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ABSTRACT

This paper discusses the need for understanding reliability within the context of helicopter structures and presents a case for why such understanding is essential to successfully implementing better usage monitoring programs.

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Executive Summary

1. For helicopters, the safe-life methodology is used to determine when a fatigue life-limited structural component needs to be replaced.

2. The safe life of a component is defined as being reached when the risk of fatigue failure is considered to be unacceptable.

3. Predicted safe lives can be increased by very large amounts if the accepted level of reliability is permitted to drop.

4. Premature failure of critical helicopter components is likely to result in the loss of the aircraft and crew so the 'acceptable' level of risk has to be extremely low.

5. Surprisingly, relatively little work has been done to quantify the risk of failure or conversely the reliability (probability of non-failure).

6. There is a perception that the current methods used to calculate lives are very conservative (highly reliable) and that may well be true but some authors question the analysis methodology that has been used in support of the six-nines reliability that is sometimes claimed in the field.

7. The quantification of reliability is very difficult; the processes and parameters that influence the calculation of safe life are inherently stochastic in nature and many sources of variability arise.

8. The capability to quantify reliability becomes extremely important when advanced structural component retirement philosophies (eg. retirement of individual components according to their individually measured usage) are considered.

9. While no attempt is made in this report to examine how reliability can be quantified, a strong case is made regarding the need for such quantification and for further research in the field.

Contents

1. Introduction

Sooner or later, all systems fail.

This simple sentence encapsulates the reasons why a lot of effort is expended to determine the life span of systems ranging from mechanical and electronic systems that we create to biological systems found in nature. We know that a system will fail; what we often don't know is *when* the system will fail. For important or critical systems, we would be uncomfortable in using them without having an estimate of their time to failure. Hence, manufacturers specify the life of their system in terms such as hours, or operating cycles. When these lives are reached, some maintenance action is required that could range from a cursory visual inspection to overhaul to replacement. However, what maintenance manuals fail to show is that each stated life is associated with a defined level of reliability[1].

This paper discusses the need for understanding reliability within the context of helicopter structures and presents a case for why such understanding is essential to successfully implementing better usage monitoring programs.

2. Background

Many structural components in the dynamic system[2] of a helicopter have lives that are limited by fatigue strength considerations. For helicopters, the safe-life methodology is used to determine when a fatigue life-limited structural component needs to be replaced. The safe life of a component is defined as being reached when the risk of fatigue failure is considered to be unacceptable. Premature failure of critical helicopter components is likely to result in the loss of the aircraft and crew so the 'acceptable' level of risk has to be extremely low. Overall risk (probability of premature failure), f, and reliability, r, are simply related:

$$f = 1 - r$$

In an ideal world, r would equal 1 and f would equal 0, but in practice this is not achievable. Two questions are of great importance:

1. What level of reliability is considered acceptable to operators (this could vary from one operator to another)?

2. What level of reliability is being achieved with current helicopter design and lifing practice (this could be highly variable)?

In respect of question 1, it has been widely stated that the retirement of a fatigue life-limited helicopter structural component should be based on the life at a reliability of 0.999999, commonly known as the six-nines or 0.9_6 reliability. The implication of this criterion is that only one out of a million components should fail prior to the designated retirement life. As there are a number of fatigue life-limited components in a helicopter, the reliability of the complete helicopter would be much lower than six-nines if the six-nines reliability were to

[1] The definition adopted here for reliability is the probability that a functional unit will perform its required function for a specified interval under stated conditions.

[2] The structural components referred to here are those in the rotor system, flight control linkages and the rotor masts (which are usually integral parts of the rotor gearboxes).

apply to each critical life-limited component. The six-nines reliability and related topics are considered by a number of authors [1-6].

In respect of question 2, a number of helicopter manufacturers claim to be achieving six-nines reliability with current design and lifing practice. Tong and Wang [7], and Polanco [8] postulate that the manner in which reliability has been calculated leads to significant over-estimation.

Reliability assessment of helicopter component lives is a complex issue that appears to be not well understood by the helicopter community. It should be central to the calculation of component safe lives but that hasn't been the case. There is the perception that the current methods used to calculate lives are very conservative, but the level of conservatism (ie. reliability) is rarely subject to in-depth studies. A stated life without a stated level of reliability is fairly meaningless, as predicted safe lives can be increased by very large amounts if the accepted level of reliability is permitted to drop.

3. Helicopter Structural Component Lifing Methodology

The defined replacement lives for structural components in the dynamic system of helicopters are usually termed CRTs (Component Retirement Times). Such CRTs are calculated using safe-life methodologies. Figure 1 shows a simplistic representation of the safe-life methodology.

As shown in Fig. 1, for each component, a damage hypothesis is used to combine a fatigue spectrum applied to the component with the fatigue properties of the material from which the component is made. The CRT is the output from the damage hypothesis. The methodology is described in more detail below.

3.1 Fatigue spectrum

The fatigue spectrum shown in Fig. 1 may be expanded as shown in Fig.2. The spectrum is created by combining a design usage spectrum with a set of corresponding flight loads applicable to each flight condition in the design usage spectrum.

3.1.1 Design Usage Spectrum

The design usage spectrum attempts to encompass the worst aspects of all the anticipated roles that the aircraft will undertake. It is usually specified in a form such as that shown in Table 1. The spectrum consists of a list of flight conditions and a set of corresponding percentage times and/or number of occurrences.

The list of flight conditions and the associated percentage times and occurrences are mostly obtained from data derived through qualitative methods such as questionnaires filled out by aircrew. In some cases, quantitative data that are derived through methods such as in-flight recording of flight conditions via electronic data acquisition systems, are used.

The usual method of creating a design usage spectrum is to obtain usage spectra for each role that the helicopter is anticipated to undertake (e.g. transport, search and rescue, anti-tank, anti-submarine) and then to combine these individual mission spectra into a final spectrum. Figure 3 shows the methodology as applied to the creation of a usage spectrum

for the Australian S-70A-9 Black Hawk. This is an example of the application of a worst-case methodology.

3.1.2 Flight Loads

The flight loads are usually those measured on a prototype aircraft, which is flown through the flight conditions specified in the usage spectrum so that corresponding component loads can be measured. Some of the flight conditions are repeated many times, while others may only be repeated once; this is driven by time and cost constraints. The manufacturer's flight test engineers attempt to ensure that at least all the most critical flight conditions are repeated several times. Due to physical constraints, not all the components of interest can be instrumented with strain gauges, so the manufacturer must determine the relationships between the loads in the instrumented and in the un-instrumented components. This is usually done by testing the appropriate component assemblies in static test fixtures.

Once the flight loads are recorded for each flight condition, they are then processed and converted into either "block" loads or "cycle-counted" loads. Figure 4 shows this process.

3.2 Material Fatigue Properties

The material fatigue properties are normally described in terms of S-N curves. To generate the S-N curve, the following steps are taken:

- A generic S-N curve is chosen that is appropriate for the component's material and the loading conditions to which the component is subjected (e.g. aluminium with chafing).
- Fatigue tests are conducted on a number of the components. These are usually constant-load-amplitude tests.
- A mean S-N curve is then plotted through the test results.
- Reduction factors on both life and stress are applied to the mean curve.
- Further reduction factors may be applied to the curve if the number of components tested is below a given value.
- The curve resulting from the application of these reduction factors is termed the "Working S-N" curve and is used in the fatigue analysis.

3.3 Damage Hypothesis

The standard damage hypothesis used is Miner's rule. This says that, for each load in the fatigue spectrum, the damage, D, incurred per unit time is given by:

$$D = \frac{n}{N}$$

where n is the estimated number of load cycles experienced at the specified load level.
N is the allowable number of cycles at the same specified load level and is defined by the working S/N curve for the component.

Using the subscripts

i ith flight condition in the fatigue spectrum
j jth load level in the ith flight condition (for block loads, j = 1, for cycle counting, $1 \le j \le k$, where k is the number of sub-blocks)

then the fatigue damage incurred by the ith flight condition on a particular component is given by Miners Rule as:

$$D_i = \sum_{j=1}^{k} \frac{n_j}{N_j}$$

The total damage is then given by:

$$D_{Total} = \sum_{i=1}^{m} D_i$$

Where m is the total number of flight conditions. For many components, most of the D_i values are zero. In other words, for most components, only a few flight conditions cause fatigue damage.

3.4 Component Retirement Times

By combining all the above, a CRT is calculated for a particular component and that CRT then becomes the retirement life for all those components, no matter in which aircraft they are installed. This means that the level of reliability of that CRT will vary; components on those aircraft in the fleet that are flown severely will have lower reliability levels than components on aircraft that are flown gently. The question arises as to what the level of reliability actually is for the CRT and how sensitive it is to the variability that is inherent in the above processes and parameters.

4. Sources of Variability

All the processes and parameters identified above are inherently stochastic in nature and the table on the next page indicates where the sources of variability arise in each one.

The variations listed in the table mean that the results arising from each process or the data corresponding to each parameter have an associated level of reliability. Quantifying the reliability of each process and parameter, and how they interact to produce an overall reliability for a CRT is the problem that needs resolution (Fig. 6).

Solving this problem is one of the key elements in implementing a fatigue damage tracking program for individual components and thus lifing each component according to the damage that it experiences rather than setting a global life for all components of the same type. Without understanding the interactions between the reliability levels of each step in calculating a CRT, there will be no way of knowing how the reliability of a CRT is going to be affected when changes are made to the lifing methodology.

Item	Source of variability
Design Usage Spectrum	
Aircrew surveys	Human memory
Recorded data	The existence of recorded data implies the presence of sensors and an on-board recording system. The sources of variability here are: • Sensor measurement errors • Installation errors (e.g. inappropriate placement of a sensor, or incorrect wiring leading to excessive electronic noise in the signal) • Calibration errors • Signal conditioning and processing errors (including any arising when converting the signal from analogue to digital form)
Worst-case methodology	In going from individual mission spectra to a fleet usage spectrum, a worst-case methodology inevitably changes the relative proportions of times between flight conditions (Fig. 5). The final spectrum often bears no relation to an actual flight.
Flight Loads	
Repeatability of flight condition – Pilot	A pilot is not able to reproduce exactly a flight condition. Small variations in airspeed, altitude, bank angle, etc., will lead to variations in the measured loads for two nominally identical flight conditions.
Repeatability of flight condition – Machine	Even if a pilot were able to reproduce exactly a flight condition and all ambient conditions were also exactly the same (e.g. wind speed, air temperature), the aircraft itself would not necessarily adopt the same state. Small changes in the configuration of the aircraft, arising from the manufacturing tolerances inherent in all machined parts, will cause variations in the measured flight loads.
Component Fatigue Strength	
Material properties	So called 'identical' components will exhibit different fatigue strengths. Some differences will arise due to manufacturing process variability (machining, surface finish, heat treatment etc.). Significant reduction in strength can occur if defects, even minor ones, are not picked up and the component rejected during inspection.
Curve shape	The shapes used by manufacturers for their SN curves are what they consider to be "best-fit" based largely on experimental fatigue failure data. This is a very imprecise process..
Reduction factors	Reduction factors are applied to SN curves to produce the "working SN" curve for a fatigue analysis. The working curve is intended to be drawn such that the probability of a fatigue failure at a point below the curve is considered to be 'acceptably' low and is thus meant to take into account fatigue strength variability. Even slight differences in these factors can produce large differences in the predicted safe life.
Damage Hypothesis – Miner's Rule	
Cycle-counting scheme	The type of method used to count load cycles will change the loads used in a fatigue analysis. Different methods will be more or less conservative, and so yield different results for the predicted safe life.
Spectrum sequence effects	Miner's rule does not take account of the sequence of loads and hence that information is not incorporated in a design usage spectrum. This is known to be incorrect in that the occurrence of large positive or negative stresses in a component can significantly affect the growth rate of fatigue cracks. Mean load effects may or may not be taken into account. If they are taken into account, a Goodman correction is often used.

5. An Example

So where is all this leading us? Some years ago, the Australian Army had a problem with its Black Hawk helicopters; the CRTs of some components were drastically reduced. In particular, two components had their CRTs provisionally reduced from 1800 and 2400 hours to less than 1000 hours. The problem arose because two-thirds of the fleet was already at or above 1000 flight hours. It would take another 12 months to provide definitive rather than provisional CRTs and new replacement components would also not become available for at least 12 months so the dilemma was: were the revised limits to be adhered to, and hence most of the fleet grounded or were the limits to be ignored until the definitive CRTs and/or new parts were received? The Army chose the latter option, but based its choice on a DSTO risk analysis showing the magnitude of the reduction in the reliability of the CRTs as a function of operating hours beyond the 1000 hour mark.

The risk analysis was necessarily crude because of the dearth of reliable supporting data available, the limitation in the authors' understanding of this complex topic, and the very short time available for the authors to respond. The analysis initially provided relative increases in the risk of overflying the 1000 hour mark, but the Army wanted absolute values. Saying that the risk has increased N-fold is not of much help when the magnitude of the baseline is not known. Hence, absolute reliability values were calculated. This is a much more difficult proposition and was, of necessity, also crude.

6. Concluding Remarks

The fatigue lifing methodology used to determine retirement lives for helicopter components depends on processes and parameters that are inherently stochastic.

CRTs thus have an associated level of reliability and understanding the reliability levels of each process and parameter and how they interact will provide an understanding of the reliability of a CRT. An understanding will also be gained about the sensitivity of the CRT reliability to variations in any of the inputs to the fatigue lifing methodology. An essential pre-requisite to implementing advanced fatigue damage tracking systems is the capability to quantify CRT reliability when such systems are used.

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Table 1: Example Usage Spectrum

Flight Condition	% Time	Occurrences per 100 hours
Hover	1.8	
Left or right sidewards flight	1.5	
Rearwards flight	0.8	
Climb	4.2	
Level flight at 20 knots	1.8	
Level flight at 40 knots	3.8	
Level flight at 70 knots	7.1	
Level flight at 100 knots	8.2	
Level flight at 120 knots	19.6	
Level flight at 140 knots	21.9	
Level flight at 150 knots	10.2	
Sideslip to the left or right	1.0	
Autorotation	1.3	
Partial power descent	2.4	
Dive	2.2	
Take off	0.6	380
Left hover turn	0.6	150
Right hover turn	0.6	150
Left turn at 30° angle of bank	3.3	800
Right turn at 30° angle of bank	3.3	800
Left turn at 45° angle of bank	0.5	155
Right turn at 45° angle of bank	0.5	155
Left turn at 60° angle of bank	0.1	52
Right turn at 60° angle of bank	0.1	52
Autorotation turns	0.5	80
Hover approach	0.5	400
Normal landing	0.3	350
Run-on landing	0.1	50
Pedal reversal – hover	0.1	110
Pedal reversal – level flight	0.1	294
Longitudinal cyclic stick reversal – hover	0.1	110
Longitudinal cyclic stick reversal – level flight	0.2	294
Lateral cyclic stick reversal – hover	0.1	110
Lateral cyclic stick reversal – level flight	0.2	294
Moderate pull-out	0.3	100
Severe pull-out	0.1	18
Droop stop pounding		500
Ground-Air-Ground load cycle		400
Total	100	

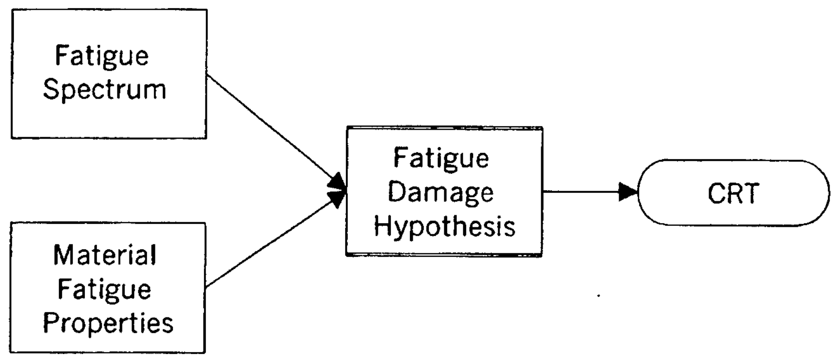


Figure 1: Simplistic representation of the safe-life methodology

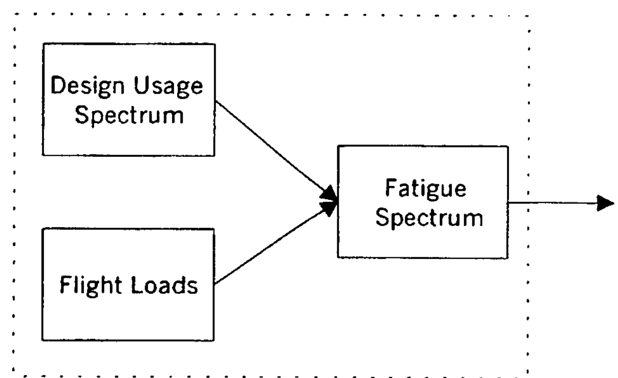


Figure 2: Inside the "Fatigue Spectrum" box of Figure. 1

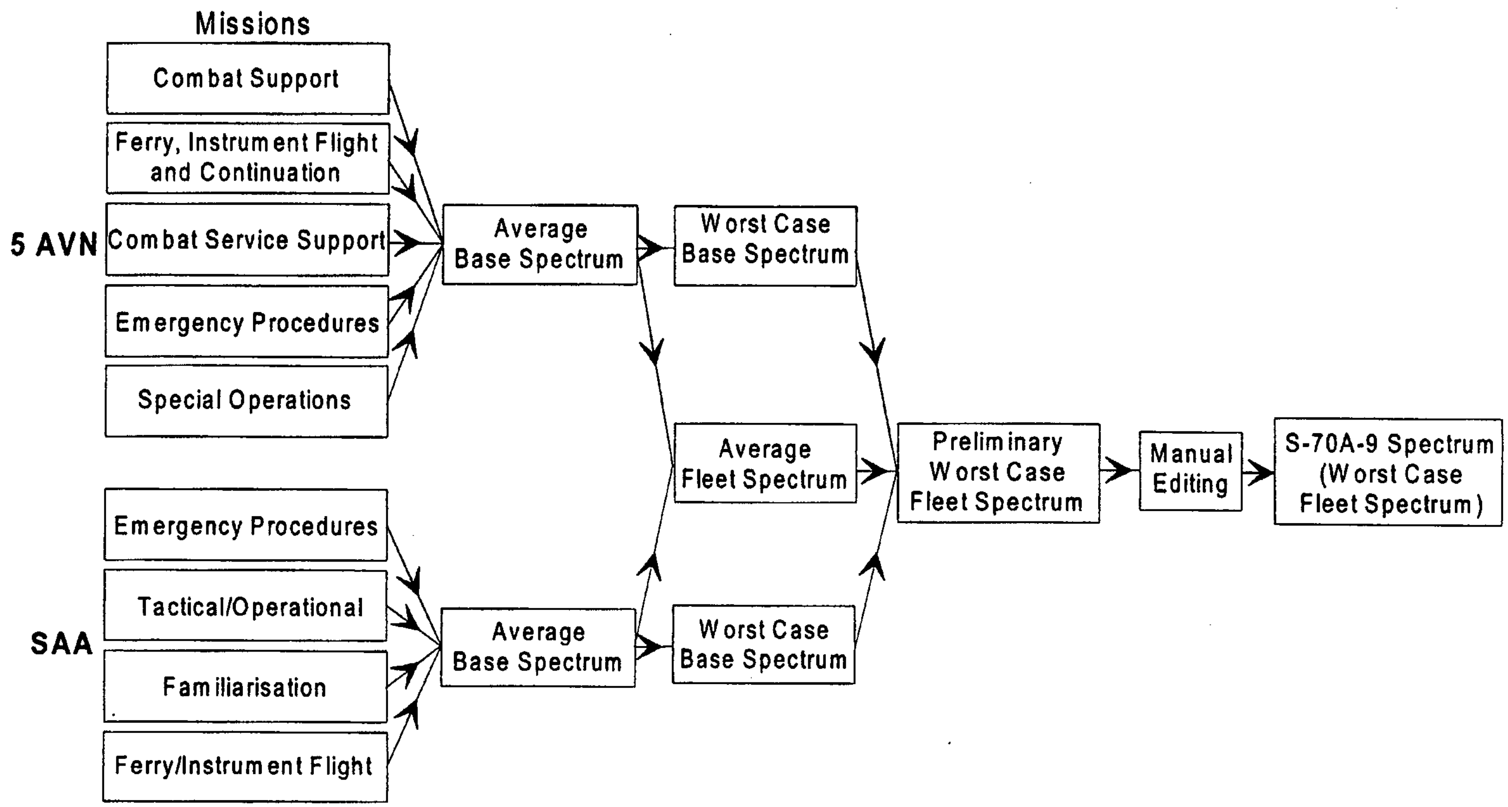


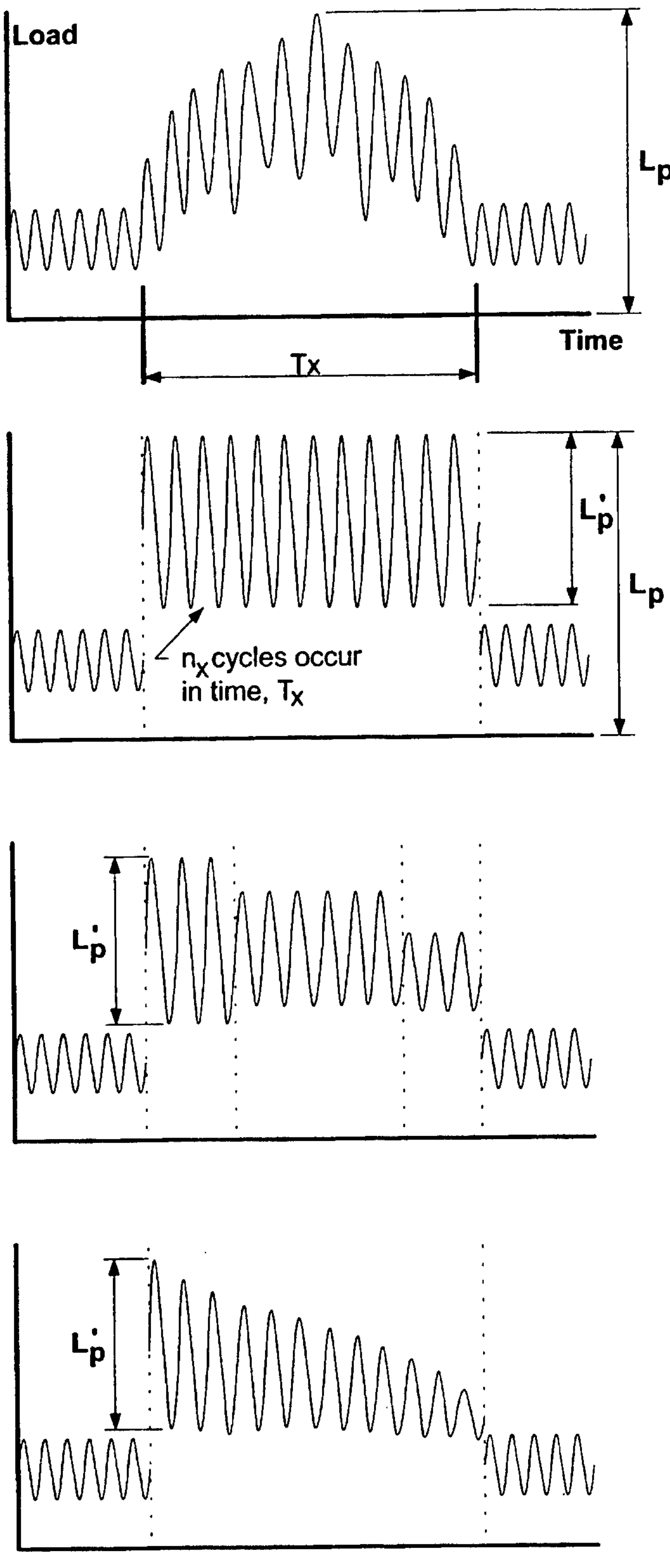
Figure 3: Creation of the S-70A-9 Black Hawk Usage Spectrum.

Step 1: The missions undertaken at the two main Black Hawk bases 5AVN (5 Aviation Regiment), Townsville and SAA (School of Army Aviation), Oakey were identified and usage spectra for each mission were created.

Step 2: The mission spectra were combined into an average spectrum for each base and a worst-case spectrum for each base.

Step 3: The average base spectra were combined into an average fleet spectrum.

Step 4: The average fleet spectrum was combined with the worst-case base spectra to produce a worst-case overall spectrum that was then slightly changed to become the worst-case fleet spectrum for the S-70A-9. This was then defined as the S-70A-9 usage spectrum.



(a) Loads measured during hypothetical flight condition X, which occurs over time period Tx and which has a peak load Lp as shown.

(b) Block Counting Method
The block counting approach takes the largest load cycle, here shown as L'_p (measured using a cycle counting scheme such as the rainflow method) and assumes that it applies over the entire time, Tx.

(c) Cycle Counting Method I
This counting method divides the flight condition load cycles into sub-blocks. In the example shown, Tx has been divided into three sub-blocks.The block counting method is then used in each sub-block. The total number of load cycles is still n_x (i.e. $n_{x_1}+ n_{x_2}+ n_{x_3}= n_x$) and the peak load used is still L'_p.

(d) Cycle Counting Method II
The least conservative of the three counting methods shown here, this method looks at each cycle individually, so n=1.

Figure 4: Converting a measured load time-history into block or cycle-counted loads data.

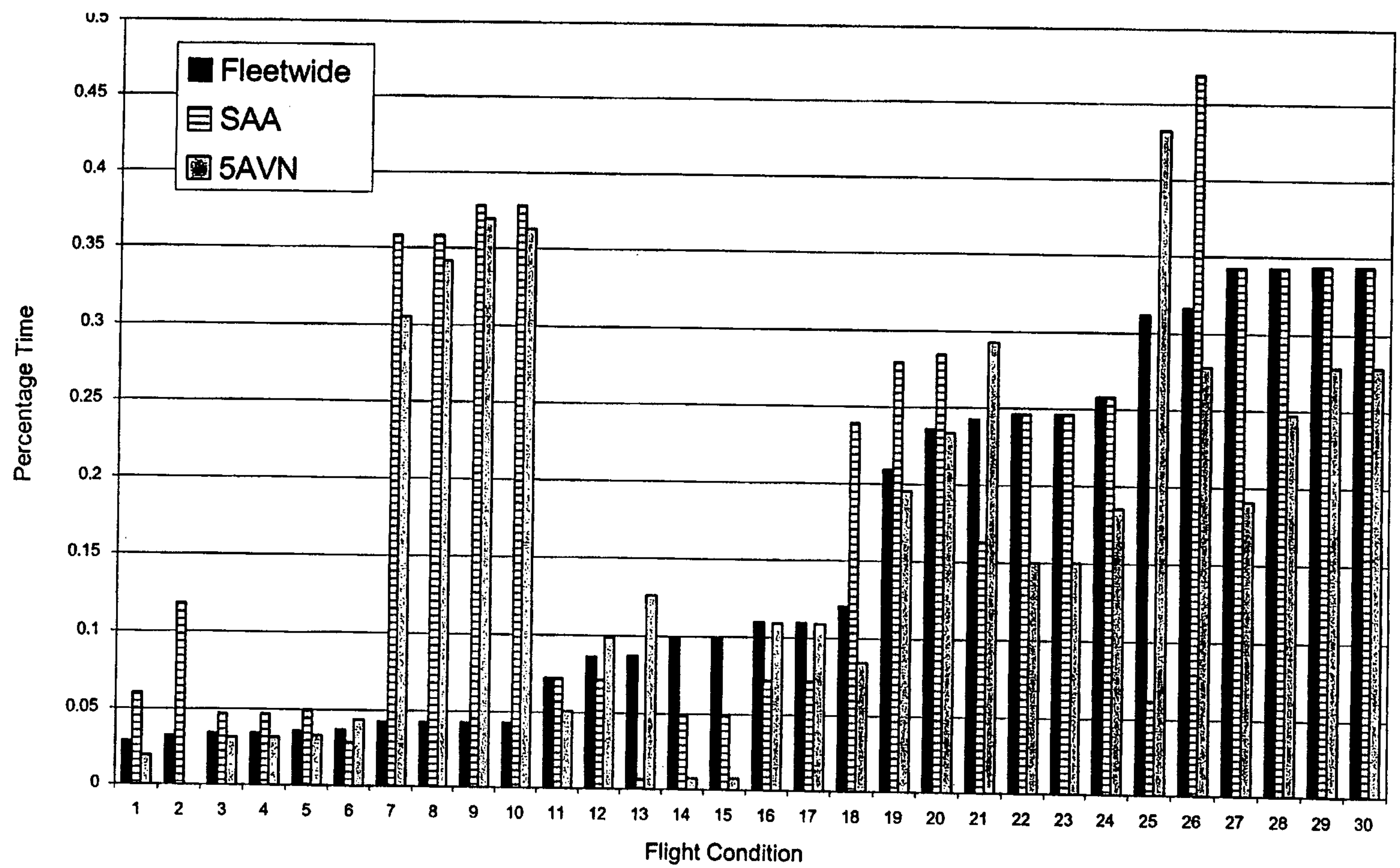


Figure 5: *Comparing the percentage times allocated to a sub-set of the flight conditions specified in the Australian Army Black Hawk usage spectrum. The fleetwide spectrum was created by combining the spectra for the Australian Army's two main Black Hawk bases: SAA (School of Army Aviation, Oakey) and 5AVN (5th Aviation Regiment, Townsville). The percentage time in the fleetwide spectrum is sometimes the same as that in the SAA or 5AVN spectra (e.g. flight condition 27), sometimes it is the average of those two spectra (e.g. flight condition 12), and sometimes it bears no relation to the times in either of those two spectra (e.g. flight condition 7). This is what typically happens when a worst-case methodology is employed to combine several spectra into one spectrum.*

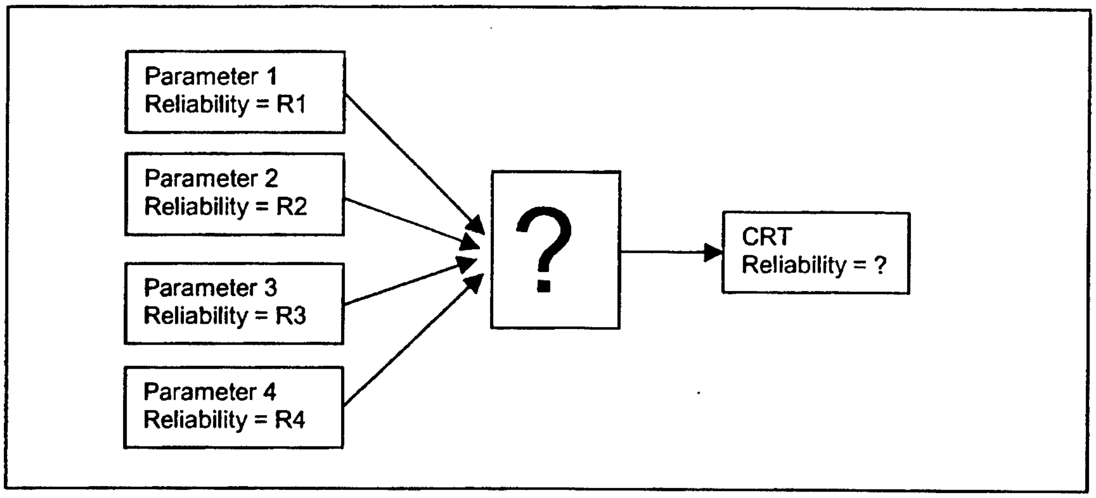


Figure 6: How do the reliabilities associated with each parameter or process interact and what is the overall reliability of the calculated CRT?

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