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**LA THÈSE A ÉTÉ
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A Damage Tolerance Approach for The Life Extension of J85-CAN 40/15 Compressor Disks

by

Gordon F. Aker

*A thesis submitted to the Faculty of Graduate Studies and Research
in partial fulfillment of the requirements for the
Degree of Master of Engineering*

Department of Mechanical and Aeronautical Engineering

Carleton University

Ottawa, Ontario

Canada

September 1986

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Abstract

As part of the NAE/NRC investigation into possibilities of extending the usable life of gas turbine engine components through a damage tolerance approach, low cycle fatigue crack growth rate testing was performed on compact tension specimens machined from retired J85-CAN 40 compressor disks. This was done to estimate a Return To Service interval (RTS) once the safe life of the component is consumed. An equation was developed to allow calculation of this interval on the basis of LCF crack propagation data. This equation also takes into consideration the relative magnitude of the stress intensity factors at the crack tip due to both the engine start up and shut down and major throttle excursions experienced during a typical mission hour. The RTS interval calculated using the most severe crack propagation rates determined was approximately 669 hours which compares favorably with the 555 hour interval supplied by the engine manufacturer. The LCF crack propagation rates of the compressor disk material were found to depend on both the disk service temperature and the disk service exposure time. A Direct Current (DC) potential drop technique was successfully assembled, calibrated and employed in this work with a resultant precision of 0.05 mm. The use of combined analysis based on using the DFM damage tolerance principles in conjunction with field experience for the life extension of compressor disks beyond their safe lives is also discussed.

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Chapter 1

Introduction

1.1 Present Gas Turbine Component Lifting Methodology

Traditionally, gas turbine engine manufacturers have used a lifting procedure known as the "initiation criterion" to determine the safe lives of gas turbine rotating components such as disks, spacers and shafts [1][2][3][4][5] [6]¹. These components are typically low cycle fatigue (LCF) life limited, the total fatigue life consisting of a crack initiation segment and a crack propagation segment as illustrated in Figure 1.1 [1][4][5][7][9][10]. The crack initiation life is loosely defined as the amount of time or the number of LCF cycles required to produce a crack of a size that is large enough for detection through currently available Non Destructive Inspection (NDI) techniques. As its name implies, the initiation criterion is concerned only with this portion of crack growth. The initiation criterion is used to determine the safe life of LCF life limited components in the following manner.

Low cycle fatigue crack initiation data is generated on either coupons of the disk material or by testing complete disks in a spin pit. The number of cycles required to produce a crack that is barely detectable, typically 0.8 mm (0.0315 in) long, is then recorded for a number of tests involving different "heats" or "batches" of the material used in the manufacture of the components. As is common in fatigue testing, there is a large amount of scatter inherent in this data, especially since different heats of the material are tested. This scatter is illustrated in Figure 1.2. In order to compensate for this scatter, a statistical analysis is performed to determine a lower bound for the data. This lower bound is determined by multiplying the standard deviation from the mean of all the data generated by a factor of three. This -3σ bound typically encompasses 99.9% of all the data generated. The number of cycles indicated by this lower bound is taken as the safe life of the component, as is illustrated in Figure 1.2.

¹ Numbers in brackets refer to references

In order to provide the engine user with a more meaningful life limit, a mission spectrum analysis is performed to determine the number of low cycle fatigue cycles which can be expected per mission flight hour. This analysis allows the safe life of a component to be specified in terms of either mission equivalency cycles or mission flight hours [4].

The initiation criterion is considered to be an overly conservative method of estimating the safe life of gas turbine rotating components [1][3][4][5][6][9] [10][11][12]. Firstly, it ignores the damage tolerance inherent in many of these components. Secondly, conservatism is associated with the statistical analysis that is performed on the fatigue initiation data to determine a lower bound. Since this lower bound encompasses 99.9% of all the data generated, the material and thus the component are assigned a fatigue life based on the minimum fatigue properties observed during testing. No allowance is made for the fact that the majority of the material tested has vastly superior properties. This analysis results in the retirement of a thousand components on the basis that one of those components may have developed an 0.8 mm crack. As a consequence of this methodology, 999 components out of every thousand are retired in a crack free condition when they may still have a significant amount of useful life remaining in them. In a study done by Pratt and Whitney, it was estimated that approximately 80% of the F100 engine components that are retired using the initiation criterion have at least 10 lifetimes remaining [4][5][9]. These results are illustrated graphically in Figure 1.3. The problem with this method of lifetime determination lies not with the statistical analysis itself, but in classifying all components as a single group when in fact they may exhibit a wide variation in material properties due to microstructural variations. Instead of applying the knowledge and experience to determine the status of all components on an individualistic basis and thus exploit them to their fullest extent, the initiation criterion attempts to cover all possibilities with an overly conservative lifing approach. The inadequacy of this approach is therefore associated with the use of a very large safety factor that results in a significant waste of material and money.

As modern gas turbines are designed to be more fuel efficient in civil aviation applications and attain higher performance levels in military applications, the necessary increases in compressor pressure ratio (CPR), and turbine inlet temperature (TIT), have made the use of expensive high strength, high temperature superalloys mandatory for rotating components [6] [7]. Use of these high cost materials has increased the interest in the development of a more cost effective lifing methodology. Another aspect worthy of consideration is the

limited availability of strategic materials thus making their more efficient use desirable [4][9].

Research and development work in the areas of fracture mechanics, damage tolerant design, mission profile monitoring and analysis and material degradation analysis when coupled with statistical data bases of user experience can now eliminate many of the unknowns and uncertainties associated with the initiation criterion approach. These advances have provided the means through which an improved gas turbine lifing philosophy can be developed without jeopardizing the safety of the aircraft or the crew. [1][2][3][4][6][10][11][13].

1.2 The Damage Tolerance Based Lifing Methodology

The major driving force in gas turbine development is the USAF Engine Structural Integrity Program (ENSIP). It was formulated to decrease the overall life cycle costs of gas turbine components by increasing their usable life without adversely affecting engine performance or safety. The program itself is very wide in scope, covering the design and development of engine components, a comprehensive engine monitoring program, an improved maintenance methodology and investigations into rejuvenation of service induced damaged components through heat treatment and hot isostatic pressing (HIP) [2][14][15][16]. Although the full economic benefits of such a program can only be realized by its complete implementation, the development of an improved maintenance methodology can be considered independently of the other aspects of the program. This thesis deals only with the development of an improved maintenance methodology based on damage tolerance concepts. Details of the other aspects of the program are available elsewhere [17].

It was desirable that a new maintenance methodology be developed to eliminate the obvious shortcomings of the initiation criterion approach as outlined in Section 1.1. This led to the development of the damage tolerance based maintenance methodologies. Subsequently, engine manufacturers such as Pratt and Whitney and Rolls Royce have been investigating this maintenance philosophy under programs called "Retirement for Cause" (RFC) and "Life on Condition" (LOC) respectively [1][2][3][4][6][7][9][10][11][13][18][19][20][22].

The philosophy behind the new lifing procedure is quite simple. Components remain in

service until a predetermined safe life limit has been reached. Once this design life has been exhausted, the engine is dismantled and the life expired components are inspected for service induced damage. This is accomplished by using current state of the art Non Destructive Inspection (NDI) techniques for macrodamage assessment, eg. through cracks, corner cracks etc., and/or by destructive testing of a few components for microdamage assessment. If a component is found to contain a flaw or LCF crack then it is retired from service for that reason. Unflawed components are returned to service for a period of time known as a "Return to Service" (RTS) interval either in "as is" or reheat treated condition. After this interval of service has been completed, the engine is once again dismantled and the process is repeated. Components are returned to service as long as they exhibit no signs of service induced LCF cracking or until the expected life remaining no longer warrants the expense of further inspections and reinstallations. This maintenance philosophy is illustrated in Figure 1.4. By following this procedure and treating each component individually, the inherent lifetime of each component is utilized more effectively.

1.3 The NAE/NRC Component Life Extension Program

The NAE/NRC program to develop an improved lifing methodology for low cycle fatigue life limited gas turbine components was initiated to establish a technology base from which a new lifing procedure could be formulated and implemented. This project was set up as a feasibility study to improve Canada's capabilities in this area and to initiate the development of the technology required to implement a new lifing procedure in the future. The future implementation of a damage tolerance based maintenance methodology could involve the General Electric F404 engine which powers the CF-18 Hornet [12][23].

The engines chosen for the development of the damage tolerance based lifing procedure and maintenance methodology were the General Electric J85-CAN 15 and J85-CAN 40 series of engines. These engines power the the CF-5 and CT-114 Tutor trainer aircraft respectively. These engines were selected for several reasons. First of all, the Canadian Armed Forces has many of these engines in use which have reached or are approaching the safe life limit specified by the manufacturer. This has provided a significant stockpile of retired components which can be used to develop statistically significant data bases as well as provide material for testing. Secondly, mission profile data had previously been obtained

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for both types of aircraft in order to assess a mission severity factor (MSF), which was used to compare the severity of the American and Canadian missions [23][24]. This information had previously been obtained and analysed at NRC and there was no need to repeat this part of the analysis for the present purposes.

The largest incentive to develop the maintenance methodology for this series of engines was the fact that a damage tolerance based maintenance procedure was prescribed by the engine manufacturer when they could not supply the required number of replacement components in time. In the procedure provided, the manufacturer specified that uncracked components could be returned to service for a maximum of one "Safe Inspection Interval" (SII) after which the component was to be retired regardless of its condition [2][12][23][24]. The duration of the SII varied according to the component as presented in Table 1.1. The events leading up to the implementation of this procedure are described in detail in Section 1.3.1 of this report. Although General Electric did not provide very much information regarding how they determined the specified SII, the fact that it was specified both confirmed the feasibility of doing the development work on this engine as well as providing ball park figures for verifying or comparing the NAE-NRC life prediction results. General Electric did confirm that a Weibull analysis of past failures and some mechanical testing was done to arrive at the SII values provided in Table 1.1.

1.3.1 The Historical Background of the Canadian J85 Engines

The General Electric J85 series of engines are single spool, turbojet power plants with an eight stage axial compressor and a two stage axial turbine. There are both afterburning and non-afterburning versions of this engine in use today. The J85 engine was originally designed in the 1950's as a missile and/or drone engine. Its design life for these applications was fifteen hours. It was then redesigned for aircraft applications and there are now over 13,000 engines of this type in operation around the world [25]. There are three variants of the J85 engine which are discussed in this study. The J85-5 is a 1736 kg, (3850 lb) thrust afterburning version of the J85 used in a twin mount in the Northrop T-38 Talon aircraft. This aircraft is used as an advanced trainer in the USAF and was also used in a flight aerobatic demonstration team, the Thunderbirds, until 1983. The Canadian version of this engine, the J85-CAN 15, is used in the Northrop CF-5 Freedom Fighter by the Canadian Armed Forces. It is also an afterburning engine mounted in a twin configuration. The

final version of interest is the J85-CAN 40 engine. This non-afterburning turbojet has a maximum thrust of 1195 kg (2633 lbf) and is used in the Canadair CT-114 Tutor in a single engine mounting. This aircraft is the basic jet trainer used by the Canadian Armed Forces and is also used by the Canadian flight demonstration team, the Snowbirds.

The J85-CAN 15 and CAN 40 engines were first purchased by the Canadian Armed Forces (CAF) in the 1960's. At the time of purchase there was no safe life specified by the manufacturer. General Electric first provided a safe life limit for the engine in 1970. The safe life limit specified to the CAF by G.E., was apparently determined on the basis of failures ie. cracked components, experienced by the American J85-5 fleet. Initially no provision was made for any variation in the mission severity between the two countries [23][24].

Two other discrepancies from the initiation criterion approach to component lifing were also apparent. First of all, G.E. defined the safe life on a time to dysfunction basis and not the more accepted time to "initiation criterion". They also departed from the procedure of specifying the probability of engine component failure as 0.1% or one in a thousand as outlined in Section 1.1 of this report. They instead employed a 0.135% or one in 741 component cumulative failure rate as the acceptable criterion. Both these variations have the effect of increasing the safe life specified while simultaneously reducing the safety factor inherent in the initiation approach. In summary, the components of the J85 engines were assigned a safe life by the manufacturer on the probability that one in 741 components would develop a crack large enough to warrant engine shut down [2] [12][23][24].

In 1973 the first evidence of cracking was noted in the fifth stage spacer of the Canadian engines. In 1976-1977 cracking was a major problem with the fifth and seventh stage compressor disks and their attached spacers. These cracks were discovered during routine maintenance inspections before the components had reached their specified design lives. The fifth stage compressor disk was one of the most crack prone components and had the lowest specified design life. For these reasons it was designated as the focus for this work. In order to clearly present the many changes in component safe lives as supplied by G.E., only the safe life of the fifth stage compressor disk will be presented numerically. It should be noted that similar variations in safe lives also occurred in the other rotating engine components.

The design life originally specified in 1970 for the fifth stage disk was 3000 hours as presented in Table 1.2. As was mentioned previously, the American and Canadian missions were assumed to be identical and thus the safe lives determined for the J85-5 engines were also applied to the J85-CAN 15/40 engines. In June of 1978, G.E. increased the specified safe life of the fifth stage compressor disk from 3000 hours to 4500 hours for unknown reasons. At the same time, the manufacturer completed a study to determine if there was any difference in mission severity between the two countries. They concluded that the Canadian mission was less severe than the American mission and thus assigned a mission severity factor (MSF) of 0.94 to the safe lives of the Canadian components. The mission severity factor was divided into the American safe life to obtain the Canadian safe life. The description of the mission profiles supplied to G.E. by the CAF for the purposes of this study remain unclear, nevertheless the application of the MSF increased the safe life of the fifth stage disk up to 4787 hours.

From June of 1978 to September of 1979, G.E. continued to change the safe life of the J85-5 components for unexplained reasons. The MSF was not changed during this period and the fifth stage of the Canadian engines was left with a 4680 hour safe life. This is presented in Table 1.2.

In mid November of 1981, G.E. drastically reduced the safe life limit specified for the fifth stage disk in the J85-5 engine. The reasons for this drastic reduction are uncertain, however it is surmised that it may have been the result of an increased incidence of cracking in the J85-5 engines [2][12][24]. The lifetime of this disk was reduced from 4400 hours to 3200 hours.

Simultaneously, G.E. revised its estimation of the Canadian mission severity. The techniques used in this evaluation were not made available to NAE although it was possibly a result of a Weibull analysis of the Canadian failure data. The MSF was thus revised from 0.94 to 1.8. This increase in MSF when combined with the reduction of the American J85-5 safe life resulted in a reduction of the Canadian safe life from 4680 hours to 1777 hours.

It was this drastic reduction in the specified safe life that resulted in the implementation of the retirement for cause type of maintenance procedures. At the time G.E. revised the fifth stage disk safe life from 4680 to 1777 hours many of the engines in the CAF inventory had been in service for well over the 1777 hours defined as the safe life. This gave the

CAF the option of either grounding the CF-5 and CT-114 fleets or getting replacement parts from G.E. The latter option was obviously preferable, however G.E. required at least twenty months of lead time for this type of order. They were also faced with the shortage of replacement parts for the much larger USAF T-38 fleet. In order to allow the CAF to continue operations until the parts could be supplied, G.E. proposed the implementation of damage tolerance based maintenance by specifying a Safe-Inspection Interval for parts which had exceeded the revised safe life limits. From here on this shall be referred to as the General Electric J85 Continued Service Program.

The G.E. Continued Service Program consisted of supplying the CAF with the fracture critical locations on the components and the critical crack lengths. They also supplied safe inspection intervals for all the components in question. They stipulated that: a) components should be re-introduced to service for a maximum of one safe inspection interval and b) that cracked parts should never be reinstalled. The safe inspection interval specified for the fifth stage disk was 555 hours. The SIIs specified for the other components are listed in Table 1.1. G.E. would not reveal how they determined the SIIs. However, they did indicate that fracture mechanics and spin pit testing had been carried out to verify their predictions [24].

In June of 1983, the CAF was advised by G.E. that the continued implementation of a safe inspection interval was unsafe and that the continued service program should be stopped. By July of 1984, G.E. had revised the MSF once more, this time lowering it from 1.8 to 1.4. This increased the design life of the fifth stage disk from 1777 to 2285 hours as presented in Table 1.2. They were also supplying sufficient replacement parts with an improved LCF configuration such that the continued service program was no longer required. Presently, J85 components are once again being retired because a safe life limit has been reached, and not because of the condition of the components.

There were several observations made by NAE/NRC and the engine maintenance contractor, Orenda, after the continued service program had been completed. First of all, compressor and turbine disks were used well beyond the safe life limits supplied by the manufacturer. Secondly, it was apparent that the components were returned to service for more than one Safe Inspection Interval despite the recommendation by G.E. that only one extension interval should be used. As well, although the dysfunction crack length for the fifth stage disks was supplied by G.E. as 4.5 millimeters (0.177 in), disks were operated

with cracks exceeding this limit with no apparent harm to the engine. The longest crack discovered in the retired disks was 19 millimeters (0.75 in) in length [23][24].

Another aspect worthy of note is that there was no record of cracking in the first to fourth and eighth stage compressor disks and in either of the two turbine disks. The reasons for this are not clear. The fact that these components were retired in an uncracked condition indicates that they had usable residual life remaining at the time of their retirement. Table 1.3 illustrates the average life extension realized through the CAF implementation of the continued service program. While some of the longer life components had been used for over three and a half times the stated design life or safe life, on an average, the components were used for almost twice the stated safe life. This life extension is obviously significant and one could expect this extension to be even greater for a component that is originally designed on the basis of damage tolerance concepts.

As previously mentioned, this work was initiated as a study to determine the feasibility of developing a damage tolerance based maintenance methodology in Canada. This study could then be responsible for the development of technological and statistical data bases from which a comprehensive lifing program could be developed and implemented. This is desirable for two reasons:

- 1) to decrease the life cycle management costs of the future engines acquired by the Canadian Armed Forces.
- 2) to ensure that the possibility of grounding the Canadian Armed Forces aircraft fleet due to questionable lifing procedures based on the engine manufacturers recommendations is minimized.

It is important to realize that even if it may not be feasible to implement a damage tolerance based lifing philosophy in older engines due to financial, safety or technology limitations, the type of work performed on the J85-CAN 40/15 engines has increased our understanding of how materials degrade during service, how components fail and how these failures can be usefully analysed. This information can in turn be recycled back to the designers of new components who can attempt to increase their reliability and safety in newer engines.

Chapter 2

Evaluation of the RTS Interval

2.1 Theory and Approach

The development of a damage tolerance based lifing procedure centres around the determination of a safe Return to Service (RTS) interval and the development of accurate and reliable Non Destructive Inspection (NDI) techniques. This study is concerned with the former while the latter is examined in references [1][4][13][19][23][24].

As can be seen from Figure 1.3; the RTS interval is obtained by determining the number of cycles or amount of time required to propagate a crack from a size " a_i " to a predetermined dysfunction crack length " a_{dv} ". There are three common approaches used to evaluate the RTS interval. These are the Probabilistic Fracture Mechanics (PFM) approach, the Deterministic Fracture Mechanics (DFM) approach and the Combined Analysis (CA) approach. These approaches are used in order to avoid failures resulting from the overlapping of the probability distributions for the applied load or stress with the probability distribution representing the variation in the magnitude of a material property. This concept is illustrated in Figure 2.1. As can be seen from this illustration, the cross hatched portion of the curve signifies the intersection of the material strength distribution and the applied stress distribution and thus represents the probability of failure for this system. As mentioned previously, the PFM and DFM approaches have been developed in order to avoid failures resulting from this overlap [23].

2.1.1 The PFM Approach

The PFM approach differs somewhat from more "conventional" approaches in that specific magnitudes for the various parameters required to determine a RTS interval are not used. Instead, the parameters are evaluated in the form of probability distributions from which a probability of component failure or component reliability is determined [23][24]. For example, instead of specifying an initial crack size " a_i ", an occurrence probability distribution of various crack sizes is used. Similarly, the operating stress is not taken as the

maximum observed value but rather as a distribution of stress magnitude with occurrence. The required distributions are defined by their median and standard deviations and are combined to provide the probability of failure as a function of time. This function can then be used to evaluate the RTS interval by defining a low but cost effective failure probability. More detailed information regarding the PFM approach is available in references [7][8][23][24].

There are several disadvantages associated with the PFM approach. Although theoretically by employing the PFM approach, one is obtaining the maximum possible safe life of each component, in actual application the approach is very costly to implement. This is due to the extensive testing which must be performed in order to obtain a statistically significant data base from which the appropriate distributions can be evaluated. Similarly, there is still concern that many of the assumed distributions may be unrealistic and/or incorrect which either increases the risk involved with the maintenance program or results in reduced component utilization. Due to the extensive testing required to implement the PFM approach, it was not applied in this study.

2.1.2 The DFM Approach

The Deterministic Fracture Mechanics approach for the development of a damage tolerance based maintenance methodology represents a "worst case" analysis. In the DFM approach, an initial crack is assumed to be present in the component at a critical location. As well, the crack is assumed to be the largest possible size that could go undetected during an inspection. In order to estimate how fast such a crack will propagate under normal operating conditions, stress analysis is performed to determine the cyclic and steady stresses at the crack tip. Under the DFM approach, the stress analysis is performed assuming the component is under realistic operating conditions and only the maximum observed stresses are employed in the subsequent analysis. Fracture mechanics principles are then employed to determine the crack tip stress intensity factor, ΔK , in the component. The crack tip stress intensity factor, K , is proportional to the applied stress, σ_a and the square root of the crack size which yields:

$$K \propto \sigma_a \sqrt{\pi a} \quad (2.1)$$

Similarly, in fatigue the cyclic stress is applied in order to determine the cyclic stress

intensity factor ΔK , or

$$\Delta K \propto \Delta \sigma_d \sqrt{\pi a} \quad (2.2)$$

Once the cyclic stress intensity factor has been determined it can be applied to the Paris crack growth rate relationship which is a power law of the form:

$$\frac{da}{dN} = C(\Delta K)^n \quad (2.3)$$

where da/dN is the crack growth rate, ΔK is the cyclic stress intensity factor and C and n are material constants. In order to be consistent with the Deterministic approach, the maximum crack propagation rate observed is used in the analysis.

The crack propagation interval, (CPI), assuming a constant ΔK condition, can then be evaluated using the crack growth rate, the assumed initial flaw size, and the maximum acceptable crack length or dysfunction crack length as follows:

$$CPI = \frac{a_{dys} - a_i}{C(\Delta K)^n} \quad (2.4)$$

It should be noted however that the Paris Law used in the above equation is only applicable to the linear or secondary portion of the fatigue crack growth rate curve as plotted on log-log axis. However assuming that the crack growth rate experienced by a cracked component in service is a linear function of the stress intensity factor, (when plotted on log-log axis), provides a conservative estimate of crack growth rates for stress intensity factors below this linear region in the primary crack growth region. Similarly, by defining the dysfunction crack length as some fraction of the critical crack length, a_{crit} , and thus K_{crit} , the non linear tertiary region of crack growth will not be reached. These three regions of fatigue crack growth are illustrated in Figure 2.2. It is also vital to realise that Equation 2.4, assumes that ΔK is constant across the entire crack propagation length. This is of course incorrect given that the stress intensity factor ΔK is a function of crack length as illustrated in Equation 2.2. The variation of ΔK with crack length is an important consideration and is discussed further in Section 2.2.1 of this report. The DFM approach also allows for a safety factor which is divided into the crack propagation interval as determined in Equation 2.4 to give the RTS interval or:

$$RTS = \frac{CPI}{SF} \quad (2.5)$$

The magnitude of the safety factor used will depend largely on the individual component and the expected service cycle although a typical factor of 2 has been reported in reference [23].

The major disadvantage associated with using the DFM approach to develop a damage tolerance based life procedure is that it is too conservative and thus prevents the full economic benefits from such procedure from being obtained. The DFM approach does however require a much smaller data base than the PFM approach and due to its inherent safety, represents the more viable approach for implementation.

2.1.3 Combined Analysis Based On DFM

The Combined Analysis (CA) approach combines aspects of both the PFM and the DFM approaches to provide a method for extending the useable life of gas turbine components. By combining statistical data bases of user experience with laboratory test data and the results of post facto investigations of component failures, a realistic estimation of component life extension can be made which is both safe and cost effective. The CA approach also allows for modification of the maintenance parameters such as the RTS interval, on the basis of new data. This then allows continued improvements in the safety and/or the cost effectiveness of the program to be made.

There are many different methods that can be used to estimate a CPI using the CA approach. A brief description of three of the methods employed in the NAE/NRC program is provided here, however a more detailed description of the procedures employed is available in references [23][24].

One of the methods used in the determination of a crack propagation interval involves performing Weibull analysis using crack length and service time data obtained from the retired component inventory. In such an analysis, all retired disks from one particular stage having a crack size approximately equal to the detection limit are used to define a Weibull function. This function then provides an estimate of the probability that a component will have a crack of this length after a specified number of hours of service. A similar function is derived for component from the same stage having cracks of dysfunction length or greater. A suitable cumulative reliability for the component determines an estimate of the number of hours of service required to propagate a crack to both detection limit crack length and the dysfunction limit crack size for that component. The cumulative reliability usually selected in this application is 0.1% or 0.135%. The difference between the number of hours required to propagate a crack to dysfunction size or greater and the number of hours to propagate a crack to the detection limit size provides an estimation of the CPI based on

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actual user service data.

$$CPI = T_{dys} - T_{detect} \text{ at } 0.1\% F \quad (2.6)$$

This CPI is then divided by an appropriate safety factor to yield the RTS interval. An illustration of this procedure is presented in Figure 2.3. It should be noted that there are many variations to this method employing different Weibull parameters, different crack sizes and different data bases. A more detailed description of the Weibull analysis used in this program is available in reference [12].

A second method of estimating the CPI involves fractographic analysis of service induced fracture surfaces. By examining the fracture surfaces of failed components, an estimate of the crack propagation rates at various points along the fracture surface can be made. This is done by counting the LCF striations on the fracture surface between two points and dividing the number of striations found by the distance between the points. If one follows the assumption that each striation represents an applied stress cycle this analysis yields an estimate of crack propagation distance per cycle or da/dN . This is also referred to as the striation spacing. This information is then combined with the maximum undetectable crack size a_{detect} and the dysfunction crack size a_{dys} in the formula:

$$CPI = \frac{a_{dys} - a_{detect}}{(\text{ave. striation spacing})(M_{st})} \quad (2.7)$$

where M_{st} is the number of striations or cycles expected per mission hour. It is important to realize that the striation spacing will vary according to the applied crack tip stress intensity factor and thus this analysis assumes a constant ΔK over the crack propagation range of interest. As before, the CPI determined through this analysis would be divided by an appropriate safety factor to give the RTS interval. More information regarding the fractographic analysis performed for this project is available in references [23][24].

The final method which shall be discussed here involves the experimental determination of the crack propagation rate. The experimental determination of crack propagation rates in retired disk material is the principal focus of this thesis. In this instance the formula for determining the RTS interval is the same as equation 2.4 which was described in section 2.1.2 on the DFM approach. The equation has been repeated below for clarity.

$$RTS = \frac{CPI}{SF} = \frac{a_{dys} - a_{detect}}{C(\Delta K)^n M_f} \quad (2.8)$$

where a_{dy} is the dysfunction crack length, a_{detect} is the detection limit crack length, "C" and "n" are material constants, ΔK is the crack tip stress intensity factor and M_f is the number of LCF cycles per mission flight hour. The main difference between this method under the CA approach is the means through which an estimate of the stress intensity factor is obtained.

In the combined analysis approach, three separate methods were used in an effort to estimate the crack tip stress intensity factor. The first of these methods was a Finite Element Method (FEM) stress analysis. Both two and three dimensional analyses were performed. The FEM stress analysis of the disk offers the capability of determining the crack tip stress intensity factor under realistic service conditions. Details of the analysis performed in this study are presented in Section 2.2.1.

The second method that was used to estimate the crack tip stress intensity factor involved determining the plastic zone size formed ahead of the LFC cracks found in retired disks. This analytical procedure is also explained in Section 2.2.1.

Finally, a post facto investigation of service induced fracture surfaces was performed so that striation spacings observed in failed components could be compared with similar striations formed under controlled laboratory conditions. Since the loading and geometry of the laboratory specimens are known and the crack lengths monitored, the stress intensity factor ahead of the crack tip in the test specimens can be calculated using the formula:

$$\Delta K = \frac{\Delta P(2 + \alpha)}{B\sqrt{W}(1 - \alpha)^{1.5}} (0.886 + 4.64\alpha - 13.32\alpha^2 + 14.72\alpha^3 - 5.6\alpha^4) \quad (2.9)$$

Where $\alpha = a/W$. Since the striation spacing is directly related to the crack growth rate and thus the stress intensity factor, by matching the striation spacing observed on the service induced fracture surfaces with those produced in the laboratory an estimate of the crack tip stress intensity factor experienced by the failed component can be made. A further description of this technique is presented in Section 2.2.1

By using various parameters obtained through the analysis of service induced failures and user experience in deterministic analytical formulas, a safe estimate of realistic RTS intervals can be made. This forms the basis for the CA approach used throughout this study.

2.2 Associated Analysis

As can be seen from Equation 2.8, the crack propagation rate is not the only factor which must be determined in order to estimate the RTS interval. The NDI detection limit, the critical crack lengths, dysfunction crack length as well as the number of LCF cycles experienced by the engine components per mission hour must be evaluated in order to allow calculation of the RTS interval. Also, of paramount importance is the evaluation of the stress intensity factor at the crack tip under service conditions. These parameters have been estimated as part of the NAE/NRC post-facto investigation into the failures of the fifth stage compressor disks of the J85-CAN15/40 series engines. A summary of the approaches and methods used to compute these parameters are presented below.

2.2.1 Estimation of Crack Tip Stress Intensity Factor

One of the methods used to estimate the crack tip stress intensity factor was a Finite Element Method (FEM) stress analysis. In this analysis a rather coarse mesh was generated using twelve node isoparametric elements to represent a radial section of a fifth stage disk. An illustration of this mesh is presented in Figure 2.4. Several simplifications were made in order to facilitate the analysis. These simplifications included neglecting the blade mass, the aerodynamic loads, bolt and washer mass and any interaction effects from the tie bolts and adjacent spacers. The disk was modeled at 16500 rpm. In order to evaluate the stress intensity factors at various crack lengths, the nodes along the crack path were "released" in succession and by using crack tip singularity elements the stress intensity factors were evaluated. The results of the two dimensional analysis are presented in Figure 2.5 and those for the three dimensional analysis in Figure 2.6 [21]. As can be seen from these graphs the results obtained for the two analyses are quite different. The two dimensional analysis results form a "plateau" of constant ΔK between crack lengths of 2.5 mm and 15.75 mm while no such trend is illustrated in the three dimensional results. Although there is some evidence to support the 2-D results on the basis of observed striation spacing on service induced fracture surfaces there is some question as to why such a plateau would occur. The plateau at a stress intensity factor of approximately $44 \text{ MPa}\sqrt{\text{M}}$ is also supported somewhat by the results of the other ΔK estimation techniques employed. However, there is also some doubt that the results of these particular FEM analysis are at all realistic

because of the drastic simplifications employed in the analysis. It is now thought that the main reason for the cracking at the bolt holes is the variation in torque transfer through the tie bolts during major throttle excursions. If this is indeed the case then an FEM analysis which ignored such interaction would hardly be representative of the in service conditions.

There are however several new Boundary Element Analysis packages available which could be used to provide a more realistic estimation of the crack tip stress intensity factor, given that more realistic assumptions are made concerning the service loading conditions.

Quantitative fractography is another technique that can be used to estimate the stress intensity factors ahead of LCF cracks. In fractography, the fracture surfaces are separated and examined under a Scanning Electron Microscope (SEM). Qualitative observations of the fracture surface are usually used to determine the failure modes. In the case of fatigue failures, some metals and alloys form fatigue striations on the fracture surface as the crack propagates. These striations can offer quantitative information regarding the conditions of failure. An indication of the crack propagation rate can simply be obtained by adding the total width of striations observed over a specified distance of the fracture surface and dividing the total striation widths by the number of striations counted. Typically each striation is taken to represent one crack growth increment or cycle [20][28], and thus by knowing the number of striations over a certain crack length an estimation of da/dN can be made [26][27]. This striation spacing in turn is representative of the stress intensity factor experienced by the crack tip during fracture as indicated by Equation 2.3. By comparing the striation spacing of a cracked disk with that of controlled fracture mechanics crack growth rate tests of the same material it may be possible to correlate the two and determine the actual stress intensity factor operative ahead of the crack tip in actual disks [20]. The results of this analysis are presented in Section 4.4.1 of this report.

The plastic zone size measurements ahead of crack tips on LCF cracks formed during spin pit testing and/or during service can be used as a means of evaluating the crack tip stress intensity factor as well. A microhardness measurement technique can be used to estimate the size of the plastic zone formed at the crack tip during loading [2][24][28][29][30]. The typical results of this type of measurement technique are reported in reference [24]. The microhardness contours ahead of the crack tips in J85-CAN40 fifth stage compressor disks that were cracked during service are summarized in Figure 2.7. Once an approximate plastic zone size has been determined, an estimate of the cyclic stress intensity factor (ΔK),

can be calculated using the theoretical relationship suggested in references [29][30]:

$$R_v = C \left(\frac{\Delta K}{\sigma_v} \right)^2 \quad (2.10)$$

or

$$\Delta K = \sigma_v \sqrt{\left(\frac{R_v}{C} \right)} \quad (2.11)$$

Where ΔK is the cyclic stress intensity factor, σ_v is the material yield stress, R_v is the plastic zone size and C is a constant dependant on the material and the type of loading used. The results obtained from this analysis are presented in Section 4.4.2.

2.2.2 Determination of the Number of LCF Cycles per Mission Hour.

A mission profile analysis of the J85-CAN15 engines has been performed by NRC in order to determine the average or typical number of LCF cycles imparted on the engine disks per mission flight hour. Unfortunately, only the results obtained for the J85-CAN 15 engines in the CF-5 aircraft were available for analysis. The J85-CAN 40 engine used in the Tutor trainer was also instrumented, however the data from these was not available due to incompatibilities between the data acquisition system and the computer used for analysis purposes [24][34].

The instrumented CF-5 aircraft was capable of recording several engine parameters including compressor inlet temperature, turbine inlet temperature and engine speed. For the purposes of compressor disk study, the engine speed was considered to be of prime importance. The instrumented aircraft did not fly typical mission profiles but rather performed various maneuvers which may be encountered over the operating envelope of this particular aircraft. This allowed the number of engine speed variations per maneuver as well as the time required for each maneuver to be recorded [35]. However no indication of the number of each maneuvers experienced during a full mission was available. It was therefore necessary to supplement this flight data with typical mission profiles supplied by Canadair [36]. In this manner, the typical number of specific maneuvers per specific mission could be obtained and analysed in conjunction with the number of throttle variations per specific maneuver. Thus an estimation of the number of throttle variations per specific mission was obtained. The number of throttle variations per mission flight hour was then determined by analysing the number of different missions flown during the total flight time

of the aircraft over a period of one year. This provided the typical number of engine speed variations per mission flight hour on the basis of one years use in various roles.

In order to limit the amount of data analysed to that which may constitute a LCF cycle on the engine components, several assumptions were made. First of all, it was assumed that engine dwell times (ie the time at constant speed), would have little effect on the lifetime of the engine's cold section components. Subsequently, the dwell times were eliminated from the analysis, leaving only a series of peaks and troughs of engine speed variation. It was also reasoned that engine speed variations of less than 25% of maximum speed would not constitute a significant LCF cycle to the engine components and thus peaks and troughs of less than 25% variation from maximum were eliminated from the analysis as well. This assumption was deemed reasonable in that, had all of these speed variations been considered LCF cycles, the resultant time to component dysfunction would have been significantly less than that indicated by user experience. A more detailed explanation of the analysis used is available in reference [34]. On the basis of this analysis, the following results were reported.

The CF-5 aircraft with two J85-CAN15 engines was found to impart an average of 6.35 or 6 LCF cycles to the engines per mission flight hour on the basis of the yearly flying rate. The CT-114 Tutor trainer aircraft mission profile analysis was performed solely on the basis of typical mission profiles supplied by Canadair since in flight engine data for this aircraft was unavailable. The results from this analysis indicated that a typical mission flight hour contained approximately 3 LCF cycles that were applied to the engine disks. However, this number may be somewhat lower than what is experienced under actual flight conditions [24][34]. Since the two versions of the J85 engine in use with the CAF share the same cold end components and even exchange these components between the two types, it was felt that in compliance with the deterministic approach employed in this study, a value of 6 LCF cycles per mission flight hour would be regarded as typical for this engine in both applications. This number was also indicative of the typical number of small striations observed between major striations on the fracture surfaces of service induced cracks, as explained below.

One of the more significant observations made in the examination of the service induced cracks, was the grouping of several smaller striations by two or more large striations. This phenomenon is illustrated in the fractographs presented in Figures 2.8a and 2.8b. The number of smaller striations observed in a band tended to vary between five and fifteen

with five, being the most common. It has been suggested that each of these bands may represent a flight cycle of the aircraft. The major or large striations may result from the stress state imposed on the engine during start up (0 to 100% RPM) while the smaller striations may result from engine speed variations implemented by the pilot during flight maneuvers. This hypothesis is presently being investigated in a combined effort of spin pit testing of the disks at the University of Toronto and fractographic analysis of the resultant cracks at the Structures and Materials laboratory of NAE/NRC.

2.2.3 Determination of Dysfunction Crack Lengths

The dysfunction crack length, a_{dys} , defined as the maximum crack length with which the aircraft operate safely, is an important consideration both in the determination of the length of the RTS interval and from an overall safety stand point. Obviously, the longer the maximum crack length allowed in a disk, the longer the RTS interval can be and thus the more economical a damage tolerance based lifing procedure becomes. On the other hand however, the longer the crack is, the closer it becomes to the critical crack length a_{crit} where the critical stress intensity factor ΔK_{crit} occurs resulting in the fast fracture of the material. The dysfunction crack length therefore must be some fraction of the critical crack length. In following a deterministic based combined analysis approach, the critical crack length is assumed to be that which results in a stress intensity factor which causes tertiary crack propagation. This concept is illustrated in Figure 2.9. Selection of an appropriate value for the dysfunction crack length is a difficult task because of the significance it plays in determining the success of the maintenance program. There is some evidence that the dysfunction crack length could be as long as 10 mm as indicated by the striation spacing plotted in Figure 2.10. Similarly, the longest crack found in any of the J85-CAN40 disks was 19 mm long and yet the disk had not suffered catastrophic failure. Despite these observations, the dysfunction crack length employed in this study was 4.5 mm for the fifth stage disk as supplied by General Electric as part of its continued service program. According to G.E., this value was selected on the basis of engine component vibration problems, so in this case the limiting factor for a_{dys} was operational considerations and not material considerations.

2.2.4 Determination of NDI Detection Limits

The final parameter of significance in the determination of an RTS interval based on the

results of crack growth rate testing, is the NDI detection limit. Previous studies at NRC on compressor disk bolt holes have determined the detection limit of the NDI technique used by Orenda, the overhaul contractor, using a combination of statistical and empirical analysis as reported in references [23][24]. The NDI technique used by the overhaul contractor is the liquid penetrant inspection (LPI) technique. In this technique, the components are cleaned and then submerged in a liquid fluorescent dye for a specified length of time. This is to allow the dye to penetrate all the flaws and cracked surfaces on the component. The component is then removed from the dye bath and dipped in a washing/neutrallizing solution to remove the dye from the unflawed surfaces of the component. The component is then examined under special lighting which makes the dye glow noticeably. All the flaws and cracks where the dye has penetrated and thus has not been neutrallised will glow making their detection easier. The length and/or dimensions of the flaw or crack are then determined using a calibrated travelling microscope. The detection limit of 1.75 mm as reported in references [2][24] for the LPI technique has been selected for use in this study.

2.3 Influence of Service Induced Material Degradation on the RTS Interval

Two types of metallographic analysis were performed in order to determine the extent of service induced degradation of the compressor disk material as reported in reference [24]. Optical microscopy was used to examine the microstructure of the fifth stage disk at different radial distances from the bore. This was done in order to determine if the microstructure of the material had been affected by the stress and temperature cycling the disks were subjected to during service. A disk with a relatively long service life was sectioned and each of the radial pieces was cut into five separate sections from the rim towards the bore. These coupons were then mounted, polished and etched to reveal the microstructure. The microstructure of each of the coupons was then examined under an optical microscope. No difference between the microstructure at the rim and the bore of the fifth stage disk was observed. This result was not surprising given the relatively low operating temperature of the component during service [2][24].

The precipitate morphology of the fifth stage disk was examined as well, using a replica transmission electron microscopy technique. In this investigation plastic replicas were made from the crack tip regions around service induced cracks. These replicas were then exam-

ined under a transmission electron microscope. A significant change in the precipitate morphology was noted in the regions of intense cyclic straining. As can be seen from Figures 2.11a and 2.11b, the precipitates at the crack tip regions seem to have coarsened along slip lines, whereas those observed away from this region are more uniformly distributed. This is indicative of severe deformation and therefore quite high stress intensities at the crack tip [24].

The variation in microstructure with increased service becomes an extremely important consideration in that the component may no longer maintain the material properties and characteristics of the virgin stock. This variation in material properties with service exposure can then make predictions concerning possible life extension as well as the RTS intervals on which the life extension is based, erroneous thereby seriously compromising the safety of such a program. It is quite possible that the change in component microstructure with service exposure time will warrant decreased RTS intervals as the component is used. This reduces the economic benefits of a damage tolerance based maintenance methodology from the ideal retirement for cause philosophy. The change in material microstructure also adds another degree of uncertainty to the analysis and may warrant increased safety factors for all the RTS intervals thus further reducing the economic benefits of such a program.

2.4 The Direct Current Potential Drop Technique

2.4.1 Background

The measurement of crack length during fracture mechanics testing can be accomplished by a number of techniques. Of the techniques applicable to fatigue crack growth rate measurement, the three most popular and widely used methods are: optical tracking of the crack tip using a travelling microscope, the compliance measurement technique using a crack opening displacement (COD) clip gauge, and the electrical potential drop method. A more detailed account of these crack length measurement techniques is available in reference [40].

The optical tracking technique is limited in application because the crack tip must be visible at all times. This precludes its use in applications requiring the use of a surrounding furnace or an environmental chamber. This technique is also very inconvenient in that an operator must be present for the entire test in order to take readings on a periodic

basis. Various attempts have been made to automate the optical technique using cameras. However, this has not proven to be very successful [41]. Another problem associated with this technique is the visual interpretation of the crack tip which can vary significantly from operator to operator. However, the largest drawback of the optical technique is its inability to provide crack length data to a computer for conducting a controlled test [41][42][43][44].

The compliance technique has been successfully employed in both automated and computer controlled tests [40][45]. This technique measures the crack opening displacement (COD) across the notch of the specimen while it is under a loading cycle. As a crack propagates through the specimen, its relative stiffness is decreased resulting in an increase in COD for type I cracks. Thus the COD can be calibrated as a function of crack length. The COD is measured by means of a clip gauge which is a self contained bridge circuit consisting of strain gauges. At least two of these strain gauges are mounted on arms that are attached to the specimen by clipping onto knife edges which are either machined into or bolted onto the specimen edge. The movement and resultant variation in strain in these arms is indicative of the COD.

There are however, several drawbacks to the compliance technique. First of all, friction between the pin-specimen interface can significantly affect the readings [45]. Secondly, in high temperature tests the clip gauge must be removed from the heated area, thus necessitating the use of an extensometer which adds a further source of error to this technique. Thirdly, testing a specimen under random or mission profile waveforms can pose problems in maintaining a consistent datum from which to measure the relative COD. And finally, in thin specimens, such as used in this work, difficulties in attaching the clip gauge arms to the thin edge of the specimen can make the readings unreliable.

In light of the disadvantages noted with the optical and compliance crack length measurement techniques for this particular work, it was decided to implement the potential drop (PD) method. It was also reasoned that once the PD technique was built in its functional form, it would prove to be very useful for future work at the Structures and Materials Laboratory of NAE/NRC.

The PD technique offers several advantages over other crack length measurement techniques [41] [42][43][44][45]. Firstly, there is little operator interpretation required. The

Absolute accuracy of the technique varies with the set up, the equipment used and the test parameters however, accuracies of better than 0.1 mm, (0.004 in) have been obtained [41][43][44][46][47][48]. In crack propagation testing, greater emphasis is laid on the precision of the technique. The potential drop technique has proven to be very precise and precisions of greater than 0.05 mm (0.002 in) are not uncommon [41][43][44][46][47][48]. In specialized applications of this technique used for the detection of crack formation from small defects, instantaneous resolution of the electrical potential technique equalled between 0.8 and 2.4 microns [47]. The PD technique has also proven to be remarkably stable over long periods of time, thus promoting its use in instances where fatigue test frequencies in the range of 0.1 to 1 hertz are used. Similarly this stability has also proven to be valuable for threshold stress intensity factor determination testing [42][46][49][57]. The PD technique is also very suitable for high temperature work because no observation of the crack tip is required and the use of cumbersome extensometers is avoided. The technique has proven to be particularly valuable in tests where gross in-elastic deformation or creep occurs because the calibration of the technique does not rely on elasticity theory [43][44][47][49][50][51][52][53][55][56][57]. The effect of creep void formation ahead of the crack tip has been investigated by Nicholson and this has been shown to produce a negligible effect on the potential drop measurement relative to the potential drop measurement due to crack extension [57].

Another distinct advantage of the PD technique is the ease with which it can be used for corrosion fatigue testing. Once again, since no direct observation of the crack tip is required, the use of an environmental chamber poses no difficulty in the crack length determination. The technique has proven successful in both salt water and hydrogen environments [44][46][49][51][58][59][60][61][62]. There are however, several concerns which must be noted in the use of the PD technique in corrosive environments. First of all, the application of a current through the specimen being tested can affect the extent and nature of the corrosion experienced. This results from the fact that corrosion is typically a chemical reduction process that is associated with the transfer of ions and electrons. Secondly, the voltage readings obtained from the specimen may be affected by variations in potential incurred by the reduction process. As well as these considerations, one must be aware that the build up of corrosion products on the crack face may provide a means for the current to short circuit its intended path around the crack tip thus giving a false

reading. However, careful attention to test procedures can reduce, if not eliminate, these problems [47].

The PD technique also offers a virtual independence from the applied loading cycle thus allowing various mission spectrum loading sequences to be employed. This also allows for changes in test parameters to be implemented without stopping the test. This loading versatility and the ability of the technique to be applied in tests involving creep are particularly valuable in testing materials that are used in gas turbine engines.

The PD technique also allows versatility in the selection of specimen geometry or configuration, allowing the optimum selection for any given application. Specimens can also have multiple crack initiation sites so that many cracks can be monitored simultaneously during a single test. The PD technique is also particularly appropriate for use with small or thin specimens which can cause problems with other techniques. Since the PD technique is effectively an electrical resistance technique, decreasing the size of the specimen increases its resistance and thus the accuracy and the precision of the technique are improved.

Another important advantage of the PD technique is its simplicity and low cost equipment requirement. Although some familiarization is required before one can employ the technique successfully, in general the concepts and principles associated with the technique are easy to understand. It should be noted however, that certain types of specimens, for example the CT specimens, may require specialized techniques in preparing the specimen for use.

A very significant advantage offered by the potential drop technique over other methods is its computer compatibility. Since the technique uses electrical inputs and outputs, it is very easy to perform a computer controlled test and to acquire the data directly on a computer. This computer control not only requires a minimal amount of supervision during testing but it also allows one to employ a more sophisticated and precise PD technique (refer to section 2.4.3). Since the computer can be dedicated to one test at a time and operates in real-time, the test parameters can be controlled using the crack length data supplied by the technique. This is very useful for precracking a specimen or for constant and threshold stress intensity factor testing. Another important advantage of computer control and data acquisition is that data reduction can be performed simultaneously using the arithmetic capabilities of the computer. In this way test results can be monitored to ensure correct

operation [41][43][46][51][52][62].

2.4.2 The Principle and Operation of the DC PD Technique

Two types of potential drop technique were available for implementation; the direct current (dc) method and the alternating current (ac) method. While the ac method has been reported to offer higher sensitivity and improved accuracy over the dc method [43][64], the ac method is much more complex and difficult to implement. In addition, the equipment required for the ac method is more elaborate. It was therefore decided to employ the direct current PD technique for this work.

The principle behind the dc PD technique is quite simple. A constant current is applied through the specimen which is being tested. This is illustrated in Figure 2.12. As can be seen from this illustration, as the crack grows from the notch, the distance through which the current must travel to maintain continuity is increased. Simultaneously the cross sectional area through which the current can flow is decreased. This has the net effect of increasing the electrical resistance of the specimen according to the equation:

$$R = \frac{\rho L}{A} \quad (2.12)$$

where R is the electrical resistance of the specimen, ρ is the material resistivity, L is the length of current travel and A is the cross sectional area available for current flow. Now referring to Ohm's Law, when a constant current is maintained through a body of increasing resistance, the voltage drop or potential drop across that body increases proportionately or:

$$V = IR \quad (2.13)$$

where V is the potential drop across the specimen, R is the electrical resistance of the specimen and I is the constant dc current magnitude. This increase in voltage is then measured and calibrated as a function of crack length.

2.4.3 The Modifications to the Basic DC PD Technique

Increased interest in the potential drop technique in the past decade has lead to its continual development. This has resulted in several major modifications to the basic technique which have increased its accuracy, precision, reliability and long term stability. All

of the improvements developed have been incorporated into the technique employed in this effort.

The first modification is the addition of a reference voltage block. Basically, an identical test coupon is attached to the current source in series with the specimen being tested. The voltage across the notches of the two coupons are then monitored simultaneously and the ratio of the two is taken as the crack length dependent parameter. This set up is illustrated in Figure 2.13. Use of a reference voltage block has two advantages over a single voltage system. First of all, because the coupons are connected in series with the current supply, any variation in the current magnitude however small, will effect both coupons similarly and thus will have no effect on the voltage ratio. Secondly, if the two specimens are maintained at the same temperature and environment, any temperature dependency of the material resistivity will similarly have no effect on the voltage ratio. Thus any major deviations from the calibration curve determined at room temperature are unlikely. It should also be noted that this reduces the heating effects resulting from the resistive power dissipation in the coupons [41][44][46][49][52] [63][65][66][67].

Another problem arises from the fact that metals are inherently excellent conductors of electricity. This results in very small initial and incremental voltage across the specimen as the crack propagates through the specimen. In order to ensure that the voltage rise due to the increase in crack length is the dominant measurement, a relatively large current must be applied through the specimens. This large current however, causes heating in the current leads and the lead-specimen junctions which not only effects the resistance of the leads but also turns them into red hot elements which are an undesirable safety hazard. These problems can be minimized if not eliminated by employing a pulsed current system. In such a system, the current is only turned on for the data acquisition sequence, approximately once every four hundred cycles. Since data acquisition is typically accomplished in under four seconds, the leads and junctions do not have time to heat up appreciably [43][46][68].

Another modification to the technique which works in conjunction with the pulsed current system is automatic voltage zeroing. This involves measuring the voltage with the current on and then with the current off and then taking the difference of the two readings. In this way, only the voltages induced by the application of the current are considered, eliminating the effects of amplifier offset, voltages induced by the thermocouple effect at dissimilar metal junctions and other external interference which may be present.

in the laboratory. This automatic voltage zeroing is performed simultaneously at both the reference voltage block and test specimen [43][46][51][68].

As was previously mentioned, a computer is used in this application for both control of the test and data acquisition. This flexibility in control allowed several other improvements in the technique to be employed. In order to enhance the accuracy of the technique, the computer is programmed to stop cycling and maintain a maximum load plateau during data acquisition. This is done for two reasons. First of all, by holding the crack open at maximum load for all data acquisition sequences, the maximum crack extension is consistently measured at each sequence. This also eliminates any closure effects at the crack tip as well as ensuring that the current could not short circuit across the crack resulting in an inaccurate voltage measurement. Secondly, by halting the cycling during data acquisition, the voltage induced by the magneto-mechanical effect is eliminated [46][69]. Bardal et al have reported that voltage measurements can be significantly altered by the magneto-mechanical effect or the "inverse effect of magneto-striction." This effect is the result of the magnetic domains in the material rotating under stress such that the direction of magnetization becomes parallel to the direction of deformation. In fatigue tests however, the deformation is cyclic in nature thus causing a cyclic variation in the magnetic intensity. This in turn can induce a current in the specimen which then affects the voltage measurement. This effect is not neutralized through the use of a reference voltage block as the reference block is typically unstressed during a test. However, by holding the load constant during data acquisition this effect is eliminated [46].

Another technique employed through the use of the computer to increase the accuracy and precision of the technique, is that of multiple data acquisition and averaging. In other words, the voltage across the sample and reference coupons are measured "N" different times in both the current on and off modes. These measurements are then averaged and the mean value is used to represent the measured voltage. This in effect smooths out any noise or voltage spikes which may occur, thus providing a more consistent measurement. Increasing the number of data points taken during a measurement to over twenty, did not demonstrate a more consistent baseline reading. It is felt that as more data readings are taken above a certain number, the incidence of noise and or spikes increases so as to nullify any benefits obtained from the increased sample size.

The net effect of these modifications is a temperature, environment, current, interfer-

ence independent dimensionless parameter known as the beta (β) parameter. The resultant calculation performed to determine its value for each data acquisition sequence is presented below.

$$\beta = \frac{(V_{on} - V_{off})_{sample}}{(V_{on} - V_{off})_{reference}} \quad (2.14)$$

Notice that the parameter is dimensionless and when calibrated versus a/W , where a is the crack length and W is the width of the specimen, provides a non dimensional calibration curve which is material and specimen size independent as well. However, the beta parameter is still dependent on the location of both the current input leads and the voltage measuring probes.

2.4.4 The Optimum Location of Voltage Probes and Current Leads

A drawback to the potential drop method is its dependence on current lead and voltage probe location and placement. In other words, not only must the attachment of these wires be accurately placed, their location must also be consistent from one specimen to another. The first question to be addressed is where on the compact tension specimens should they be located and secondly, how they should be attached in order to ensure consistency.

The question as to where the leads and wires should be placed has been addressed by several authors. Their methods of determining the optimum local have varied from experimentally determining the potential field distribution using graphitized electrical analogue paper [42][65], analytical analysis using Johnson's Equation [43][70][71][72] and solutions for the Laplace equation [65][73], to methods involving finite element method analysis [42][44]. Fortunately, there is relatively consistent agreement as to the optimum location for the potential probe wires. Virtually all agree that the optimum location for the voltage wires as far as maximum sensitivity and maximum reproducibility are concerned is on the notch face of the specimen. Although the distance from the notch tends to vary from author to author, for this work, it was decided to follow the recommendations outlined in the "Working Document for the AGARD Cooperative Test Program on Titanium Alloy Engine Disk Material" [74]. This report suggests that the probes be located on cross corners of the notch and specimen edge as illustrated in Figure 2.14. While there is some concern that locating the probes this close to the notch isolates them from the active current flow and potential distribution, this location was selected on the basis of enhanced reproducibility and decreased sensitivity to placement error. In other words, this location requires no

linear measurement of distance from notch to desired probe location, thus eliminating this as a source of error. Aronson and Ritchie also promote this as the optimum location on the basis of their finite element work [44][49][71]. They found that when positioned as close to the notch as possible, the voltage probes were insensitive to small errors in positioning and provided the maximum increase in voltage over other locations.

The selection of an optimum position for the current leads is somewhat arbitrary, although the most popular location appears to be along the top and bottom edges of the CT specimen. Once again it was decided to follow the recommendations of the AGARD co-operative test procedures [74] and locate the current leads on the top and bottom edges at the load line, as illustrated in Figure 2.14.

The second major concern involves the consistent attachment of the wires to the specimen. This was accomplished using a laser welding technique which allowed accurate placement of the wires with a minimum weld size and heat affected zone. Tungsten inert gas (TIG) welding was initially attempted and proved to be unsatisfactory.

Chapter 3

Materials and Mechanical Testing

3.1 Aims and Goals of Testing

The main focus of this study was the determination of crack propagation rates in the fifth stage compressor disks of the J85-CAN40/15 engine. These fatigue crack propagation tests were conducted in order to attain the following five objectives:

- 1) Development of a direct current (dc) potential drop (PD) crack length measurement system for use at the Materials Laboratory of NRC/NAE.
- 2) Determination of a statistically significant data base of LCF crack propagation rates over a wide range of ΔK values.
- 3) Investigate the effect of operating temperature on LCF crack propagation rates in this case.
- 4) Determination of a return to service (RTS) interval on the basis of LCF crack propagation data generated on compact tension specimens machined from retired compressor disks.
- 5) Determination of whether such an RTS interval should remain constant or vary according to the service exposure of the component.

3.2 Implementation and Calibration of the DC PD Technique

3.2.1 Equipment and Potential Drop Parameters

The equipment used in the direct current (dc) potential drop technique consisted of a dual voltage recording device, a power supply and two amplifiers. The voltage recording device used in this work was a Digital PDP-11/04 series computer. As well as performing the data acquisition for the tests, the computer was also used to turn the power supply on and off as required. This was accomplished by supplying a ten volt ramp to a switching unit designed and built by the electrical workshop of the Structures and Materials Laboratory

of NAE/NRC. This unit provided the appropriate signal to turn the power supply on and off. The computer itself is described in detail in Section 3.5.3 of this report.

A Trygon Electronics Super Mercury power supply was used as the constant current source in this work. It had an output capability of 0-50 amps or 0-40 volts dc. A current of ten amps was used for all experiments. Using a current of smaller magnitude would have had a detrimental effect on the accuracy and precision of the technique, while increasing it significantly created a heating problem in the stainless steel leads attached to the specimens. The current magnitude was set using the controls and built in ammeter of the power supply. As stated in Section 2.4.3 of this report, the absolute magnitude and consistency of the current supplied were not important parameters, however once set, the current magnitude was not varied either during a test or from one test to another. The power level of the power supply was maintained at a level which could supply the constant 10 amp current despite any minor variation in the resistance of the power supply circuit. The current was turned on for a total of four seconds in the room temperature data acquisition sequences and a total of three seconds in the elevated temperature data acquisition sequences. The first two seconds in both cases were allowed for the current to stabilize at its steady state value while the remainder of the current on time was used for data acquisition purposes. The shorter data acquisition time of one second was employed in the elevated temperature tests because as was mentioned earlier in this report, the load is held at a maximum level during data acquisition and thus reducing this period would subsequently reduce the possibility of creep deformation occurring during the testing. There were no problems noted in using this cycle. Scanning Electron Microscope pictures of the fracture surfaces formed during the tests at two different temperatures are presented in Figure 3.1. As can be seen from these photographs, there is no evidence of any variation in the type of fracture experienced at the two different temperatures.

Two amplifiers were used to increase the measured voltage of approximately 1.4 millivolts to a value more compatible with the 0-10 volt range of the analogue to digital converter used to interface with the computer. The amplifiers used were a Pacific model 8255 and an Accudata model 218. Both were bridge type amplifiers with similar capabilities. They were each set to a gain of approximately one thousand which provided the computer with an initial (uncracked) voltage of 1.4 volts, which was within its optimum operating range. The amplifiers were adjusted prior to each test so that the beta parameter was initialized

to a value of unity to match the calibration curve. Amplifier drift and/or noise did not pose a problem over the duration of the tests.

Other equipment directly associated with the operation of the potential drop technique include various wires, electrical connectors and insulating material. The current carrying wires were fourteen gauge copper wires from the power supply to the specimen leads. The copper wire and the stainless steel leads were joined by insulated crimp type connectors. No heating problems were noted in either the copper wires or the connectors.

The potential wires leading from the specimen potential leads to the amplifier inputs were twenty-two gauge copper wires. These wires were connected to the amplifiers through five pin plugs designed for use with the amplifiers. In the room temperature tests, the stainless steel probe wires were soldered to the copper amplifier wires. However, this was impossible in the elevated temperature tests because test temperature was above the melting point of the solder. This problem was rectified by welding longer potential probes to the specimens and making the junction between the two wires at a terminal block. This proved to be the superior method by far as soldering stainless steel is rather difficult at best.

The test frame was isolated from the current carrying components of the system by nylon and teflon studs inserted into the load train above and below the specimen grips. Nylon was used in the load train above the furnace because of its relatively high modulus of elasticity and reasonable fatigue properties. Nylon was unsuitable for the lower portion of the load train because at the testing temperature of 200 C, the studs were subjected to approximately 170 C which is very close to the melting point of nylon. For this reason teflon was used in the lower load frame insulation point. Teflon is more suitable for elevated temperature operation. There were no fatigue or elongation problems noted with the studs. This system of isolation is pictured in Figure 3.2. This method of electrical isolation can be used for temperatures up to approximately 225 C after which point it is much preferable to use ceramically sputter coated grip pins. This arrangement is good for temperatures well above 1000 C. However, it was not deemed the optimum solution in this instance because of the extremely small load bearing surface of the coupons on the grip pins. It was felt that over the course of a test, the coatings could wear off and this would affect the potential drop readings.

In this work, the potential drop technique was calibrated optically, thus necessitating

the use of a travelling microscope. A Geertner telemicroscope was used for this purpose in conjunction with a model 7109-C-46 open type micrometer slide. The slide had a range of 50 millimeters and minimum graduations of 0.01 millimeters. The tracking system also came with an attachable lamp for illuminating the specimen and crack tip.

A summary of the parameters used in this application of the dc PD technique is presented in Table 3.1.

3.2.2 Calibration and Precision of the Technique

As was previously mentioned, the direct current potential drop technique employed in this work was calibrated optically using a travelling microscope as described above. In order to aid in the visualization of the crack tip, the region of crack propagation was first polished. An electropolishing technique was used instead of a mechanical technique in order to minimize the amount of material removed, and in order to avoid imparting any stresses in the coupons. A focused light attached to the microscope also aided in discerning the crack tip. The resultant calibration curve is presented in Section 4.4.1 of this report.

3.3 The Test Coupons Machined from the Fifth Stage Disk

3.3.1 Composition of Compressor Disk Material

The compressor disks in the J85-CAN 40/15 engines are made from AM-355 material. It is a martensitic precipitation hardening stainless steel with excellent corrosion resistance and weldability [37].

A chemical analysis was performed on one half of a compact tension specimen machined from a retired disk and reported by Raizenne in reference [20]. These results are compared with the material specifications MIL-S-62471(AT) [38], and are presented in Table 3.2. The chemical composition of the coupon was within the specified tolerances.

3.3.2 Tensile Testing

The physical constants of AM-355 as reported in Alloys Digest are presented in Table 3.3 [37].

The mechanical properties of AM-355 are extremely sensitive to the heat treatment used and thus can be varied according to application. The material supplied to General Electric

for the compressor disks had been given a proprietary heat treatment. This makes the determination of the baseline material properties difficult. Typical mechanical properties of forgings given a double transformation treatment are presented in Table 3.4 [37].

In order to better define the mechanical properties of the material being used in this work, three tensile tests were performed on flat specimens machined from retired disk material. An illustration of the specimen geometry used is presented in Figure 3.3. The tests were performed under a constant strain rate of 0.1 mm/mm/min using an Instron tensile testing machine and an Instron clip gauge with a 25.4 mm or 1 inch gauge length. A picture of this apparatus is presented in Figure 3.4.

3.3.3 Preparation, Geometry and Orientation of CT Specimens

The LCF crack propagation rate testing in this work was performed on compact tension specimens machined from retired components. In all, the LCF crack growth rate testing was performed in three phases. Firstly, the crack growth rates in a high service exposure time disk (4203.6 hrs.) at room temperature were determined by testing five coupons. These tests were also used to calibrate the room temperature application of the direct current potential drop technique. Secondly, six additional coupons were then tested in a furnace at the operating temperature of the disk i.e. 200 C (392 F). The third phase of testing involved coupons from a medium time disk (2381.4 hrs.), which were also tested at 200 C (392 F). The medium time disk coupons were only tested at the elevated temperature because it was determined from the high time disk coupons that the higher temperature resulted in increased crack propagation rates.

The geometry of the compact tension specimens used in this work is illustrated in Figure 3.5. The specimen met the ASTM standard E-647 minimum size recommendations in all dimensions excepting the specimen thickness. Since the CT specimens were machined from retired compressor disks their thickness was limited by the minimum thickness of the disk. This corresponds to the 0.94 mm (0.037 in) dimensions in the illustration shown in Figure 3.5. The compact tension specimen was selected for this application because its configuration allowed the maximum number of specimens to be machined from each disk [28], while maintaining both a constant radial position from the bore and the appropriate direction of crack propagation. By machining the CT specimens from the disk as close to the bolt holes as possible, the microstructural degradation which may occur around

the bolt holes in service, is automatically taken into consideration. Similarly, in order to simulate the crack growth rate behaviour of service induced cracking as closely as possible, the compact tension specimens were machined in a manner such that the direction of crack propagation through the specimen was radial towards the bore of the disk. This orientation is illustrated in Figure 3.6.

The CT specimens used in these tests required a precrack of 6.67 mm (0.2626 in) and had a valid final crack length of 17.6 mm (0.693 in.). These crack lengths correspond to a stress intensity factor range, ΔK , of $30 - 85 \text{ MPa}\sqrt{\text{m}}$ ($27 - 77 \text{ ksi}\sqrt{\text{in}}$) at a stress ratio (R) of 0.1. The specimens also meet the elasticity requirements set by the ASTM standard E-647 [39].

3.4 Test Parameters

The compact tension specimens were tested using a sawtooth loading cycle as illustrated in Figure 3.7. This loading cycle was selected because it was deemed more severe than a comparable sine wave form and because it more closely matched the trapezoidal waveform as recommended by Morn in reference [74]. The load plateaus required for data acquisition were two seconds long for all room temperature tests and one second long for all elevated temperature tests. The shorter hold time was employed on the elevated temperature tests to minimize the chance of creep effects being introduced. Although creep was not a major consideration in this instance because of the relatively low temperatures used during testing, it was felt that this test cycle would verify the applicability of the potential drop technique in other more creep prone experiments.

The load was cycled at a frequency of one hertz and at a load ratio of $R=0.1$. The maximum load applied was 1 KN (224.8 lbf).

The room temperature tests were performed at a temperature of 20 C to 29 C (68 F to 84 F) while the elevated temperature tests were performed at 200 C (392 F). The relative humidity varied from 50% to 100% over the duration of the experiments.

All specimens were precracked to a crack length of 6.667 mm (0.2625 in.) as specified in ASTM standard E647, using the same loading cycle and environment as the actual tests. This precracking resulted in a stress intensity factor (ΔK) of $30.6 \text{ MPa}\sqrt{\text{m}}$ ($27.84 \text{ KSI}\sqrt{\text{in}}$). While the thickness of the specimens used in these tests was slightly below that recom-

mended by the ASTM standard E647, only data which met the specimen requirements for elasticity were analysed. This requirement resulted in an allowable crack propagation range from $\frac{a}{w}=0.2625$ to $\frac{a}{w}=0.6135$, where "a" is the crack size and "w" is the width of the specimen.

3.5 The Test Equipment

3.5.1 The MTS Series 810 Materials Test System

An MTS series 810 automated fatigue test system was used in this work. The series 810 system consisted of a 100 KN (22 KIP) fatigue rated two column load frame, a 25 KN (5.5 KIP) fatigue rated actuator, and a 15 KN (3.33 KIP) rated single bridge load cell plus the required hydraulic hoses, couplings and fittings. As the Materials Laboratory of NAE/NRC operated several similar systems at the time of acquisition, the new system was connected to the existing hydraulic pump and associated systems.

The system was monitored and controlled by an 810A electronic control package. This package consisted of a type 410 digital function generator, a type 442 controller, a type 417 control panel, a type 413 master control panel and a type 464 data display unit. A type 433 processor interface was also purchased to allow the system to be computer controlled.

The system was capable of operation in any one of three modes; load control, strain control or stroke control. In this work the system was operated in load control. The system provided for several ranges of full scale load cell capability. The ranges offered included, 10, 20, 50 and 100 per cent of the load cell rating. In this work, a range of ten percent was used giving an effective load cell rating of 1.5 KN. Thus, the required 1 KN load was well within the accurate range of the load cell.

3.5.2 The Load Train

The grips used to hold the compact tension specimens in the load frame were designed according to ASTM standard E-647 specifications [39]. The design drawing for these grips is presented in Figure 3.7. The material selected for the grips was Nimonic 105, a nickel based superalloy. It was chosen so that the grips could be used at elevated temperatures such as may be required in the testing of turbine disc materials.

The pins used in the grips were also manufactured from a nickel based superalloy, MAR M-200. The selection of MAR M-200 was a result of consideration of its high temperature, high strength, good toughness and good creep resistance properties. These requirements were particularly important in the case of the pins where the load bearing area of the specimens may be quite small. The pins were ground to a diameter of 0.8 mm. (0.03125 in) below the ASTM specification E-647 requirement, to allow a ceramic coating to be applied. The sputter coating of ceramic on these pins would provide the necessary electrical isolation allowing the use of the potential drop technique in high temperature testing. The pins were held on to the grips with "C" type clip retainers on either end.

The load train was also equipped with a universal joint to ensure correct alignment of the load train under loading.

3.5.3 The Computer and Software

The computer used to control the loading and data acquisition sequences in the fatigue crack propagation testing was a Digital PDP-11/04. Specific functions such as turning the power supply on and off and controlling the potential drop technique were performed through an MTS 433 processor interface. The computer operated two disc drives, one for operating software and the other for data storage. Access to the computer was provided by either a Tektronix graphics display terminal or a Digital Decwriter III. The decwriter also operated as a hard copy printer.

The software used in this work was written and implemented by Mr. Tak Terada of NAE/NRC and is examined in reference [78][79]. It was written in MTS basic, a modified form of basic employed by the MTS company for their test systems. The software prompts the user for specimen specifications and test parameters and provides a check list to ensure the correct set up of the hydraulic system.

Once the test has been initiated, the program performs its first data acquisition cycle to provide the initial potential drop reading and beta parameter. The maximum and minimum loads and the number of cycles completed are also output. Once the data acquisition sequence has been completed, the computer cycles the load frame according to the function generation system specified in the software. This can be modified to allow the use of various loading cycles including saw tooth, sinewave or various combinations thereof. The function generator can also impart minor and major cycle loading if desired. The loading sequence

is repeated for the specified number of cycles between data acquisitions. After completing this number of cycles the function generator is once again stopped at maximum load to allow for data acquisition.

The data from each data acquisition cycle is stored on the data disk to free the hardware memory for the next cycle. The program is also written so that the cycling may be stopped by the operator at any time to either change a parameter or input an optical crack length measurement. The test can also be stopped permanently at any time but is set up so that it can be resumed with the data contained in the same data file.

The software developed also includes a data reduction program which takes the data from the data disk and calculates the stress intensity factor and crack growth rate on the basis of a crack length function derived from the crack length measurements. The calculations are based on the minimum crack length interval of 0.25 mm. (0.01 in.) as specified in the ASTM E-647 standard. The program was written by NEA/NRC personnel [78][79].

3.5.4 Design and Development of the Furnace

In order to simulate the crack propagation rates experienced by the discs in service as closely as possible, compact tension specimen tests were performed at the typical fifth stage disk operating temperature of approximately 200 C. It was also required to verify the crack length measurement determined by using the potential drop technique with corresponding optical measurements. This precluded the use of any of the available furnaces in the laboratory so one was developed specifically for this task.

The designed furnace consisted of a 3/4 of an inch plywood frame with a five thousandth of an inch thick nickel foil lining. The insulation on five sides of the furnace was ceramic brick. The sixth face was covered with a sheet of pyrex glass 3.137 mm or 0.125 in. thick. The elements used were thirty thousandth of an inch ni-chrome wire laid in a horizontal cross pattern on the face opposite to the pyrex glass face. All wires entering or leaving the furnace were sheltered in alumina ceramic tubes. The furnace was designed to be self supporting and had four leveling screws at the bottom for height adjustments of up to three inches. A picture of the furnace is presented in Figure 3.9.

Temperature control was provided by an Electric Control System, power supply model

7102-01-1-20, through an Intertechnology Doric DC7000 three mode proportional controller. The temperature was monitored with a dual K type thermocouple and a Doric Trendicator, series 400A, digital display unit.

The furnace was tested at 210 C. The resultant temperature profile is presented in Figure 3.10. As can be seen from the profile, there is significant variation in temperature in the vertical direction. The horizontal direction shows a much more consistent temperature. Since the test coupon is very small and the crack propagation is horizontal, the furnace was deemed to be adequate for the tests performed in this study. Also, the temperature inside the furnace was controlled by a thermocouple on the side of the specimen opposite of the element thereby ensuring that the temperature of the specimen was 200 C or slightly higher. This is consistent with the deterministic approach followed throughout this project. A picture of the entire test set up is presented in Figure 3.11.

Chapter 4

LCF Crack Growth Rate Results

4.1 The Calibration of the PD Technique

Readings of the crack length were taken at approximately 0.1 millimeter intervals in order to provide a consistent calibration curve. These optically measured crack lengths were input into the computer after which a corresponding potential drop data acquisition sequence was performed. After completion of the test, the calculated beta parameter was plotted versus a/W , where a is the crack length and W is the width of the specimen as defined by ASTM standard E-647-83. A curve was then fitted to this data. Trial and error indicated that a third order equation provided the optimum fit. This calibration equation was then put into the computer software for the next test.

In all, five calibration tests were performed. Of these five, one calibration test was aborted when the crack tip became difficult to locate. This was due to insufficient polishing. A second test was discounted because of an electrical leakage problem encountered between the grips and specimen retaining pins. This leakage was due to the electrical insulation wearing off the pins during the test. Leakage did not pose a problem once the system employing the nylon and teflon rods was installed.

The remaining tests were deemed valid and the results are presented in Figure 4.1. The equation of the calibration curve was:

$$\frac{a}{W} = -0.5611 + 0.9884(\beta) - 0.2579(\beta)^2 + 0.0359(\beta)^3 \quad (4.1)$$

4.1.1 Precision of the PD Technique

In crack growth rate testing it is important to realize that the precision of the measuring technique being used is far more important than its absolute accuracy. In other words, it is far more important to correctly measure the crack extension from point A to point B consistently than it is to know the absolute crack length at point A is "X" millimeters. Incremental crack extensions are extremely significant in determining the crack growth

rate da/dN , while the absolute crack length is only of minor significance in determining the change in stress intensity factor, ΔK .

The precision of a crack length measuring technique is defined in the ASTM standard E-647-83 as "the standard deviation on the mean value of crack length determined for a set of replicate measurements" [39]. In order to determine the precision of the PD technique employed in this effort, the beta parameter was calculated at four hundred cycle intervals before a crack was initiated from the starter notch. Only data obtained after 2000 loading cycles was considered in determining the precision of the technique because of the increase in measured potential resulting from the formation of a plastic zone ahead of the starter notch. The standard deviation from the mean was then calculated using all data obtained between the 2000 cycle point and the onset of cracking.

The largest standard deviation from the mean and hence the least amount of precision obtained from the valid calibration tests was 0.05 mm. (0.002 in.). This is well within the minimum required precision of 0.1 mm. (0.004 in.) as specified in the ASTM standard E-647-83 [39].

The ASTM does not provide a means by which the accuracy of a crack length measuring device can be determined. However, because the PD technique was optically calibrated in this work, the accuracy of the PD technique can not exceed that of the travelling microscope. It is also worthwhile to note that in order for the PD technique to take into consideration crack curvature as may be present in thicker specimens, the calibration procedure used must also account for crack curvature if the resultant calibration curve is to be accurate.

4.2 Tensile Testing Results

The results of these tests are presented in Table 4.1. Plots of the stress strain curves are presented in Figure 4.2. It should be noted that the properties determined in these tests were within the specifications set by General Electric, even after being exposed to over four thousand hours of service.

4.3 LCF Crack Growth Rates in AM355 Disk Material

The crack growth rate data from all tests were analysed using the Secant or point to point technique as described in ASTM standard E647. This analysis involves the computa-

tion of the slope between two adjacent points on the crack length versus the number of LCF cycles curve. In the elevated temperature tests, this computation was performed by the PDP 11/04 computer using the data reduction software developed for this specific purpose [78][79]. The computer performs a least squares curve fit operation to data points that conform to the minimum crack extension criterion of 0.25 mm (0.01 in) as specified in the ASTM standard E647. The slopes between these points are then determined. The room temperature test computations were performed by hand using the same technique.

It is well known that LCF crack growth rates show a large amount of scatter due to the inherent scatter in material properties. In order to ensure that the test data obtained was statistically significant and representative of the material, test conditions and environment, a number of tests were performed in all three test groups. Five room temperature tests were performed on coupons from the high time disk at room temperature, while both of the elevated temperature test groups consisted of six tests each.

The Paris relationship between the crack propagation rate and the stress intensity factor (equation 2.3) [80], for each test group, was determined using the least squares method of linear regression as outlined by Guttman et al [81]. Use of this technique to determine the parameters in the Paris relationship for crack growth rate is quite common and work by Stephens et al [82] and Swanson et al [85] are referenced as examples. The line determined using this method shall be referred to as the *mean line* from now on.

In order to account for the scatter of the data obtained through these tests, the results from each test group were bound by a scatter band. According to Pook and Smith [28][84], standard statistical techniques cannot be used to determine the scatter bands for fatigue crack growth rate data because crack growth on a microscopic scale is both a three dimensional and an irregular process. As a result, the stress intensity factor at any point on the crack tip and the corresponding crack growth rate must be considered to be random variables. Therefore, the scatter bands used in this study were found by drawing a line parallel to the mean line through the most extreme point in each test group. The distance between these two parallel lines was then applied to the other side of the mean line as well, to form the scatter band. It is felt that this method of upper limit of the scatter band determination is consistent with the deterministic approach followed throughout this study.

4.3.1 The Medium Time Disk

Six tests were performed on coupons machined from a disk retired after 2381.4 hours of service. This disk was one of a group of disks defined as "medium time" disks. These coupons were tested at the estimated service temperature of 200 C because of the faster propagation rates noted with the elevated temperature tests on the high time disk as reported below. The results of these tests are presented in Figure 4.3. The mean line equation fit to this data was:

$$\frac{da}{dN} = 2.14 \times 10^{-12} (\Delta K)^{3.16} M/cycle \quad (4.2)$$

and the upper scatter band line had the equation:

$$\frac{da}{dN} = 3.68 \times 10^{-12} (\Delta K)^{3.16} M/cycle \quad (4.3)$$

Once again ΔK is in units of $MPa\sqrt{M}$. The correlation coefficient which measures how well the curve fits the data was calculated to be 0.979:

4.3.2 The High Time Disk

The results of the five tests performed on the high exposure time coupons (4203.6 hours) at room temperature are presented in Figure 4.4. The mean line through this data has the equation:

$$\frac{da}{dN} = 4.05 \times 10^{-12} (\Delta K)^{2.93} M/cycle \quad (4.4)$$

and the equation of the upper scatter band line is:

$$\frac{da}{dN} = 7.69 \times 10^{-12} (\Delta K)^{2.93} M/cycle \quad (4.5)$$

where ΔK is in $MPa\sqrt{M}$ in both cases. The correlation coefficient determined for this series of tests was calculated to be 0.957.

4.3.3 Effect of Temperature

The results of the six tests performed using coupons from the high time disk at the elevated temperature of 200 C are presented in Figure 4.5. The equation of the mean line through this data was determined to be:

$$\frac{da}{dN} = 1.31 \times 10^{-12} (\Delta K)^{3.32} M/cycle \quad (4.6)$$

while the upper scatter band line had the equation:

$$\frac{da}{dN} = 3.04 \times 10^{-12} (\Delta K)^{3.32} M/cycle \quad (4.7)$$

Again ΔK is in $MPa\sqrt{M}$. The correlation coefficient for these tests was found to be 0.956.

The upper scatter band lines determined for the two high time disk results are presented in Figure 4.6. As can be seen from this graph, the operating temperature appears to have a significant effect on the crack growth rate in this instance.

4.3.4 Effect of Service Exposure

The graph of the elevated temperature results for both the high and medium time disk presented in Figure 4.7, illustrates an apparent dependence of crack growth rate on service exposure time as well. Therefore, the fastest crack growth rates were obtained by testing the high service exposure time disk at its estimated operating temperature.

4.4 Calculation of ΔK in Disks from Test Results

4.4.1 Fracture Mechanics Test Coupons

At NRC striation spacing measurements were carried out on twelve different bolt hole cracks of J85-CAN40 fifth stage compressor disks. The longest crack (19mm in length) noted in the retired fifth stage disk inventory was examined by a qualified fractographer. The results of these striation spacing measurements have been reported by Thamburaj in reference [24] and these are illustrated in Figure 2.10. As can be seen from this graph of striation spacing versus crack length, the striation spacing (and thus the crack growth rate), appears to remain relatively constant over a significant portion of the crack length. The remaining eleven bolt holes were analysed by an inexperienced fractographer and the results of these measurements were less consistent. The results of this analysis are reproduced in Figure 4.8. Because quantitative fractography relies heavily on the experience and interpretation of the fractographer, it is unknown how much of the scatter present in these results is a consequence of the material properties and/or stress state variation and how much is a result of interpretation error. However, it should be noted that all fractographic analyses results are subject to significant scatter as a result of operator interpretation and

the inherent scatter in fatigue properties. In addition, unlike aluminum alloys, AM-355 does not form striations which can be easily recognized or counted [20][33]. To provide a rough approximation of the average stress intensity factor experienced by these cracks under service conditions, the average of these striation spacing measurements was calculated. This average was determined to be 0.65 micrometers per cycle.

A fractographic analysis was also performed on two high time, room temperature compact tension specimens which were tested under constant amplitude low cycle fatigue conditions as reported in Chapter 3. The results of this analysis are presented graphically in Figure 4.9. As can be seen from the graph, the correlation is reasonable (corr. coef. $r = 0.71$) although once again there is significant scatter present. Two fractographs from this analysis are presented in Figures 4.10a and 4.10b.

By applying the average striation spacing observed in the service induced cracks to a linear equation fit to the striation spacing data obtained from the CT specimen tests, a rough estimation of the stress intensity factor experienced at the crack tip was obtained. The line fit through this data had the equation:

$$\Delta K = 20.2 \left(\frac{da}{dN} \right)_{\text{striation}} + 41.1 \quad (4.8)$$

Where ΔK is in $MPa\sqrt{M}$ and the striation measurement is in micrometers per cycle. The average striation spacing of 0.65 micrometers yields an approximate stress intensity factor of $61MPa\sqrt{M}$. If the average striation measurement of 0.65 micrometers per cycle is substituted into the upper bound crack growth rate equation determined for the high time disk at the service temperature, the resultant stress intensity factor obtained is approximately $52MPa\sqrt{M}$. Because this value is smaller than the value obtained through the former analysis, the value of $61MPa\sqrt{M}$ was assumed to be the more realistic value.

4.4.2 Tensile Test and Plastic Zone Size Measurement

The plastic zone size in the case of the J85-CAN40 fifth stage compressor disk was determined to be approximately constant at 0.45 mm over a range of crack lengths ranging between 2.0 and 6.7 mm. Using the suggested value of $\frac{1}{2}\pi$ for "C", [24][29][30][31] [32] and substituting the measured plastic zone size and yield stress value of the material for the J85-CAN40 fifth stage compressor disk into Equation 2.11, an approximate stress intensity factor of $59MPa\sqrt{M}$ was calculated for this component. Although this is only an estimate

of the crack tip stress intensity factor, it appears to be a reasonable approximation of the operative value [20][24]. This value also correlates very well with the stress intensity factor of $61\text{MPa}\sqrt{\text{m}}$ which was determined using the fractographic analysis of the CT test coupon fracture surfaces.

Although these analyses are very poor and inexact, they were used only because no better means of estimating the stress intensity factor was available. The results from these analyses are being used only as an illustration of the RTS interval calculation and should be replaced with the results of a more rigorous analysis when available. It is expected that such an analysis will involve a three dimensional finite element method stress analysis of the bolt hole region on the disk under service conditions. This type of analysis was scheduled for completion during the course of this particular work, but it has not yet been completed.

Chapter 5

Discussion

5.1 PD Technique for Crack Growth Rate Measurement

The PD technique is particularly suited for tests of moderate duration which are too time consuming for constant operator participation, and yet require data acquisition on fairly short regular intervals. Long term tests, in which only one measurement is required every day, may be better monitored using another technique. The PD technique is also more useful where there are many tests to be performed. Due to the somewhat involved set up procedure, most notably the laser welding of the probes and leads to the specimen, the PD technique may not be warranted for use where only one or two tests are required. However, when there are many tests to be performed and the welding can be done in a reasonable time period, the PD technique is ideal because of its almost automatic operation. The D.C. PD technique was used successfully in both the room temperature and elevated temperature tests performed in this work. While most of the advantages and disadvantages of the D.C. potential drop technique for measuring crack growth length have already been discussed earlier in this report, there are several points which warrant further discussion.

5.1.1 Calibration and Precision of the Technique

The calibration curve determined for CT specimens as presented in Section 4.1 agrees with that determined by Pickard and Hicks [42], who had identically placed potential and current leads. They used three different Finite Element Method (FEM) programs and a conducting paper method for calibration purposes [75][76][77]. The consistency of the calibration curves proves the material and size independence of the modified potential drop technique. A plot of potential drop determined crack length versus optically determined crack length for the room temperature tests is presented in Figure 5.1. A similarly determined plot for the elevated temperature tests is presented in Figure 5.2. These results show excellent agreement.

5.1.2 Advantages and Disadvantages of the Technique

The most obvious advantage of the PD technique was the minimal operator intervention required once a test was underway. In fact, approximately one half of the tests were performed during the night when there was no supervision. The computer that was used to control the tests and perform the data acquisition provided an output after each test consisting of a plot of crack length versus number of cycles. A typical output of this type is presented in Figure 5.3. By using this procedure, the crack length measurement could be verified by noting any irregularities in this graph. As mentioned previously, ASTM standard E-647 makes no recommendations for the determination of the accuracy of a crack length measurement technique. However, a comparison of Figure 5.3 with a similarly obtained plot of optically measured crack length versus applied cycles presented in Figure 5.4, reveals the consistency of the two techniques.

Another important advantage noted with the PD technique, was the ease with which it could be transferred from room temperature to elevated temperature use. This would be an even greater advantage at higher temperatures ie. 500-800 C, where a specific calibration curve could be determined at room temperature and then used for elevated temperature testing. The above method would be preferable to the high temperature clip gauge and extensometer arrangement for the reasons outlined in Chapter 2.

Despite the advantages of using the D.C. PD technique listed above, several problems were noted in its application. First of all, the use of a reference voltage block was supposed to eliminate any effects due to stray magnetic fields in the laboratory. This was found not to be the case in that the operation of a magnetic induction heater on another load frame within the laboratory totally disrupted any voltage measurement that was made while the heater was on. Subsequent grounding of all equipment and leads to a common ground did little to eliminate the problem. Whether the voltage measurements taken from the reference block and the test coupon were affected because of currents induced in the coupons themselves or in the leads and/or amplifiers was not determined. It is therefore recommended that the apparatus used for the potential drop measurement be adequately shielded from magnetic interference even when employing a reference block system.

Another significant problem was associated with the extreme fragility of the 0.2 mm (8 thou of an inch) diameter PD voltage probes at the weld where they were attached to

the samples. The fragile nature of these probe joints was a problem despite attempts to restrict their movement by reinforcing the weld-probe joints with epoxy. Although no welds were broken during actual testing, many failed during the placement of the specimens in the furnace and grips. These breakages caused delays in the testing due to the lead time involved in getting them rewelded. The only solution found for this problem was to increase the amount of epoxy used.

5.2 LCF Crack Propagation Rates in AM355 Disk Material

5.2.1 Effect of Service Exposure

The results obtained from the five room temperature tests on the high time disk material compare very well with the results obtained by Raizenne [20] using the identical specimen geometry and the material from a similar high time (4297.3 hours) fifth stage J85-CAN 40 compressor disk. Although a sinewave loading cycle at a frequency of 20 Hertz was employed by Raizenne, his data falls well within the limits of the scatter band of the high time room temperature test results generated in this study. This is clearly illustrated to be the case in Figure 5.5. This correlation is significant for the following two reasons.

First of all, if future tests of this type show a similar correlation, then it may be concluded that the results obtained from the disks tested are representative of the overall disk population and therefore these results are applicable to the population as a whole. In other words, the results obtained from testing a number of sample disks from a group, (say above 4000 hours of service for instance), can be applied to the entire group of disks. Ultimately, this means that the tests performed provide results which can be used to determine an RTS interval for components of that test group. Secondly, because the two high time disks tested had different service histories and yet provided similar crack propagation results, it may be possible to correlate the crack propagation behaviour of the material solely as a function of the service exposure time and not of the actual service history. In other words, the crack propagation rate used to evaluate the RTS interval would account for the expected mission severity by appropriately modifying the stress intensity factor while the material properties, (nominally C and n) would remain a function of the length of service exposure time only.

The dependance of the AM355 disk crack propagation rate on the amount of service exposure was confirmed by the tests performed in this study. The crack propagation rates

determined on coupons from a high time disk (4203.6 hours) and a medium time disk (2381.4 hours) at the same testing temperature were significantly different as illustrated in Figure 4.7. As can be seen from this graph, the crack propagation rates for the high time disk coupons were significantly faster than the propagation rates for the medium time disk coupons. Although some variation in the material properties could be a result of minor variations in the heat treatment and manufacturing process from one batch of disks to another, it is significant that a relatively large difference exists over the entire range of ΔK values. This is particularly noteworthy since no such difference was apparent between the two high time disk room temperature test results. The hypothesis that the material crack growth properties are dependant on the length of service exposure would have been far more conclusive had some coupons from a "zero time" disk been tested. Unfortunately, a zero time disk was unavailable for testing over the duration of this study. If this hypothesis is confirmed by future work, the determination of crack propagation rates of the components as a function of the amount of service exposure may provide the safest and the most economical means for extending their lives beyond their safe life limit.

5.2.2 Effect of Service Temperature

Coupons from the high time disk were tested at both room temperature and at the estimated disk operating temperature of 200 C. A comparison of the upper limits of the crack growth rate scatter bands for coupons from these two test groups is presented in Figure 4.6. As can be seen from this graph, the crack propagation rates at the elevated temperature are significantly faster than those observed in the room temperature tests. The dependance of the crack propagation rates on test temperature has been well documented in the literature [6][7][10][85][86][87]. However, the difference in the crack propagation rates determined in this study are quite large given the relatively small difference in test temperature. It is thought that the main reason for this difference was a reduction in the modulus of elasticity of the material at the elevated temperature. Since the test temperature of 200 C was less than a third of the melting temperature of the disk material, it is doubtful that there was any creep interaction present. It is possible however that the higher test temperature increased the oxidation rate of the fracture surface thus increasing the crack propagation rate somewhat.

The fact that there was a significant difference in the crack propagation rates as a

resulting from the difference in test temperature indicates that the temperature of the disks is an extremely important consideration when determining an RTS interval for these components. Therefore crack propagation rate tests should be performed at the estimated operating temperature of the component.

5.3 Relevance of Small Coupon Data to Full Scale Components

In the crack propagation rate testing performed for this project, several steps were taken in order to make the test results applicable to the full scale components. As mentioned previously in this report, the coupons used in these tests were machined from retired compressor disks. This was done to ensure that the material tested was representative of that used in actual service. This similarity included the basic material properties as well as any variation in microstructural characteristics caused by commercial heat treatment and manufacturing process. The material was taken from retired components such that the service induced degradation of material properties was also taken into consideration. The coupons were also machined close to the bolt hole area of the disks where the LCF cracking was most prevalent during service. This was done to ensure that the material being tested had been subjected to the same loading and temperature cycles as that seen by the cracked components during service. In addition, the prominent direction of crack propagation, ie. radially propagating cracks from the bolt holes towards the bore was also simulated in the test coupons.

The test parameters selected for the crack propagation were also chosen to represent the estimated stress conditions of the disk under operating conditions. To this end, the applied load, the stress ratio and the cycling frequency of loading were selected to conduct testing under low cycle fatigue conditions. Fractographic analysis of the service induced component cracks had indicated that these cracks propagated under LCF conditions [28][85][88]. The environment was also modeled reasonably well by the laboratory air at an estimated operating temperature of 200 C.

Despite these measures, there were several discrepancies between the laboratory and the service induced crack propagation processes. One of the most significant discrepancies between the two cases was the fact that the coupon tests were conducted under a constant

amplitude, tension-tension loading cycle. In order to more accurately model the in service situation, a loading spectrum typical of that experienced by the engine in flight would have to be applied instead of the constant amplitude loading that was applied in this instance. Secondly, the tests were conducted under tensile loads only, whereas the actual disk bolt holes would be subjected to significant bending and flexural loads resulting from the torque transfer through the compressor tie bolts. In addition, it is likely that the stress cycles applied at the crack tip of a service induced crack would also include a compressive stress. This compressive stress could result from the contraction of the components during cooling if the opposing fracture surfaces fail to line up exactly. The plastic zone ahead of the crack tip can also cause a compressive stress at the crack tip when the load is released. This stress is caused by the elastically deformed material returning to its undeformed state and exerting a stress on the plastically deformed zone which by definition, does not return to its original unstressed state. These compressive stresses can also introduce closure effects on the crack tip. None of these effects can be accurately modelled with a tension-tension type of loading sequence.

Other loading parameters which were not taken into consideration include the variation in loading rate, the effect of minor cycle-major cycle interaction and the effects introduced by the past loading history of the component [85][86][87][89][90][91][92][93]. It is possible that the occurrence and order of occurrence of small and large stress cycles could effect the resultant crack propagation rate. These effects can be seen by an apparent reduction in crack growth rate after an over stress has been briefly introduced in the component as reported in references [87][89][90][91].

Another factor that can not be accurately taken into consideration when using small coupons for testing is the actual size difference between the coupon and the component. In some cases the stress state in the coupon may in fact be more severe than that experienced in the actual component because of its reduced capability to dissipate the heat generated at the crack tip. The coupon boundaries may also pose unrealistic constraints to the local stress state at the crack tip of a test coupon.

Despite these differences between the test and service induced cracking, the tests performed in this study were important in that the effects of service exposure time and temperature on the crack growth rate properties of the compressor disk material were determined.

5.4 Prediction of a RTS Interval for Disks

5.4.1 Standard DFM Approach

Having obtained the worst case crack propagation rates from the equations for the upper limits of the scatter bands (equations 4.3, 4.5, 4.7), the corresponding RTS intervals were calculated for constant stress intensity conditions using the equation:

$$RTS = \frac{a_{dysf} - a_{detect}}{C(\Delta K)^n M_f} \quad (5.1)$$

Substituting the values of a_{dysf} , a_{detect} and M_f as described in Chapter 2, we get the equation:

$$RTS = \frac{(4.5 \times 10^{-3} - 1.75 \times 10^{-3})}{C(\Delta K)^n (6)} \quad (5.2)$$

where the numerator is in meters and the denominator has the units meters per cycle multiplied by the number of LCF cycles per mission hour.

Substituting the high time, room temperature test results (equation 4.3) into equation 5.2 with a stress intensity factor of $61 \text{ MPa}\sqrt{\text{m}}$ as estimated in Section 4.4 yields an RTS interval of 350 hours. A similar substitution with the results from the high time elevated temperature tests and the medium time elevated temperature tests yield RTS intervals of 178 hours and 285 hours respectively.

Looking at the results of the standard DFM approach first, the room temperature high time disk results provided an RTS interval of 350 hours. The most severe case was that using the high time disk elevated temperature test results. These results provided a RTS interval of 178 hours. The medium time disk also tested at the estimated operating temperature of 200.C provided a RTS interval of 285 hours. The Safe Inspection Interval (SII) provided by the engine manufacturer, General Electric, was 555 hours for the fifth stage disk. How this SII was obtained is unknown, however it was speculated in Chapter 1 that General Electric performed a Weibull analysis on the components which had cracked in service to arrive at this number. It is also possible that a limited number of spin pit tests were performed in order to confirm these values. Regardless of how the SIIs were obtained, the 555 hour interval is significantly greater than those determined using the results of this study. However, as pointed out in previously, the standard DFM approach used to

determine these numbers is likely to give a very conservative number in that all of the LCF cycles are considered to be similar to the start up and shut down cycles.

5.4.2 A Modified DFM Approach

The standard DFM approach results as determined above represent the severest possible disk loading conditions because each of the six LCF cycles per flight hour is treated as being equivalent to an engine start up and shut down cycle. If however, the six LCF cycles per flight hour are considered to be due to major throttle excursions experienced during flight maneuvers, the resultant stress amplitude applied to the disk could be significantly less than that experienced during actual engine start up and shut down. If we assume for the purposes of this study that typical in flight throttle perturbations result in stress amplitudes and therefore stress intensity factors at the bolt hole crack tip that are approximately equivalent to 50% of those experienced during engine start up and shut down, then equation 5.1 can be modified to better represent the actual service conditions. This modification results in an equation of the form:

$$RTS = \frac{a_{dysf} - a_{detect}}{[C(\Delta K_1)^n + M_f C(\Delta K_2)^n]} \quad (5.3)$$

where ΔK_1 represents the stress intensity factor experienced during engine start up and shut down and ΔK_2 represents the stress intensity factor resulting from in flight throttle excursions.

Substituting the values of the variables as determined in Chapter 2, into equation 5.3 and taking

$$\Delta K_2 = 0.5\Delta K_1 \quad (5.4)$$

the equation

$$RTS = \frac{(4.5 \times 10^{-3} - 1.75 \times 10^{-3})}{[C(61)^n + 6C((0.5)(61))^n]} \quad (5.5)$$

is obtained. Because the tests conducted in this study were all done at a stress ratio of 0.1, the effect of the six minor cycles is actually over estimated given that the minor cycles would typically have a stress ratio of greater than 0.1. However this calculation reflects the deterministic approach followed in this study by assuming that the stress ratio of the minor cycles is as severe as that of the major cycles. Substituting the values of C and n from the high time disk room temperature test results as presented in equation 4.5, yields an RTS

¹ Method of approximation in Appendix

interval of 1175 Hours. Similar substitution with the high time elevated temperature test results and the medium time elevated temperature test results yields RTS intervals of 669 hours and 1022 hours respectively.

Now looking at the results of the modified DFM approach, the RTS interval determined using the high time, elevated temperature test results was 669 hours. This is the interval of primary interest because it represents the fastest and thus worst crack propagation rate determined in this study. Given the nature of the methods used to estimate the crack tip stress intensity factor, the 669 hour interval compares quite favourably with the 555 hour interval supplied by General Electric. It would therefore appear that the modified DFM approach is capable of providing a reasonable estimate of a possible RTS interval given the crack tip stress intensity factors resulting from both typical in flight throttle excursions and engine start up and shut down are not the same.

5.4.3 Comparison of DFM Results with Engine Manufacturers Predictions

In Sections 5.4.1 and 5.4.2, a total of six RTS intervals were determined using the crack propagation test results obtained in this study. These are presented in Table 5.1. Two different approaches were used, the simple DFM approach which was developed before the effects of small amplitude LCF cycles were fully recognized, and a modified DFM approach which takes into consideration the differences between the LCF cycles resulting from engine start up and shut down and major throttle excursions experienced during a typical flight. Although the RTS interval is extremely dependant on the stress intensity factor experienced by the component at the crack tip, an accurate determination of this variable was not obtained during this study. Despite this shortcoming, the estimate of the crack tip stress intensity factor obtained in Chapter 2 and Section 4.4 shall be employed to provide an RTS interval which can be compared to the Safe Inspection Interval (SII) supplied by General Electric, the engine manufacturer. This comparison has been made in order to determine if the RTS intervals estimated in this study are of the same order of magnitude as those supplied to the CAF by General Electric. It should not be assumed that the theories or the methods employed in this study are in any way similar to the analysis performed by General Electric to determine the SII supplied to the CAF.

5.5 Uncertainties in Implementing a RTS Interval Based on Coupon Testing

The crack propagation results obtained in this work could be applied to determine a possible RTS interval for the fifth stage compressor disk of the J85-CAN15/40 engine as demonstrated in Section 5.4. This interval could then be used for economic assessment, verification of test procedures and analysis methods and confirmation of the damage tolerant design based maintenance methodology through careful tracking of components during service. The tests performed in this study are an important step in the final determination of a RTS interval which could be applied to a component that would subsequently resume service. It should be noted however that such an application of this methodology could only succeed through extremely rigorous testing under widely varied test conditions, and by the appropriate statistical analysis of the results. Even before reaching this stage, accurate determination of the stress state at the crack tip and of the actual crack propagation rate behaviour would have to be accomplished.

Various assumptions made for the purposes of this work, such as a relatively constant crack tip stress intensity factor and resultant crack propagation rate over the crack length of interest, is clearly an over simplification of the actual situation. Regardless of the amount of testing performed and of the accuracy of the analysis employed to determine operating conditions in the components, there will still be uncertainties as well as simplifications and assumptions made in order to make the development of a damage tolerance based lifing procedure feasible at all. These assumptions and simplifications will then have to be taken into consideration through the use of a safety factor. The determination of the appropriate magnitude for this safety factor is yet another issue that remains to be resolved. Obviously, the ideal safety factor would yield a RTS interval which was both perfectly safe and economical, when in reality such a number may not readily be available. However, by performing research in this area, many of the unknowns associated with the lifing of gas turbine components can be eliminated. This knowledge can then be applied to new components made from damage tolerant materials that may have properties which retard LCF crack initiation and propagation in actual components and can tolerate larger flaw sizes.

The work reported in this thesis is a step in improving the knowledge the factors which

determine the life of gas turbine engine rotating components. The results reported herein can be used as a step in the development of an experimental procedure through which improved predictions regarding gas turbine component lives may be made.

Chapter 6

Conclusions and Recommendations for Further Work

6.1 Conclusions

Low cycle fatigue crack propagation rate testing was performed on compact tension specimens machined from retired J85-can40/15 fifth stage compressor disks, with the goal of determining a return to service interval for this component. The effect of service temperature and service exposure time on crack propagation rates in these disks were also investigated. A direct current potential drop technique was used for crack growth rate measurements in this study. From the results of this work the following conclusions can be drawn:

- 1) The direct current potential drop technique for crack length measurement was successfully assembled, calibrated and employed in this work, resulting in a precision of 0.0524 millimeters.
- 2) The LCF crack propagation rate was determined to be temperature dependent in AM355 compressor disk material. The crack propagation rates increased with an increase in temperature from room temperature to 200 C. (Approximately a third of the melting point temperature of the alloy). Thus performing future tests at the estimated operating temperature of the component is strongly recommended.
- 3) The LCF crack propagation rate also indicated a dependency on the service exposure time of the component. The crack propagation rates increased with increasing service time of the disk. This may necessitate a decrease in the RTS interval as service exposure increases.
- 4) A formula based on DFM damage tolerance principles was developed to determine the duration of an RTS interval that takes into consideration the initial engine start up and shut down cycle as well as those cycles induced by major throttle excursions during a typical aircraft mission. The formula requires the knowledge typical stress intensity

factors expected to be operative at the crack tip during each of the two different cycle applications in order to provide the RTS interval in hours.

- 5) A return to service interval, using the formula mentioned above and the "worst case" crack propagation rate results, was calculated to be approximately 669 hours. This agrees quite well with the value of 555 hours supplied by the engine manufacturer.

6.2 Recommendations for Future Work

One of the most vital areas requiring future effort, is the accurate determination of stress intensity factors experienced at the crack tip during engine operation. This will probably be best accomplished through a rigorous three dimensional finite element method analysis of the bolt hole region or through the use of boundary element methods. In order to be representative of the actual service conditions however, the analysis should include consideration of the bolt hole torque transfer, blade mass and loading and aerodynamic loads.

A second area of interest is a more rigorous determination of the apparent dependency of crack growth rate on the length of component service exposure. This could be accomplished by performing tests such as those discussed in this report, on coupons machined from a zero time disk, and comparing the results with service exposed disk coupons.

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Table 1.1
Safe Lives and Safe Inspection Intervals Specified by G.E. for
Canadian J85 Components

<i>Component</i>	1981		1983	
	<i>Safe Life</i>	<i>SII</i>	<i>Safe Life</i>	<i>SII</i>
	<i>Hrs.</i>	<i>Hrs.</i>	<i>Hrs.</i>	<i>Hrs.</i>
<i>4th Stage Disk</i>	2,333	555	3,000	714
<i>5th Stage Disk</i>	1,777	555	2,285	714
<i>6th Stage Disk</i>	2,222	722	2,857	928
<i>7th Stage Disk</i>	2,222	722	2,857	928
<i>8th Stage Disk</i>	2,944	722	3,786	928

Table 1.2
 Variation in the Safe Life of the Fifth Stage
 Compressor Disk from May 1970 to Oct. 12, 1983.

<i>Date</i>	<i>Safe Life J85 5 Hours</i>	<i>Mission Severity Factor</i>	<i>Safe Life CF Hours</i>
<i>May 1970</i>	3,000	1.00	3,000
<i>June 1978</i>	4,500	0.94	4,787
<i>Sept. 1, 1979</i>	4,000	0.94	4,255
<i>Sept. 17, 1979</i>	4,400	0.94	4,680
<i>Nov. 24, 1981</i>	3,200	1.80	1,777
<i>Oct. 12, 1983</i>	3,200	1.40	2,285

Table 1.3
Canadian J85 Component Service Life Data

<i>Component</i>	<i>Safe Life Limit Hrs.</i>	<i>Safe Inspection Interval Hrs.</i>	<i>Longest Life Component Hrs.</i>	<i>Mean Life Hrs.</i>	<i>Average Life Extension Hrs.</i>
<i>Compressor Disks</i>					
<i>4th Stage Disk</i>	2333	555	4428	3863	1530
<i>5th Stage Disk</i>	1777	555	4473	3875	2098
<i>6th Stage Disk</i>	2222	722	5066	3971	1749
<i>7th Stage Disk</i>	2222	722	5066	3813	1591
<i>8th Stage Disk</i>	2944	722	5066	4080	1136
<i>Turbine Disks</i>					
<i>1st Stage Disk</i>	1111	555	3738	2364	1253
<i>2nd Stage Disk</i>	1111	555	3792	2971	1860

Table 3.1
Summary of Potential Drop Parameters.

<i>Parameter</i>	<i>Magnitude</i>	<i>Unit</i>
<i>Current</i>	10	<i>Amps</i>
<i>Amplifier Gain</i>	1000	-
<i>Current Stabilization Time</i>	2	<i>Sec.</i>
<i>Data Acquisition Time :</i>		
20 - 30° C Tests	2	<i>Sec.</i>
200° C Tests	1	<i>Sec.</i>
<i>Data Acquisition Rate</i>	10	$\frac{1}{\text{Sec.}}$

Table 3.2
Results of Composition Analysis.

<i>Elements</i>	<i>Percentage by weight</i>		
	<i>min</i>	<i>max</i>	<i>test</i>
<i>Carbon</i>	0.100	0.150	0.110
<i>Manganese</i>	0.500	1.250	0.860
<i>Silicon</i>	--	0.500	0.210
<i>Phosphorus</i>	--	0.025	0.012
<i>Sulfur</i>	--	0.025	0.011
<i>Chromium</i>	15.000	16.000	15.620
<i>Nickel</i>	4.000	5.000	4.280
<i>Molybdenum</i>	2.500	3.250	2.530
<i>Nitrogen</i>	0.070	0.130	--
<i>Other elements</i>	--	0.500	--
<i>Iron</i>	<i>Remainder</i>	<i>Remainder</i>	--

Table 3.3
Physical Constants of AM-355

<i>Constant</i>	<i>SI</i>	<i>Imperial</i>
<i>Specific Gravity</i>	7.79	7.79
<i>Density</i>	7778 kg/m ³	0.281 lb/in ³
<i>Electrical</i>		
<i>Resistivity at 28°C</i>	75.73	microhm - cm
<i>Thermal</i>		
<i>Conductivity at 38°C</i>	49.5 W/m ² K	8.72 BTU/ft/hr/F
<i>Thermal Coef. of Expansion 27 - 100°C</i>	11.5/Cx10 ⁻⁶	6.4/Fx10 ⁻⁶

Table 3.4
The Mechanical Properties of Forged AM-355

<i>Property</i>	<i>SI</i>	<i>Imperial</i>
<i>UTS</i>	1170 MPa	170 KSI
<i>Yield Stress (0.2%)</i>	1070 MPa	155 KSI
<i>Modulus of Elasticity (27°)</i>	188 GPa	29300 KSI
<i>Elongation, in 2"</i>	14 %	
<i>Reduction in Area</i>	42 %	
<i>Rockwell Hardness C</i>	37	

Table 4.1
Results of Tensile Tests

Test Number	UTS MPa (KSI)	Yield Stress MPa (KSI)	E GPa (KSI)
1	1272 (185)	1129 (164)	148.5 (21500)
2	1247 (181)	1112 (161)	151.8 (20552)
3	1248 (181)	1071 (155)	149 (21552)

Table 5.1
Results of RTS Determination

	<i>Standard DFM</i>	<i>Modified DFM</i>
	<i>Hours</i>	<i>Hours</i>
<i>High Time Disk at Room Temp.</i>	350	1175
<i>High Time Disk at 200 C</i>	175	669
<i>Medium Time Disk at 200 C</i>	285	1022

Figure 1.1 Illustration of Crack Initiation and Propagation Segments of Fatigue life. [1]

Figure 1.2 Illustration of Typical Fatigue Data Scatter and Normal Distribution. [1]

Figure 1.3

Illustration of Possible Component Lifetimes Remaining at Time of Retirement. [1]

Figure 1.4 Illustration of RFC Lining Philosophy. [1]

Figure 2.1 Illustration of Statistical Variation of Strength and Applied Stress. [24]

Figure 2.2 Illustration of the Three Regions of Crack Growth.

Figure 2.3

Illustration of Weibull Procedure for Determining CPI. [24]

Figure 2.4

Illustration of 2-D FEM Mesh. [24]

Figure 2.5 Graph of 2-D FEM Stress Analysis Results. [21]

Figure 2.6 Graph of 3-D FEM Stress Analysis Results. [21]

2

OF/DE

2

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS
STANDARD REFERENCE MATERIAL 1010a
(ANSI and ISO TEST CHART No. 2)

Figure 2.7 Results of Plastic Zone Hardness Testing Around Tip of Service Induced Crack. [24]

Figure 2.8 Fractographs of Banding Phenomenon Observed on Fracture Surfaces of Service Induced Cracks.

Figure 2.9

Illustration of Crack Length Used to Avoid Tertiary Region of Crack Growth Rate.

Figure 2.10 Striation Spacing Observed on 19 mm Crack Fracture Surface. [24]

Figure 2.11 Photographs of Precipitate Morphology Observed in Fifth Stage Disk.

A) away from crack tip, B) near crack tip [24].

Principle of Operation

$$R = \rho \frac{l}{A}$$

Figure 2.12 Illustration of Potential Drop Principle.

Figure 2.13 Illustration of Reference Coupon Test Set Up.

Figure 2.14 Position of Potential Drop Probes and Leads on CT Specimen.

Figure 3.1 Comparison of High Temperature and Room Temperature Fractographic Surfaces.

Figure 3.2 Picture of Test Set Up Showing Method of Test Frame Isolation.

Figure 3.4 Picture of Tensile Test Machine

Figure 3.5 Geometry of Compact Tension Specimen.

Figure 3.6 Orientation of CT Specimens as Machined from Compressor Disks.

Figure 3.7 Applied Load Cycle Waveform.

2 HOLES
 $0.25 + 0.002$
 $- 0.000$

• ALL DIMENSIONS IN INCHES
 • "A" SURFACES MUST BE PERPENDICULAR AND
 PARALLEL AS APPLICABLE TO WITHIN 0.002 IN. TIR.

TOLERANCES

Fractions	$\pm \frac{1}{64}$
.XX	$\pm .010$
.XXX	$\pm .005$
Angles	$\pm 1^\circ$

Designed

G. AKER

Drawn

Project

Scale

Etchelle

Date

National Research Council Canada
 Conseil national de recherches Canada
 NATIONAL AERONAUTICAL ESTABLISHMENT
 STRUCTURES AND MATERIALS LABORATORY

Material

Manufacturer

Quantity

Part Number

Next Assembly

Drawn By

2

103

Figure 3.9 Picture of Furnace in Operation.

Figure 3.10a Vertical Temperature Profile of Furnace.

Figure 3.10b Horizontal Temperature Profile of Furnace.

Figure 3.11 Picture of Complete Test Set Up.

Figure 4.1 Potential Drop Calibration Curves for CT Specimen.

Figure 4.2 Stress Strain Curves for Three-Tensile Tests.

Figure 4.3 Crack Growth Rate Results for Medium Time Disk Coupons at 200 C.

Figure 4.4 Crack Growth Rate Results for High Time Disk Coupons at Room Temperature

Figure 4.5 Crack Growth Rate Results for High Time Disk Coupons at 200 C.

Figure 4.6 Comparison of Upper Bound Lines for High Time Disk Results at Room Temperature and at 200°C.

Figure 4.7 Comparison of Upper Bound Lines for High and Medium Time Disks at 200 C.

Figure 4.8 Striation Spacing Observed on Other Service Induced Cracks.

Figure 4.9 Plot of Striation Measurements from Two Room Temperature High Time CT Specimen Tests Versus Stress Intensity Factor.

Figure 4.10 Fractographs of Room Temperature Test CT Coupon Fracture Surfaces

Figure 5.1 Plot of Potential Drop versus Optically Measured Crack Lengths for Room Temperature.

Figure 5.2 Plot of Potential Drop versus Optically Measured Crack Lengths for Elevated Temperature Test.

Figure 5.5 Results Reported by Mr. D. Raizenne With Scatter Band Determined by High Time Disk Tests at Room Temperature.

Appendix

In order to allow an RTS interval to be calculated using the Modified DFM approach, an estimate was made of the relative severity of the stress intensity factors resulting from the initial start up cycle, ΔK_1 and those resulting from major throttle excursions experienced during a typical flight, ΔK_2 . The method that was used to make this estimate is as follows:

First of all, a typical SEM fractograph of a service induced fracture surface showing the "banding" effect described in the text was selected. (see Figure 2.8 for example) It was assumed that the larger striations on this fracture surface were the result of the engine start up cycle and that the smaller striations were the result of the cycles incurred by throttle excursions experienced during a flight. The width of the larger striations was measured and the results were taken as an estimate of the crack propagation rate, da/dN . This value was then substituted in to the mean line equation (equation 4.6) determined from the high time disk, elevated temperature crack propagation test results to yield a value for the stress intensity factor, ΔK . Similarly, the distance *between* the larger striations was also measured and divided by the six smaller cycles expected per mission flight hour as determined by the mission profile analysis. This assumes that the time between engine start ups was approximately one hour. This value was also substituted into equation 4.6, to provide an estimate of the stress intensity factor resulting from these smaller cycles. This value was approximately half of that determined for the engine start up cycles so that $\Delta K_2 \simeq 0.5\Delta K_1$.

END

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