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Economic Life Determination for a Military Aircraft

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The U.S. Air Force introduced the concept of economic life of an aircraft structure in the early 1970s. They initially defined the economic life of an aircraft as the time when the structure had reached the point of widespread damage that was uneconomical to repair. The main difficulty with this definition is that structural analysts cannot determine when the economic burden is unacceptable. They can, however, make a determination of the economic burden from inspections, replacements, and repair of the structure. This paper demonstrates this process for the F-15E aircraft. For this assessment, the structural analysts generated probability distributions from durability test cracking. They also derived usage severity variations from recorded data. For each airplane, they sampled the population of the initial flaws as well as the severity of the usage for each area of the airplane that would contribute to the maintenance cost. They grew these cracks to the point of a repair or modification action. The cost from repairs and modifications was one factor that Air Force management could use to determine if the airframe was still competitive with other alternatives.

Introduction

IN 1958, in the aftermath of catastrophic fatigue failures of the B-47 aircraft, the U.S. Air Force (USAF) institutionalized a formal integrity program. They called this the Aircraft Structural Integrity Program (ASIP). The original program had three main goals:

1) Control structural failure in operational aircraft.
2) Devise methods of accurately predicting aircraft service life.
3) Provide design and test approaches that would avoid structural fatigue problems in future weapon systems.

These goals have not changed. They still constitute the basis for the program. The USAF has accomplished goal 1 quite effectively. The current rate of failure of USAF aircraft from structural causes is more than 100 times less than failures from all other causes combined. In addition, the USAF has successfully attained goal 3. In the early 1970s, the USAF adopted the damage-tolerance approach for structural design. This approach has considerably reduced the number of occurrences of fatigue problems in weapon systems designed since its adoption. The achievement of goal 2 has been much more difficult. Even today, it is not possible to state accurately the anticipated structural economic life for all of the USAF weapon systems.

One of the reasons for this is that it is rare that the USAF retires a population of their aircraft because of structural degradation from fatigue cracking. There have been, however, many cases of significant structural modification to preclude retirement. Examples of these include the B-52D, C-5A, KC-135, F-16, and C-130 aircraft. The USAF made the decision that these airplanes were still useful compared with alternatives. A structural consideration that *could* be a major factor in the decision to retire an aircraft is corrosion damage. For example, the KC-135s are incurring significant maintenance costs because of corrosion. The USAF procured these airplanes, and others, with corrosion-prone materials and inadequate processes to protect them from corrosion. There have also been projections of the time remaining on a given aircraft before fatigue cracking would be a significant economic concern. One example of this is the C-141. Before the USAF made the decision to stretch this aircraft, they made a determination of the lower boundary of the time when fatigue cracks constituted an economic concern. Another example is the B-52G/H. Before the USAF made a commitment to upgrade the avionics on these airplanes, they made a projection of the lower boundary of the time to significant fatigue cracking.

During the years following the 1958 structural failures up to the adoption of the damage-tolerance approach for ensuring structural safety in the early 1970s, the USAF used the so-called safe-life approach. The safe-life approach relied upon the results of the laboratory test of a full-scale aircraft. The USAF tested the structure with a spectrum of loading that represented the average (or baseline) service operational environment of the aircraft. The USAF established the safe life of the airplane by dividing the number of successfully tested flight hours by a factor called the scatter factor. The intent of the factor was to account for article-to-article variation in materials and manufacturing quality. The USAF believed the process to be adequate to preclude structural failure in operational aircraft that was attributable to fatigue. The minimum scatter factor allowed[1] was 2. By the middle of the 1960s, the USAF had decided to use a scatter factor of 4 as evidenced by the requirements specified for the testing of the C-5A. Originally, the USAF assumed the life of the airplanes in service was the same as the safe life. They planned to remove an aircraft from service when it had accumulated the equivalent of the durability test damage divided by the scatter factor. This practice is normally unsatisfactory. In many cases, one may operate an aircraft beyond its safe life through inspections or modifications that are economical to perform.

When the USAF adopted damage tolerance, they made the decision to separate the process for assessing safety from the process for assessing aircraft durability. For the durability process, they introduced the concept of economic life. They initially defined the economic life as the time when the structure reached the point of widespread damage that was uneconomical to repair. Further, if the USAF did not repair this damage, it could cause functional problems affecting operational readiness.[2] They believed that this situation was generally observable in tests when there was a rapid increase in the number of damage locations or in repair costs as a function of cyclic test time. (In this definition, one should not interpret the term "widespread damage" as widespread fatigue damage. Widespread fatigue damage is the loss of fail-safety because of fatigue cracking.) The initial definition of economic life was not very

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useful because there was no satisfactory definition of "uneconomical." It did introduce, however, the idea of functional impairment. Functional impairment means that the state of cracking would be obvious and detrimental enough to warrant a repair. An example would be a crack that one would readily detect in a maintenance operation. Another example would be a crack from a fastener hole in a wing spar to the spar edge. Another example would be a crack that would cause a fuel leak. The notion of functional impairment is of major importance, because it removed confusion about the definition of failure. The USAF used the concept of functional impairment initially in the design of the F-16. They later refined the concept in the design of the C-17. For the C-17, as an example, the USAF required an analysis to show that an initial corner flaw in a fastener hole of 0.254 mm would not grow to functional impairment in two design lifetimes. The USAF data indicated that an initial flaw as large as 0.254 mm would occur only remotely—approximately once in approximately 100,000 fastener holes. The stresses in the C-17 derived from the damage-tolerance requirements typically were low enough to accommodate the durability requirement.

The USAF improved the concept of economic life in 1997 (Ref. 3) to provide guidance in the design of new aircraft. It provided an update on the tasks of the current ASIP initially established in 1975 (Ref. 4). Task IV of the ASIP is the development of the Force Structural Maintenance Plan (FSMP). This plan describes how, when, and where to perform inspections and modifications on the aircraft during its design life. The USAF modified the definition of the economic life to read, "The operational service period during which there is no significant departure from the cost burden associated with the FSMP for a newly manufactured aircraft. The contractor will determine this cost burden based on an evaluation of analytical and test data developed during the full-scale development process." The USAF believes that this is an improved definition. It portrays to the contractor that the cost of structural maintenance must not be a factor in the decision to retire an aircraft before it reaches the end of its design life. Experience has shown the results from the full-scale durability test are useful for the determination of economic life. The full-scale durability test of the F-4 successfully demonstrated this. In this case, at a certain time in the testing, the number of cracks found in the structure was increasing rapidly. It was readily apparent that the structure was at the end of its life. It was also evident that the structure would need extensive modification to proceed further. The original wing on the KC-135 aircraft also demonstrated a rapid increase in cracking at a certain time in full-scale testing. The USAF, however, did not recognize until later that, even before the rapid rise in visually detectable cracking incidents, there was a sufficient number of smaller cracks to jeopardize the fail-safety of the wing. Therefore, the wing had reached the point of widespread fatigue damage[5] before reaching its earlier estimated economic life. The loss of fail-safety in the KC-135 wing structure led to the replacement of the lower wing skin. Another example of the ease of determination of economic life was the T-38 wing. This wing was not repairable if the crack length was approximately 2.5 mm or longer. Thus, the time when an inspection of the wing revealed a crack of this size was its economic life. This type of indication of economic life is rare. Therefore, in most cases, even when the durability test results are available, it is not always clear how one should determine economic life.

A further complication is the effect of the environment on crack growth. The full-scale durability test does not duplicate the in-service environment, and, consequently, the test results may be optimistic about the structural life.

It is now clear that significant changes in the procedure for economic-life determination have to take place. In all of the definitions discussed herein, the USAF places structural analysts in a position of determining when the economic burden is unacceptable. The analysts would probably not be aware of all the important factors in the assessment of the burden. For example, they may not be aware of the importance of the weapon system to the USAF strategic or tactical plans. Further, they would likely not have access to the cost of a new weapon system. Therefore, they would not have access to the range of alternatives to the current weapon system. They could, however, make an estimate of the economic burden from inspections, replacements, and repair of the structure. It is the purpose of this paper to describe a method for accomplishing this. The strategy will be to develop a random-number set that represents the distribution of initial defects in the population of aircraft structures. There will be a random-number set for each aircraft location contributing to the maintenance cost. The analysts would sample these random-number sets according to the number of airplanes in the population. They can use the crack growth functions for each of these locations to determine the number of baseline usage hours of flying before an aircraft requires a maintenance action. The USAF, however, uses each aircraft in the population differently. Therefore, the analysts use a random-number set that reflects the ratio of the actual usage severity to the baseline usage severity to determine the actual hours to each individual maintenance action. For the total population, they sum the cost of these maintenance actions to determine the total cost of maintenance dependency on flight hours. Dividing this function by the number of airplanes in the population provides a cost function that shows the average cost dependency on flight hours. This paper demonstrates the process on the F-15E that McDonnell subjected to a durability test and a teardown inspection during its development.

F-15E Aircraft

McDonnell derived the F-15E from the F-15D—a two-place aircraft. The F-15 aircraft line started with the single-place F-15A powered by two Pratt & Whitney F-100 engines. The USAF developed the F-15A as an air-superiority fighter with design work initiated in January of 1970. Consequently, they developed it before they changed the structural requirements to include damage tolerance. Fortunately, however, McDonnell designed the aircraft structure with inherent conservatism. They used the Taper-Lok fastener system in the aircraft for fatigue performance enhancement, but they did not account for their beneficial effect when they established its stresses. Further, with guidance from the USAF Flight Dynamics Laboratory, they used a much more damage-tolerant titanium material than originally planned. These two features have enabled the F-15 to absorb significant increases in weight and usage severity without a major structural change. The USAF received its first operational F-15A aircraft in November of 1974. McDonnell designed the F-15E for a dual role of air-to-air and air-to-ground missions. However, the emphasis was on air-to-ground maneuvers and terrain following. McDonnell modified the F-15E structure somewhat from the F-15D. For the modified structure, the USAF required that the structure satisfy the currently existing damage-tolerance requirements. They required the contractor to establish the stresses to permit two lifetimes of slow crack growth before the initial "rogue flaw" reached critical size. Further, the USAF changed the limit load factor to 9.0 g from the original 7.33 g used for the F-15 design. The maximum takeoff mass of the F-15E is 36,818 kg.

F-15E Durability Test

The F-15E durability test was very successful. McDonnell conducted it for two simulated lifetimes. There were no major failures or rapid rise in the number of failures found that would indicate the aircraft was clearly reaching the end of its economic life, as found in the earlier testing of the F-4. They accomplished the durability test by using a flight-by-flight spectrum of loading that they believed was representative of the actual aircraft usage. Subsequently, they performed a teardown inspection that included a complete disassembly of all skin and substructure areas of the test section. During disassembly, McDonnell inspected the structure and fasteners for cracks or other anomalies. They assigned an inspection requirement to each part based on part criticality. They then examined the structure for cracking with penetrant and eddy-current inspections. They inserted the eddy-current probe in the fastener holes and automatically rotated it as it progressed through the hole. They used an eddy-current inspection to verify all crack indications. McDonnell used penetrant inspections after paint stripping and again after etching uncovered cracks in areas other than fastener holes. In those cases they did not etch the fastener holes and examine them with a magnifying glass, as described in the teardown inspection of the C-141 (Ref. 6). However, the process was able to reliably find cracks of the order of 750 μm. They excised all crack locations and broke them open. Finally, they examined each fracture face with an optical microscope and a scanning electron microscope to identify the fracture origin, the initiating failure mode, and any surface anomalies.

Interpretation of the Durability Test Results

In the case of the F-15E, there were insufficient cracks found in the teardown inspection to confidently determine a probability distribution function that the USAF believed would accurately represent the aircraft population. Considering past experience,[7] the USAF made the decision to use the Weibull distribution for the equivalent initial flaw size (EIFS) distribution. They used the following approach for generating the Weibull numbers that defined the distribution. From earlier testing, there were a considerable number of coupon tests using smooth specimens for the various materials used in the structure. They decided to use these coupon tests from the design development testing to determine the Weibull shape number for the distribution. The design development testing used constant-amplitude fatigue testing to determine the number of cycles to failure for particular stresses. McDonnell used these data to determine the EIFS from a fracture analysis that took into consideration the short-crack effect.[8] Figure 1 shows the short-crack correction to the long-crack data for both aluminum and titanium that McDonnell used for this study. The inclusion of the short-crack effect was key to the success of the program. The inclusion of the short-crack effect enabled McDonnell to pool the coupon test EIFSs derived from constant-amplitude tests with the random flight-by-flight loaded test aircraft EIFSs. This is a significant finding in that earlier use of the long-crack threshold for crack growth made it appear that the equivalent initial flaws in the structure were spectrum dependent.

McDonnell initially intended to use the teardown inspection cracks to modify the Weibull characteristic number derived from the cracks found in the coupon test program. However, they found in most cases that there was little if any correction needed from the teardown data. This indicates that the defects derived from the manufacturing process apparently did not exceed the intrinsic defects in the coupons. For many of the materials tested, they needed a three-parameter Weibull to fit the test data points adequately. Figure 2 shows the Weibull distribution from tests of Ti-6Al-4V.

McDonnell found that the Weibull shape numbers were approximately one, as other analysts have found in previous teardown inspection work.[9] The correlation coefficients for the Weibull distribution fit to the data were of the order of 0.97. The upper boundary of the EIFSs from the coupon test program was a crack length of the order of 200 μm. This is the upper boundary of intrinsic initial flaws found in the coupons. The upper boundary of the flaws found in the teardown inspection of the aircraft was also approximately 200 μm.

As indicated here, the USAF sampled the population of EIFSs for each airplane. They considered two alternatives for doing this. The first is to randomly sample the x projection of the inverse of the Weibull distribution function (i.e., the interval 0–1) and find the corresponding ordinate (i.e., the EIFS). The second is to select a random number, x, whose frequency of selection depends on the ordinate, y, of the Weibull density function corresponding to x. They found the two methods gave comparable results, and both of them provided an adequate representation of the Weibull density function. The second method is described in the paragraphs that follow.

McDonnell found that, for a small population of aircraft, the sample points might differ considerably from the parent population. To illustrate this point, they sampled a Weibull distribution with a shape number equal to 2.0 and a characteristic number equal to 0.05 mm 100, 1000, and 10,000 times. Figures 3, 4, and 5 show the result of this sampling. One observes that the process has converged reasonably well by the end of 10,000 samples. Further, one can see that for a small sample number, such as 100, the sample points do not represent the parent population very well. Therefore, the process should include multiples of the number of aircraft to ensure the resultant average cost has converged within acceptable limits.

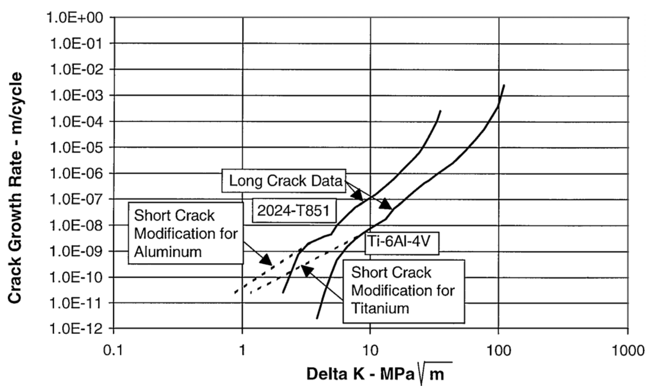

Fig. 1 Short-crack effect on crack growth rate.

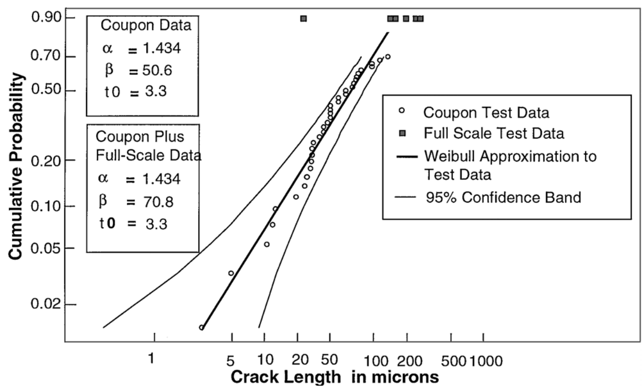

Fig. 2 Weibull distribution for Ti-6Al-4V initial flaws.

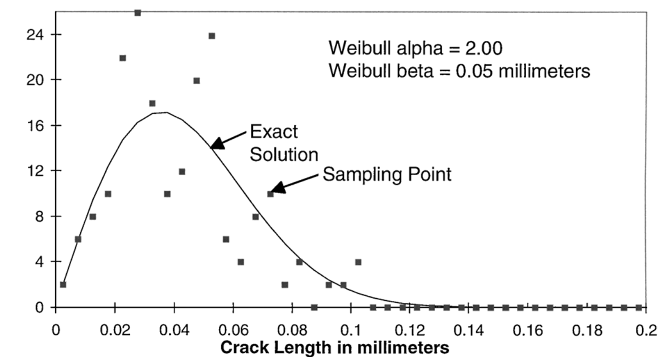

Fig. 3 Weibull sampling check with 100 samples.

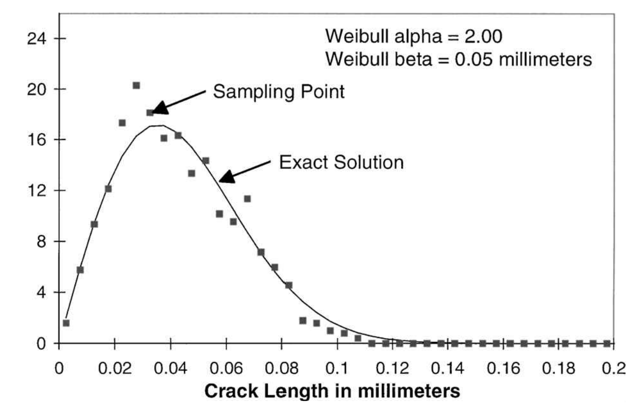

Fig. 4 Weibull sampling check with 1000 samples.

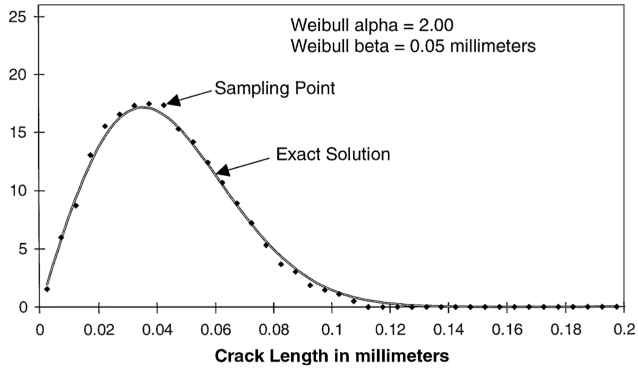

Fig. 5 Weibull sampling check with 10,000 samples.

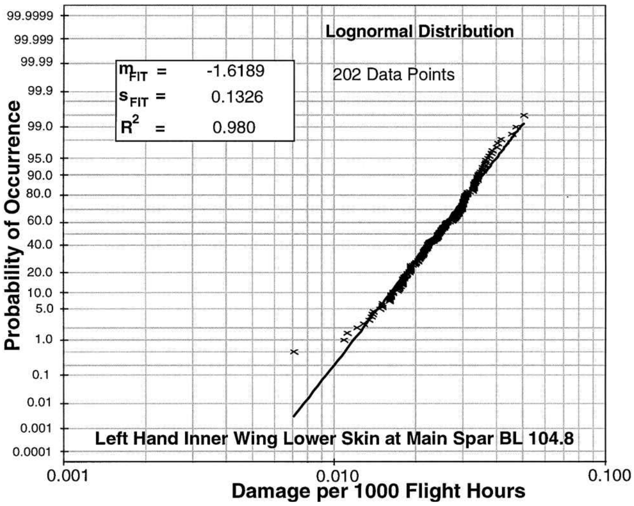

Fig. 6 F-15E damage per 1000 h.

Development of Usage Severity Ratios

The aircraft durability test from which McDonnell found the cracks represented the average usage of these airplanes. The airplanes, in service, fly with a severity that varies considerably from the average. The coefficient of variation of this usage severity is of the order of 0.3. Consequently, one cannot ignore the usage severity in the assessment of the maintenance cost dependency on flight hours. The aircraft severity by tail number is available to the USAF because they track the usage of each of the airplanes. McDonnell determined the damages for approximately 200 F-15E airplanes (Fig. 6). For this study, they used four locations in each airplane. These were the wing main spar at BL 104.8, longeron at FS 509, the horizontal tail spindle plate at BL 76, and the vertical tail root attach splice plate.

The USAF simulated the severity variations by a lognormal distribution. This is an extremely useful distribution for this purpose, because ideally it provides the ratios of usage severity for each sample randomly drawn from the parent population. As with the Weibull distribution discussed earlier in the text, the analyst must adequately sample the lognormal distribution to attain convergence (Fig. 7).

Sampling Procedure

Suppose that α is the Weibull shape number, β is the Weibull characteristic number, $t0$ is a positive number, and p_W is the Weibull density function such that if x is a positive number then

$$p_W(x) = \frac{\alpha}{\beta}\left(\frac{x - t0}{\beta}\right)^{\alpha - 1} \exp\left[-\left(\frac{x - t0}{\beta}\right)^{\alpha}\right] \qquad \text{for} \quad (x - t0) \geq 0$$

$$p_W(x) = 0 \qquad \text{for} \quad (x - t0) < 0$$

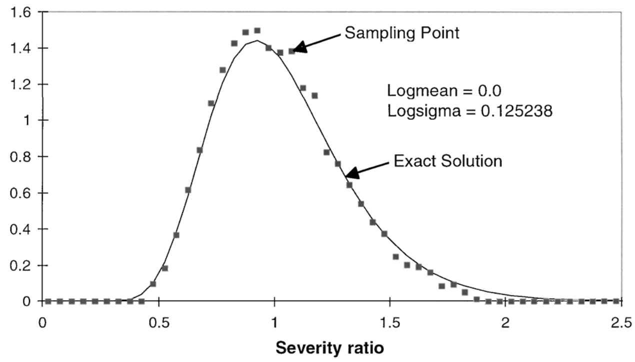

Fig. 7 Lognormal sampling check with 10,000 samples.

Further suppose that μ is a number, σ is a number, and p_{LN} is the lognormal density function with mean μ and standard deviation σ such that if x is a positive number, then

$$p_{\mathrm{LN}}(x) = \frac{1}{\sqrt{2\pi}\sigma x} \exp\left\{-\frac{1}{2}\left[\frac{\ln(x) - \mu}{\sigma}\right]^2\right\}$$

One may use approximately the same technique for random sampling of both the Weibull and the lognormal density functions. The only difference occurs when the Weibull shape number α is less than or equal to one. Suppose first that the Weibull shape number is greater than one and $t0$ is equal to zero. The first step is to determine the point $(x_{\max}, y_{\max})$, the maximum point of the density function. An iterative procedure is convenient to accomplish this. Note that $x_{\max}$ is the x projection of the maximum point of the density function whose ordinate is $y_{\max}$. The next step is to find the common points of the density function and horizontal lines, each of which have x projection the set of all numbers. For this purpose, suppose that m is an integer greater than one and that there are m equally spaced horizontal lines in the interval $[0, y_{\max}]$ such that no horizontal line contains the points (0, 0) or $(x_{\max}, y_{\max})$. The method for finding these common points through an iterative procedure is similar to that used to find the maximum ordinate of the density function. Suppose that the labels the of x projection of the common points on the left side of the maximum of the density function are $x_{L1}, x_{L2}, \ldots, x_{Lm}$, where x_{Li} is less than x_{Li+1}. Also, suppose that the labels of the x projection of the common points on the right side of the maximum of the density function are $x_{R1}, x_{R2}, \ldots, x_{Rm}$, where x_{Ri} is greater than x_{Ri+1}. Suppose that X is the number set such that x_i is a member of X only if i is in $[1, m]$ and x_i is the length of the interval $[x_{Li}, x_{Ri}]$. These intervals, arranged in juxtaposition in the order of $1, 2, \ldots, m$, define an interval whose length x_s is the sum of the lengths of intervals x_i, $i = 1, m$. Now suppose that there is a number set R such that R only contains the set of all numbers in the interval [0, 1]. Further suppose that z is a number such that if r is a random selection from R then z is $(r \cdot x_s) + x_{L1}$. It follows that z is in the interval $[x_{L1}, x_{L1} + x_s]$.

The next step is to determine which of the m intervals derived from the horizontal lines contains the selected random number. The orderly arrangement of the m intervals permits this is be done easily. It remains to determine the correct position of the sample in the interval $[x_{L1}, x_{R1}]$. To accomplish this, suppose that n is an integer greater than one and I is a set of intervals $I_1, I_2, \ldots, I_n$ of equal length such that if x is a number contained in the interval I_i then x is in $[x_{L1}, x_{R1}]$. The final step is to determine in which interval the selected random number falls. Knowing which of the m intervals derived from the horizontal lines contains the random number and the left side of that interval permits this to be done easily. The only retained knowledge of the selected random number is which of the n intervals in I contains the number. Figure 8 shows the placement of the horizontal lines and sampling intervals for the process. The USAF set the number m equal to 19 and the number n equal to 50 for the Weibull and lognormal distributions used for the F-15E.

Table 1 Weibull distribution numbers

Material	α	β (μm)	$t0$ (μm)
Ti-6Al-4V	1.434	71	3
7075-T7352	0.763	4	6
7175-T7452	1.056	33	18
2024-T851	0.659	27	8
Ti-6Al-6V-2Sn	0.730	2	1
7075-T76	1.519	48	0

Table 2 Lognormal distribution numbers

Location	Logmean	Logsigma
1. Inner wing lower torque box skin at the rear spar	0.0	0.3053
2. Outer wing ft spar hole #3 wire bundle hole at X_W 169	0.0	0.3053
3. Outer wing rear spar at hole #4 at X_W 169	0.0	0.3053
4. Inter spar fuel drain hole IS #15 at X_W 67.4	0.0	0.3053
5. Aft fus FS 791 canted frame flange	0.0	0.1662
6. Aft fus stringer #10 at FS 672	0.0	0.1662
7. Ctr fus FS 415 blkhd	0.0	0.3808
8. Ctr fus upper inbd longeron	0.0	0.3809
9. Ctr fus FS 626.9 blkhd web at lwr cap intersection	0.0	0.3053
10. Fwd fus stringer #1 at FS 452	0.0	0.3809

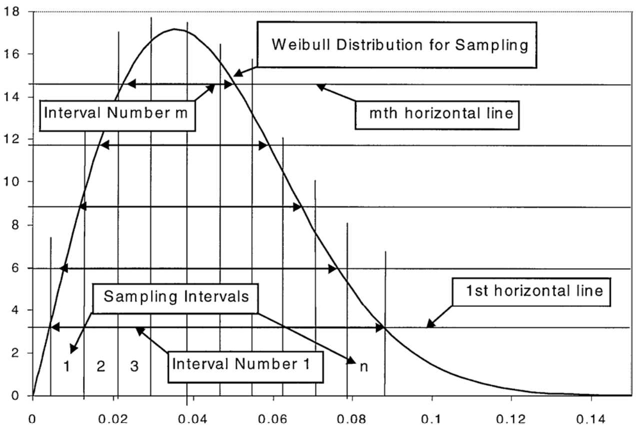

Fig. 8 Weibull density function sampling.

Table 3 Cost per maintenance event

Location	Cost, $
1. Inner wing lower torque box skin at the rear spar	91,200.00
2. Outer wing ft spar hole #3 wire bundle hole at X_W 169	12,100.00
3. Outer wing rear spar at hole #4 at X_W 169	12,100.00
4. Inter spar fuel drain hole IS #15 at X_W 67.4	61,700.00
5. Aft fus FS 791 canted frame flange	81,727.00
6. Aft fus stringer #10 at FS 672	73,823.00
7. Ctr fus FS 415 blkhd	320,000.00
8. Ctr fus upper inbd longeron	229,000.00
9. Ctr fus FS 626.9 blkhd web at lwr cap intersection	285,000.00
10. Fwd fus stringer #1 at FS 452	145,000.00

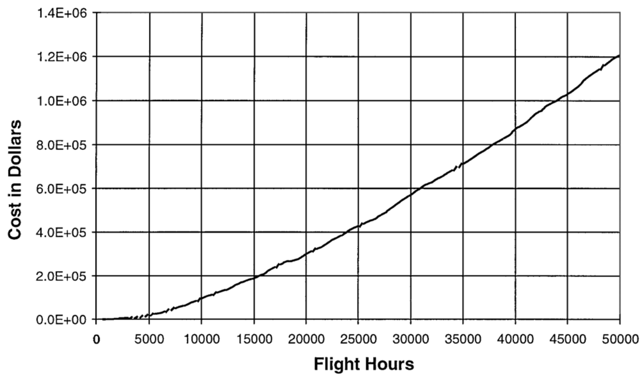

Fig. 9 Maintenance cost function for the F-15E.

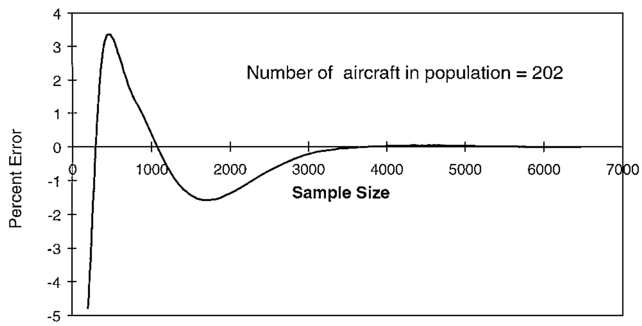

Fig. 10 Sampling error at 15,000 flight hours.

In the case in which the Weibull shape number α is one and $t0$ is equal to zero, $x_{\max}$ is equal to zero and $y_{\max}$ is equal to α divided by β. Therefore, for the m horizontal lines just defined, one needs to iterate only for the right side of the density function. The remainder of the process is the same. In the event that α is less than one, the number $y_{\max}$ is unbounded. To eliminate the problem one selects a positive number D such the point (D, y_D) is a point of the density function. One modifies the process based on the assumption that the density function's x projection includes no number smaller than D. The selection of D requires some judgment in that it must provide for an adequate distribution of horizontal lines and still not significantly change the density function.

Table 1 shows the Weibull distribution numbers for the various materials used in the analysis. Table 2 shows the lognormal distribution numbers for each of the ten locations that McDonnell found to contribute to the maintenance cost.

Determination of the Maintenance Cost Function

The USAF analyzed each airplane in the population and each area of the airplane that contributes to the maintenance cost. They sampled the Weibull distribution of the initial flaws and the lognormal distribution for the usage severity. They grew the cracks derived from the sampling to the point of a repair or modification action. McDonnell derived these crack growth functions from damage tolerance requirements for the F-15E. McDonnell found ten locations on the aircraft contributing to the maintenance costs. Table 3 shows the cost per maintenance event in these ten areas. McDonnell took into consideration the number of times these locations were in the structure. Warner Robins Air Logistics Center provided the costs of these maintenance actions. They then accumulated the costs such that they could determine the cost function. Figure 9 shows the result of this computation. This was a result that USAF management could combine with the costs from engines, mechanical systems, and avionics to determine if the aircraft had reached its economic life. The USAF usually cannot determine the economic life from the maintenance cost function from the airframe alone. It may be the deciding factor, however, in the retirement of an aircraft. This is likely not the case for the F-15.

As already discussed, the sample size used may affect the maintenance cost function. The Air Force investigated this for the F-15E, with a population of 202 aircraft. They increased the sample size arbitrarily until the results converged. They found that even for the relatively low sample size of 202 aircraft, the error was less than 5%. This magnitude of error is normally acceptable. The process converged within 1% with approximately 2500 samples. Figure 10 shows the results of this investigation.

Conclusions

The procedure outlined here provides the USAF with the cost function needed to determine when a population of airplanes would reach the end of their useful life. The USAF could make an initial estimate of the economic life by using this procedure after the completion of the teardown inspection of the full-scale durability test article. The USAF could estimate the severity distribution from data derived from previous weapon systems. For the case described herein, McDonnell tested the aircraft to the average usage spectrum. The USAF, in the future, plans to test their aircraft to a usage spectrum more severe than the average. They plan that only 10% of the operational aircraft population will exceed the test severity. This does not present a problem since the contractor can determine the initial flaw distribution by using the more severe spectrum. Accounting for the short-crack effect permits one to establish the flaw distribution independently of the spectrum. The USAF should plan to repeat the process during the aircraft's life when more data become available from actual maintenance operations. The procedure would also be suitable for aircraft that use the fail-safe approach to ensure structural integrity. For fail-safe aircraft the USAF could add to the cost function the cost associated with occurrences of widespread fatigue damage, if any. The procedure as outlined does not account for environmental effects except as they influence the crack-propagation rates. The USAF could add the cost of corrosion inspections to the cost function. However, it is not easy to incorporate the cost for corrosion repair because of the scarcity of this type of USAF aircraft maintenance cost data. Trends derived from this information would be helpful in projecting future requirements for maintenance funding. The life management program for the corrosion-prone KC-135, called Coral Reach, introduced three metrics that appear to be quite useful for judging the retirement time for this aircraft. They are safety, cost per flying hour, and availability. They are effective only if the USAF develops the database to quantify them. There is a critical need for the USAF to determine the costs of structural maintenance so that they can better predict the lives of their aircraft.

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