 2/3 dysfunction life) of the crack propagation regime is utilised.

By the analysis of a great number of recorded flight data using the monitoring algorithms, the inadequacy of the traditional method of life usage monitoring by using flight time and β-factors (damage per hour) instead of individual on-board monitoring, can be made obvious by Figures 2 and 3. Figure 2 provides an example how life consumption is really related to the corresponding flight time. Every data point in this figure represents one flight. It is easy to see that the correlation of flight time and cyclic life consumption is very weak. Figure 3 shows the frequency distribution of relative life consumption. A scaled depiction has been chosen to allow for the inclusion of all monitored areas of the engines. Life consumption is shown relative to the mean value. The data describe the typical operational life of a whole fleet. The curve shows a wide scatter (around one order of magnitude). From that scatter it becomes clear that a measure of life consumption using flying hours and β-factors would not be adequate [5].

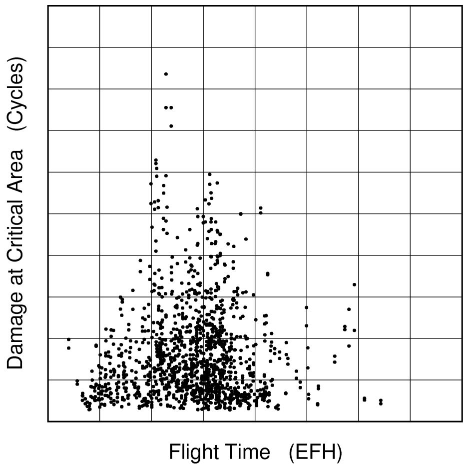

FIGURE 2: LIFE CONSUMPTION VERSUS FLIGHT TIME

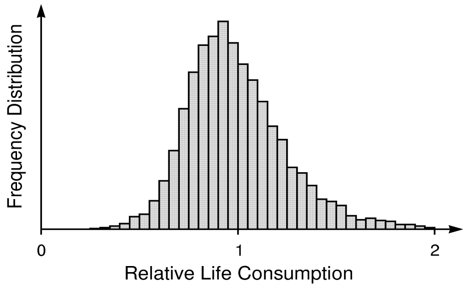

FIGURE 3: RELATIVE LIFE CONSUMPTION

It was observed by mission analyses for several aircraft that some monitored critical areas of both engines in one aircraft consumed significantly different amounts life although – obviously – both engines had flown the same mission profiles. Detailed investigations revealed that the only remarkable difference in engine operation was that generally the right hand engine was started first (around 10 minutes prior than the left hand engine) and also shut down first (around 4 minutes before the other one). It was suspected that the resulting differences in warming up and cooling down times (with the engines in idle) could be the reason. In a systematic analysis, calculations with identical mission profiles but varied periods between engine start and take off respectively landing and engine shut down were conducted which could clearly confirm this assumption. As an example, the variation of life consumption with varying warming up and cooling down times is shown in Figure 4 for a typical critical area. Generally it can be observed that shorter times between engine start and take off or between landing and engine shut down - although desired to reduce fuel consumption - have a detrimental effect on life usage. It is evident that more than a factor of two in life consumption can be caused. Based upon this investigation, recommendations for optimal warming up and cooling down times could be given.

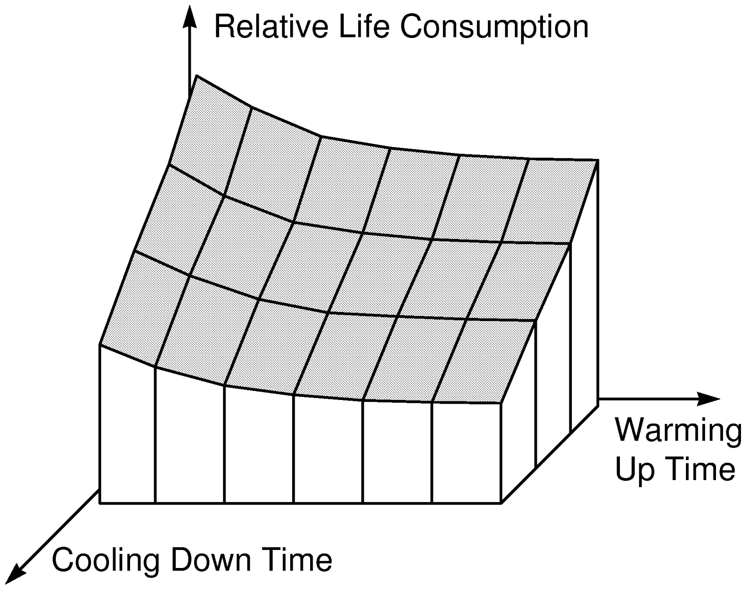

FIGURE 4: INFLUENCE OF WARMING UP AND COOLING DOWN TIME ON LIFE CONSUMPTION

One last point of experience should be mentioned. Even the best algorithms have only limited worth if the data management aspect, in particular regarding the change of engine parts, is neglected. This topic is also discussed in detail by Hall [6]. In some cases of data mismatch it might be possible by statistical considerations in connection with an application of the monitoring algorithms on typical flown missions to reconstruct the lost data. Figure 5 shows an example of the computed life usage for one critical area in the bore region of a turbine disc, including all flying engines and also spare parts [7]. The two solid lines show the results of a statistical fleet simulation performed for the engine variant "1". In case of lost damage data for a specific module, it is possible to obtain a conservative substitute damage by taking the value of the upper solid line for the specific accumulated flight time.

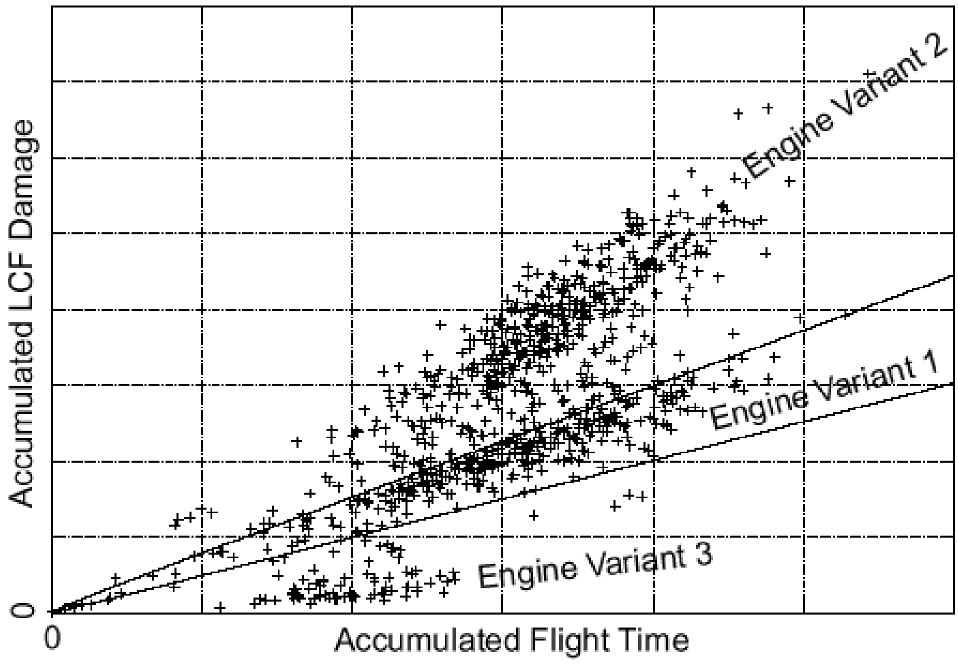

FIGURE 5: ACCUMULATED DAMAGE FOR TURBINE, DETERMINED BY FLEET-WIDE MONITORING

9 REFERENCES

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