Low-Cycle Fatigue and Creep Analysis of Gas Turbine Engine Components

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Factors that determine the damage of aircraft engines due to low-cycle fatigue and creep during service are numbers and rate of transient acceleration and deceleration, operating ambient temperature, time at temperature, and amount of time at over temperatures. The purpose of this paper is an attempt to define a method for predicting this engine damage and establishing the critical engine parameters to be monitored in conjunction with either an on-board computer or a suitable recording system that can be used on a central computer at the conclusion of a flight. The scope of the paper is limited to an analytical study of a typical fan engine to show the important engine operating parameters leading to limiting the useful engine life due to 1) low-cycle fatigue in the fan turbine disk and 2) combination of low-cycle fatigue and creep in the high-pressure turbine blades and vanes. No attempt was made to analyze the propagation of cracks, the assumption being that initiation of a crack defines a failure. For the purpose of analysis, a typical aircraft mission profile was chosen. Transient data (spool speeds, gas temperature, and pressure, etc.) required for the low-cycle fatigue study were generated by the simulation of engine thrust transients under various flight conditions. Metal temperatures for the various components were next determined by means of a finite difference network analysis program that included transient and steady-state heat transfer by conduction, convection, radiation, and film cooling. The stress-strain analysis was obtained by various special purpose finite element programs. Leading edge of both the high-pressure turbine blade and inlet vane was found to be a critical element. The analysis also shows that stress concentration due to the presence of cooling holes in the blade, which is a common feature in present high-performance cooled engines, should be examined in detail for low-cycle fatigue failure. Notch effects at fir-tree connections between turbine blades and disks also represent potential low cycle fatigue problem area. Bolt holes were found to be the most critical stress concentration area in the fan turbine disk. Holes of this type are common in multistage turbine disks to allow the tie bolts to penetrate for disk attachment. Various engine parameters that control the low cycle fatigue damages in these locations are identified and discussed. To demonstrate the effect of ambient temperature on the low cycle fatigue life of a component, the high pressure turbine inlet vane was chosen as an example. It is shown that the low-cycle fatigue damage corresponding to a rate of power lever movement is a function of the ambient temperature as well as the fuel control system of the particular engine. Areas which need further research to achieve the desired goals are also discussed.

I. Introduction

TWO maintenance problems that exist during fleet operation of aircraft are 1) determining the time at which engines should be removed from service for inspection and overhaul and 2) determining which parts should be replaced during overhaul. The "yardstick" for engine removals has usually been hours of operation rather than the extent of "damage" incurred during service period. Factors that determine the damage due to low cycle fatigue and creep during service are numbers and rate of transient acceleration and deceleration, operating ambient temperature, time at temperature, and amount of time at over temperatures. For the purpose of this paper, the term "low-cycle fatigue and creep damage" will be used to denote crack nucleation and early crack growth (i.e., crack initiation) at the critical locations of any component. Although several NDT techniques are currently available for detecting cracks during standard maintenance inspections, they are generally limited by their sensitivity to detect very small cracks that are of interest in this paper. The purpose of this paper is an attempt to define a method for predicting this engine damage and establishing the critical engine parameters to be monitored in conjunction with either an on-board computer or a suitable recording system that can be used on a central computer at the conclusion of a flight.

The scope of the present paper is limited to an analytical study of a typical fan engine to show the important engine operating parameters leading to limiting the useful engine life e to 1) low-cycle fatigue in a fan turbine disk and a disk-to-blade fir tree attachment in the high pressure turbine and 2) combination of low-cycle fatigue and creep in the high pressure turbine blades and vanes. The paper does not include methods to improve low-cycle fatigue life of a given design or material testing to evaluate its low cycle fatigue behavior. Existing data and methods for predicting cycles to initiate a crack were used. No attempt was made to analyze the propagation of cracks, the assumption being that initiation of a crack defines a failure.

II. Mission Profile

For the purpose of analysis, an aircraft mission profile, as shown in Fig. 1, was chosen. The six second acceleration and deceleration in the mission profile denote the time taken to move the power lever from the position of idle to the full take off power position and vice versa.

III. Method of Analysis

The analysis was carried out in four steps:

1) Transient and steady-state data (e.g. rotor speeds, gas temperature, and pressure, etc.) required for the low

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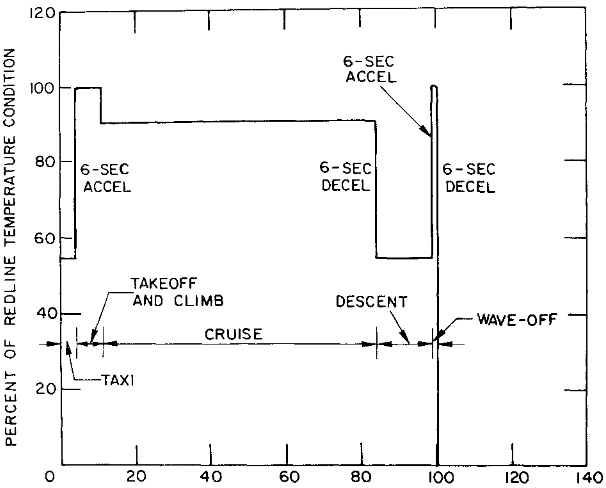

Fig. 1 Mission profile.

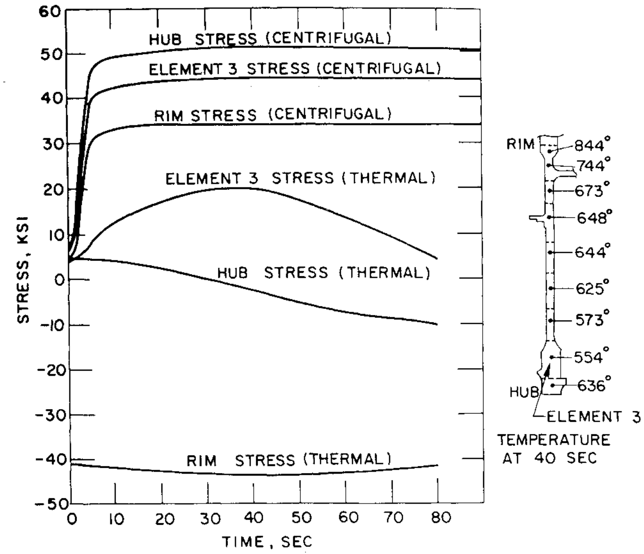

Fig. 2 Fan turbine disk tangential stress during acceleration.

cycle fatigue study were generated by the simulation of engine thrust under various flight conditions. The transients were obtained by operating an engine thermodynamic model, capable of transient operations, coupled to a basic model of the fuel control system.[1]

2) Metal temperatures for the various components were next determined by means of a finite difference network analysis program that included transient and steady-state heat transfer by conduction, convection, radiation, and film cooling. The program performs fluid network pressure drop calculations including flow area changes, friction, entrance and exit losses, heat addition, and centrifugal pumping for a given flow distribution.[1]

3) The stress-strain analysis was carried out by various special purpose elastic-plastic finite element programs. Transient and steady-state creep analysis of a blade or vane section was performed by a finite element program based on the beam theory. The program has the capability to redistribute the stresses at each time increment caused by variation in creep rate due to chordwise nonuniform temperature and stress distribution.[1]

4) Finally, from the stress-strain analysis of the previous step, the critical elements in each component for low cycle fatigue and creep, were identified. The low cycle fatigue life was computed using Manson's Equation (2) as follows:

$$\Delta\epsilon = (3.5/E)\sigma_{ult}N_f^{-0.12} + D^{0.6}N_f^{-0.6}$$

where $D = \log_e (100/100\text{-}\%\ RA)$; $\Delta\epsilon$ = total strain range; σ_{ult} = ultimate tensile strength; % RA = per cent reduction in area at rupture; E = Young's modulus; N_f = cycles to failure.

The interaction between creep and fatigue is still an unsolved problem. For the purposes of this paper a linear cumulative damage law was assumed,[3] i.e., $N = 1/[1/N_f + t/t_R]$, where N = cycles to failure; N_f = cycles to failure due to low cycle fatigue; t = hold time at stress σ and temperature T, t_R = time to creep rupture at stress σ and temperature T. The analysis of fatigue crack initiation from an area of stress concentration was performed by the method of Neuber's hyperbola.[4]

IV. Results

All the major components of the engine were analyzed for low cycle fatigue and/or creep.[1] A few of the representative problem areas are reported here.

A. Fan Turbine Disk

The fan turbine consists of three stages of axial flow turbines. The disk considered here, designated as the fourth stage disk, is cooled by discharge air from the high pressure compressor on the upstream side and by discharge air from the low pressure compressor on the downstream side. Bending stresses caused by the pressure difference was taken into account. A detailed analysis, however, showed that stresses due to temperature gradient through the thickness of the disk were small and could be neglected. Because of its mass, the thermal response of the disk is slow. Figure 2 shows the variation in the tangential stress due to centrifugal and thermal loadings with time for some critical elements of the disk during the acceleration. A maximum tangential stress of 65,000 psi occurs at element 3 (the third from the bore) at 40 sec from the start of the acceleration. Although centrifugal loading is mostly responsible for this maximum stress, the temperature profile in the disk causes the maximum stress to occur at a point away from the hub. The radial stress at this location is 7500 psi causing an effective stress of 61,500 psi which is well below the yield strength of the disk material (Waspaloy) at the temperature involved.

The disk has 12 circular $^5\!/_{16}$ in. diam holes at a radius of 2.9 in. for the tie bolts, causing an effective stress concentration factor of 2.8 due to biaxiality of the stress

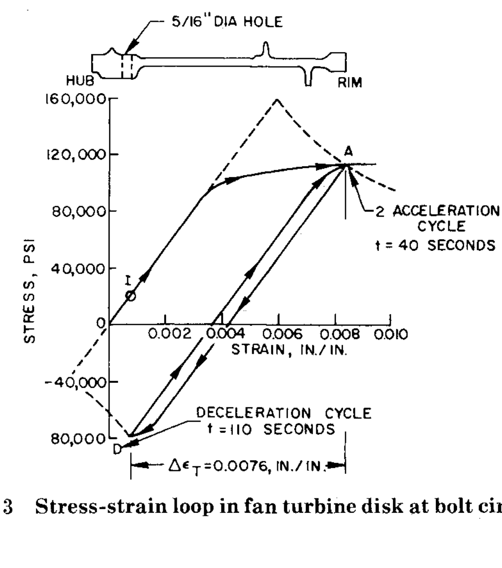

Fig. 3 Stress-strain loop in fan turbine disk at bolt circle.

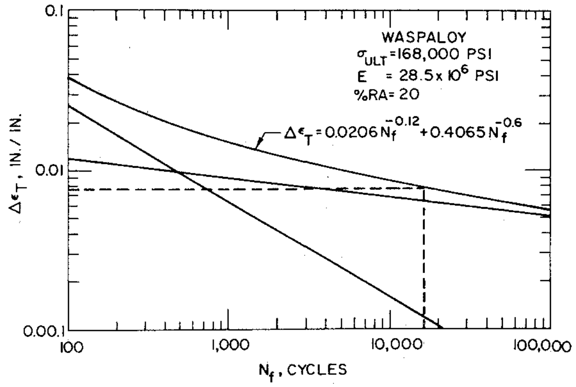

Fig. 4 Low-cycle fatigue life of fan turbine disk.

field. Figure 3 shows the stress-strain loop corresponding to a cycle consisting of a 6-sec acceleration followed by a 6-sec deceleration, as experienced by a disk element adjacent to the bolt hole. The total strain range encountered is 0.0076 in./in., and the life corresponding to this strain range is 16,500 cycles (Fig. 4). The crack initiation life for this turbine disk is thus dependent upon the rotor speed and the temperature and flow rate of the coolant air. The effect of creep on the low-cycle fatigue life of the disk is small at these operating temperatures.

B. High Pressure Turbine Inlet Vane and Rotor Blade

The high-pressure turbine is a single-stage axial flow turbine and is the hottest part of the engine next to the burner. The turbine blades and the inlet vanes are cooled by discharged air from the high pressure compressor. Typical cooling scheme for the high-pressure turbine vane is shown in Fig. 5. All cooling air is ultimately dumped into the main engine gas stream.

Inlet Vane

Since the mean section of the high-pressure turbine inlet vane was found to be most critical, the analysis of this section is reported here. From a low-cycle fatigue viewpoint, the leading-edge element, which responds rapidly during the transients, was found to be the most critical. Figure 6 shows how the difference between the leading-edge temperature, and the average section temperature varies during the transients. The maximum compressive and tensile thermal stresses occur at 3 sec after the start of acceleration and 4 sec after the start of deceleration, respectively. Bending stresses due to pressure differentials were found to be very small. Figure 6 also shows

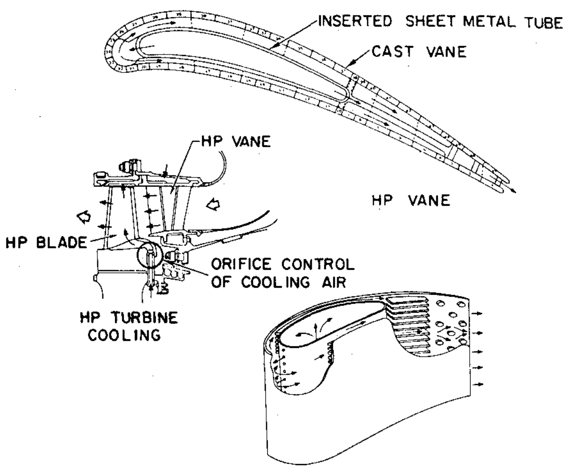

Fig. 5 Cooling scheme for high-pressure turbine inlet vane.

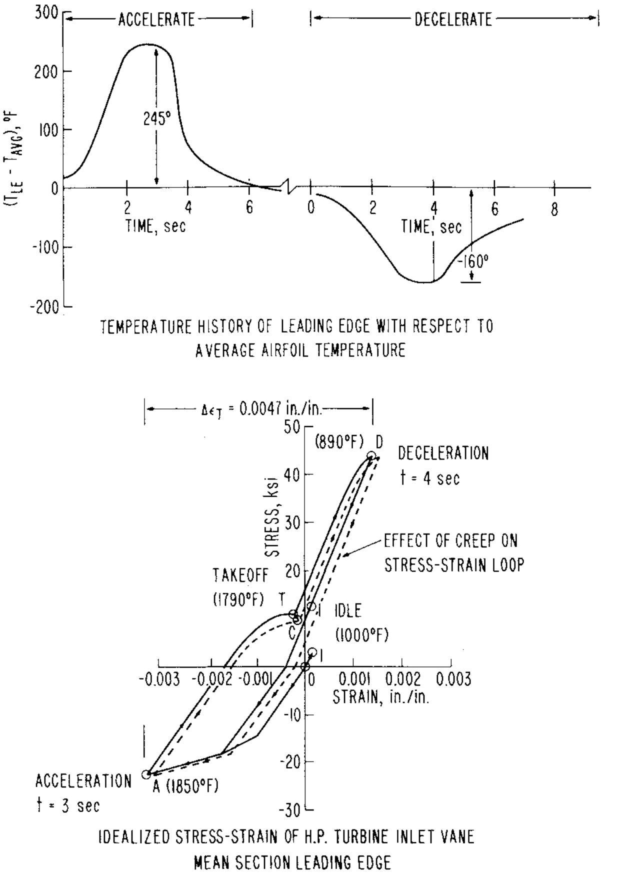

Fig. 6 Temperature and stress-strain history of leading edge of high-pressure turbine inlet vane.

the estimated stress-strain loop followed by the leading edge element during an acceleration-deceleration cycle. Note that the temperature of the leading edge changes from 890°F to 1850°F during the cycle. Ductility, ultimate strength, and Young's modulus of the vane material (WI 52) vary considerably over such temperature ranges. For example, the isothermal strain range vs cycles to failure at 900°F and 1800°F are shown in Fig. 7. Presently there is no method available for computing low-cycle fatigue life under such widely varying temperature history. For the purpose of the present paper, each cycle was treated as consisting of a hot reversal and a cold reversal at the same strain range. The number of cycles to failure at a total strain range of 0.0047 in./in. are 6700 and 11,000 at temperatures 1800°F and 900°F, respectively. Total number of cycles to failure is therefore estimated to be

$$N_f = 2/(1/6700 + 1/11{,}000) = 8340$$

Creep Effect

Figure 6 shows by means of dotted lines the estimated effect of creep on the stress-strain loop. The assumption is that creep occurs only at steady-state take off. Since the net resultant force on the section is zero, the integrated creep strain over the whole vane section is small. Thus, the effect of creep can be considered as a translation of the stress-strain loop along the strain axis accompanied by relaxation of the leading edge stress during take off. An estimate of the creep damage was obtained by carrying out a steady-state creep analysis of the whole vane section at takeoff point which showed that the leading-edge creep rate was 0.00002 in./in./hr. Each mission has 10 min of dwell at steady-state takeoff; thus the accumulated creep strain in the leading edge element per flight is 0.0000033.

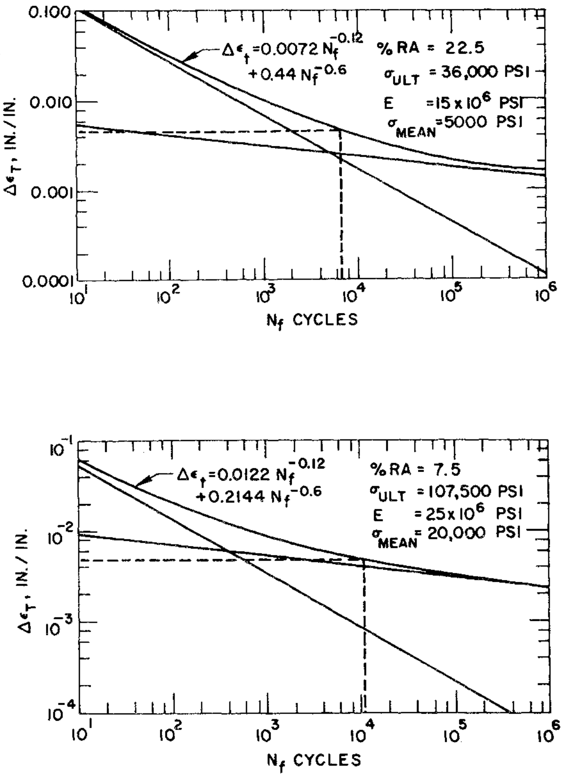

Fig. 7 Low cycle fatigue life for high-pressure turbine inlet vane.

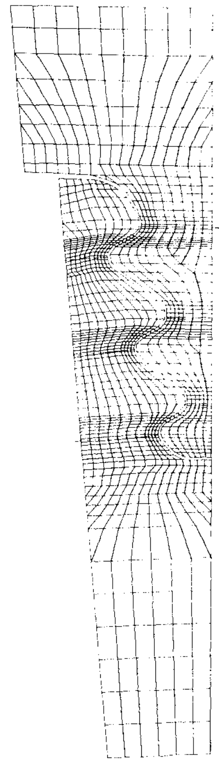

Fig. 9 Finite element model of disk and blade fir-tree assembly.

Assuming that the creep strain at rupture is 0.05, the number of cycles to creep rupture is 15,000. Using a linear damage law for creep fatigue interaction, the total number of estimated cycles to initiate a crack at the leading edge of the vane is

$$N = 1/(1/8340 + 1/15000) = 5360$$

The low cycle fatigue and creep damage in this high turbine vane is therefore controlled by the transient and steady-state gas temperatures and temperature and flow rate of the coolant air. Changes in pattern factor due to deterioration of the combustor can greatly affect the low-cycle fatigue life of an inlet vane. Operating inlet pressures are second-order effects in the low-cycle fatigue life of inlet vanes.

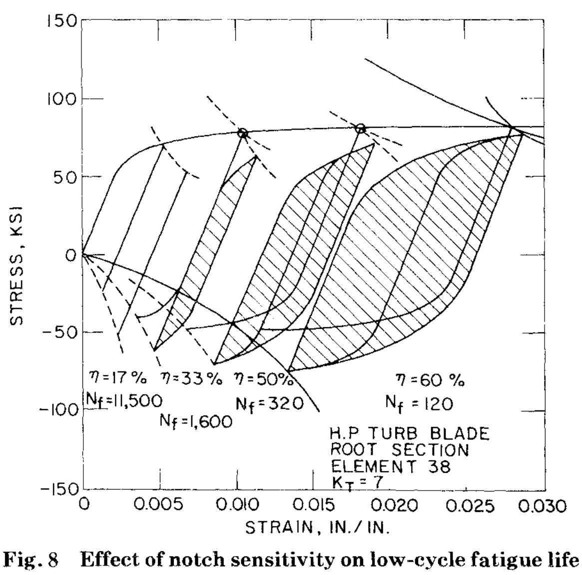

Fig. 8 Effect of notch sensitivity on low-cycle fatigue life.

Rotor Blade

A similar analysis for low-cycle fatigue and creep was carried out for the high-pressure turbine rotor blade. The leading edge was found to be a critical zone, and the crack initiation life at the leading edge was found to be dependent on the inlet gas temperature, temperature, and rate of coolant air flow and the rotor speed.

However, the most critical area for low cycle fatigue damage in the high-pressure turbine blade was located near a series of inclined circular cooling holes at the trailing edge causing a theoretical elastic stress concentration factor of 7 at the root section. Fatigue lives corresponding to constant cycling between idle and steady-state takeoff and for various notch sensitivity factors are shown in Fig. 8. The notch sensitivity factor n is defined as

$$n = (K_F - 1)/(K_T - 1) \times 100$$

where K_T = theoretical elastic stress concentration factor; K_F = fatigue stress concentration factor. The notch sensitivity is dependent upon the grain size of the material, notch radius, stress-state, and gradient, and often varies with the cycles to failure. It is evident that the fatigue life of the blade is strongly dependent upon the notch sensitivity. Testing of the blade material (IN 100) containing similar inclined holes under repeated bending indicated that this material has a notch sensitivity factor of the order of 20%.

C. High Pressure Turbine Blade to Disk Fir-Tree Attachment

The blade to disk fir-tree attachment in the high-pressure turbine is subjected to stress concentration effects and can be limited by low cycle fatigue. A plane stress elastic analysis of a section of the fir-tree was performed by means of a finite element program. Figure 9 shows the finite element model used to represent the complete blade to disk fir-tree attachment, and Fig. 10 shows a plot of the contours of constant effective stress at maximum takeoff loading, assuming that load is transferred from the blade fir-tree to the disk fir-tree at all three lobes. To account

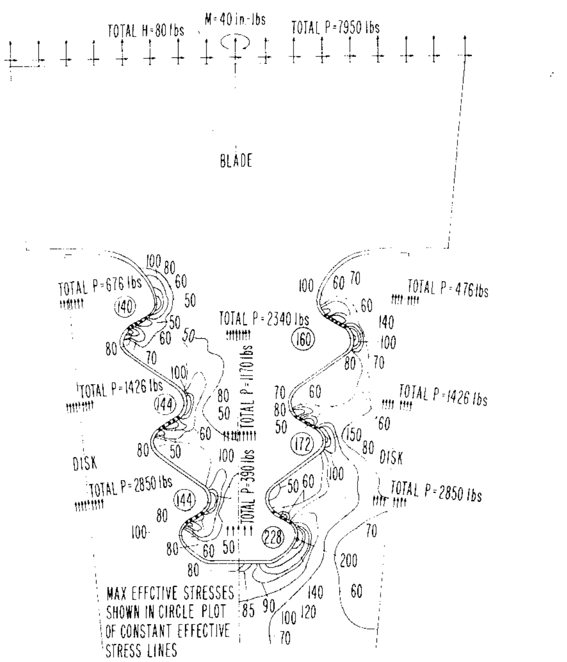

Fig. 10 Plot of constant effective stress contours for high-pressure turbine blade and disk fir-tree connection.

for local plasticity effects at notch roots, Neuber's method was used. The blade material is IN 100 with a minimum ductility of 3% at 1100°F, and the disk is made of U-700 having a ductility of 30% at 1100°F. Because of the large difference in ductility, the number of cycles to initiate a crack was calculated to be 4200 cycles for the blade fir tree and 15,000 cycles for the disk fir-tree. These numbers were based on the full elastic stress concentration factor. For a more accurate prediction, the notch sensitivity of the materials should be taken into account. Thermal stresses and creep were found to be of little importance, and consequently the low-cycle fatigue life of the blade to disk fir tree attachment is controlled mainly by the rotor speed.

It should be noted that the fir-trees become much more critical if contact between the blade fir-tree, and the disk fir-tree were restricted to one or two shoulders in which case the maximum stresses at the notch roots will exceed the present computed values significantly.

V. Effects of Rate of Transient and Ambient Condition

All the low-cycle fatigue calculations up until this point have been based on constant cycles consisting of 6-sec acceleration followed by 6-sec deceleration between idle and steady-state takeoff on an 86°F day at sea level. A real engine, on the other hand, experiences various rates of transients under various ambient conditions. To study the effects of transient rates and ambient conditions on the low cycle fatigue damage of an engine, transient data were generated for three different rates of accelerations on −60°F, 59°F, 86°F, and 120°F days at sea level and −20°F day at an altitude of 40,000 ft. Quantities like rotor speeds, turbine inlet temperatures were studied for each case, and the following observations were made.

Variations in the high-pressure turbine rotor speed with time for three different rates of acceleration of a −60°F day, and a 59°F day at sea level are shown in Fig. 11. Note that the rotor speed overshoots the steady-state takeoff value considerably for a 3 sec acceleration on a

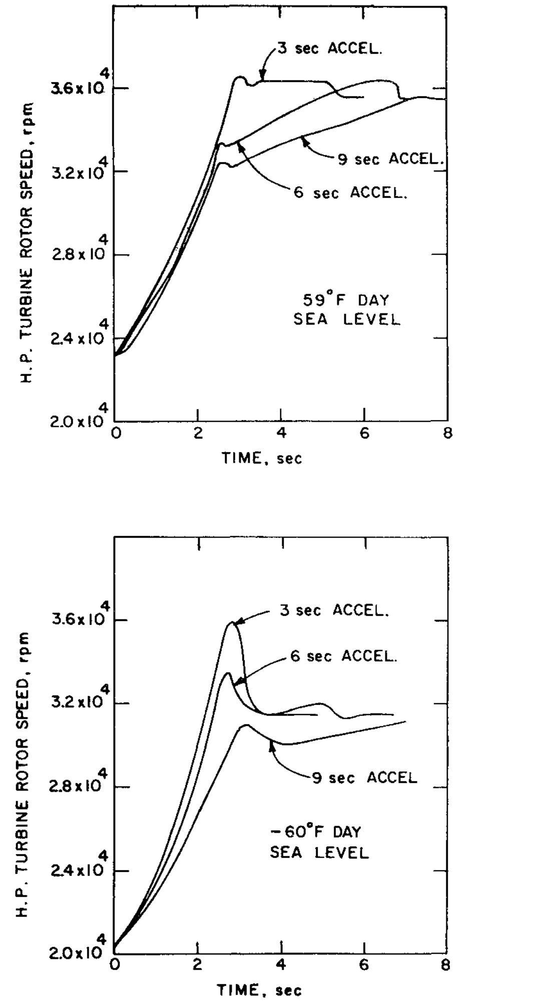

Fig. 11 Variation of high-pressure turbine rotor speed during transient.

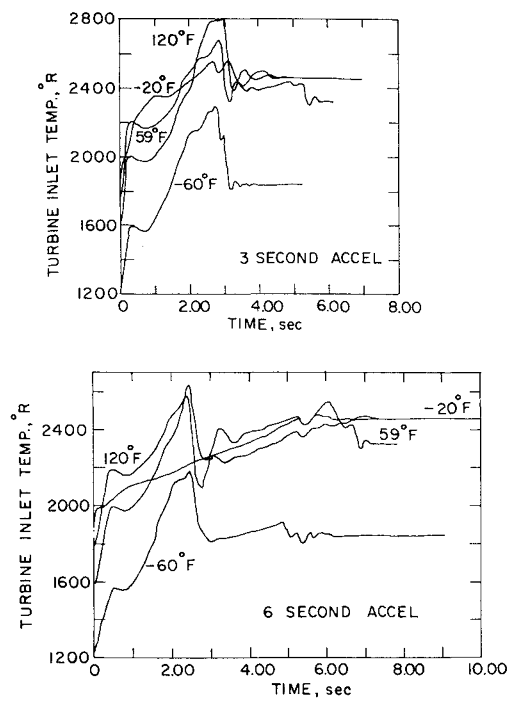

Fig. 12 Variation of high-pressure turbine inlet gas temperature during transient.

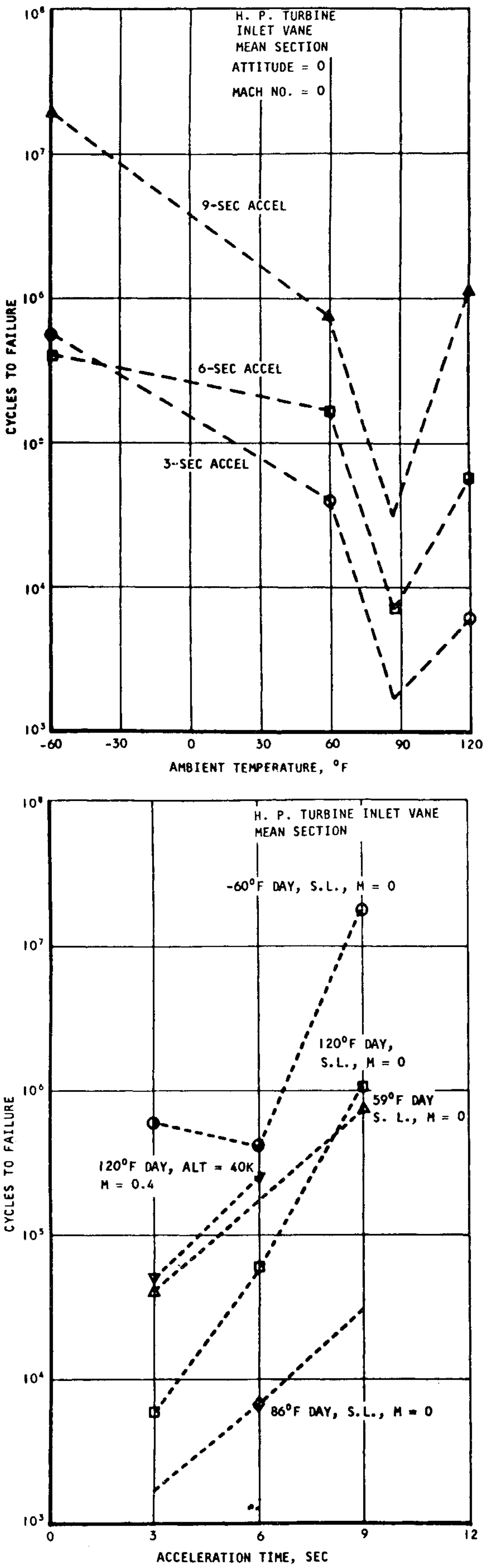

Fig. 13 Effects of rate of transient and ambient temperature on low-cycle fatigue life of high-pressure turbine inlet vane.

−60°F day. Since the centrifugal loading varies as the square of the rotor speed, such overshoots will affect the low-cycle fatigue damage of the high-pressure turbine blade to disk fir tree attachments considerably. The magnitude of overspeed reduces with increasing ambient temperatures and with lower rates of acceleration. However, the steady-state rotor speed increases with ambient temperature so that the creep damage on the high-pressure turbine blade is increased with increasing ambient temperature.

Figure 12 shows the variation in the turbine inlet temperature with time for two rates of acceleration at four different ambient conditions. At sea level there is considerable overshoot in temperature above the steady-state take off values for all ambient temperatures. Such over temperatures occur because the fuel control loop that controls the maximum high pressure turbine discharge temperature is essentially a steady-state limiter and is ineffective during a fast transient. At 40,000 ft altitude on a −20°F day, however, the rise in the turbine inlet temperature during the acceleration is much more gradual. Since the leading edges of both the high-pressure turbine inlet vane and rotor blade closely follow the pattern of the inlet gas temperature, while the average section temperatures are insensitive to such short over temperatures, any overshoot in gas temperature above the steady-state value will affect the low-cycle fatigue damage of the inlet vane and the rotor blade considerably. Also, in general, the turbine operates at a higher steady-state temperature on a hotter day, and consequently the creep damage encountered by the high-pressure turbine rotor blade, and inlet vane is larger the higher the ambient temperature.

To study how these various transients affect the low-cycle fatigue life of a component, the high-pressure turbine inlet vane was chosen as an example. Figure 13 shows the number of cycles to failure as a function of both the acceleration time and the ambient temperature. Notice that at sea level, for any given rate of acceleration, the maximum low-cycle fatigue damage occurs on a 86°F day, since the engine is flat-rated to an 86°F day at sea level. It is evident from Fig. 13 that the damage encountered by the inlet vane for any given rate of transient is highly dependent on the ambient condition. The exact dependence will of course vary from engine to engine and will be determined by the design of the fuel control computer.

VI. Discussions and Conclusions

The turbine section of an engine is the most critical in limiting engine operating life. Turbine low-cycle fatigue life is a function of the inlet gas temperature, rotor speed, and when cooled blades are used it is also dependent on the cooling flow system. Low-cycle fatigue damage is induced by the transient metal temperature distribution and rotor speed. Knowledge of steady-state temperatures and rotor speeds alone may not be sufficient to estimate the low-cycle fatigue damage because overshoots in temperature and rotor speed during rapid transients can result in significant low-cycle fatigue damage. The extent of creep damage, on the other hand, may be estimated from the steady-state temperature values.

To properly monitor the low-cycle fatigue damage of a turbine the following signal should be provided:

a) Transient rotor speeds, core engine air flow, and temperature inputs with responses fast enough to represent the rotor speeds, inlet gas flow rate, and temperature transients during jam accelerations and decelerations, are necessary. The temperature should be corrected for 1) normal combustor pattern factor to obtain maximum inlet temperature to the inlet vane, and 2) the relative gas temperature to the turbine blade. Methods for determining coolant air mass flow and temperature, either by direct or indirect means, must be developed in order that chordwise temperature profile on a cooled blade or vane may be computed. Determination of the temperature profile is essential for calculating low-cycle fatigue damage.

b) The creep and stress rupture life of a turbine blade is a function of mean blade temperature and average blade stress which is proportional to the square of the turbine speed. Direct measurement of average metal temperature (e.g., by means of infrared sensor) is attractive because it bypasses the uncertainties of computing metal temperatures from measured gas temperature.

Accurate analytical prediction for low-cycle fatigue and creep damage of actual turbine components is still beyond the state-of-the-art. Analytical procedures, such as indicated in the present paper, can at best provide a relative measure of the severity of damages encountered by the same engine under various loading histories. Coupled with extensive low-cycle fatigue and creep testing of critical engine components during engine development, these procedures might lead to an effective way of estimating engine damage incurred during the operation of the aircraft. Much research work along these lines is still needed.

VII. Recommendations for Future Research

Considerable research is needed in the area of material properties. The problem of computing life for a simple specimen subjected to a stress-strain cycle accompanied by large temperature variations needs further research. Coupled with this is the still unsolved problem of interaction between low cycle fatigue, creep, and corrosion. Considerable analytical and experimental research is necessary for devising dependable methods for predicting crack initiation life of an actual component based on the behavior of simple laboratory specimens under arbitrary stress-strain history. The problem of heat transfer in the presence of finite gas flow rate and large temperature difference between coolant and metal requires extensive study.

Much research is needed in methods of acquiring reliable and accurate data in flight. Transient data acquisition is very important from the viewpoint of low cycle fatigue. Direct measurement of metal temperature by means of infrared sensors might prove very useful for analyzing creep damage where obtaining true metal temperature level is of vital importance. In contrast, a greater error in temperature level may be tolerated for computing low cycle fatigue damage as long as the thermal gradient across the section is obtained within reasonable accuracy. Extensive research is needed for developing means of obtaining, in flight, the mass and temperature of both the core engine gas flow and the coolant air flow, either by direct or indirect means.

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