Chapter

Advantages of Condition-Based Maintenance over Scheduled Maintenance Using Structural Health Monitoring System

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Abstract

This chapter quantifies the advantages of condition-based maintenance on the safety and lifetime cost of an airplane fuselage. The lifecycle of an airplane is modeled as blocks of crack propagation due to pressurization interspersed with inspection and maintenance. The Paris-Erdogan model with uncertain parameters is used to model fatigue crack growth. The fuselage skin is modeled as a hollow cylinder, and an average thickness is calculated to achieve a probability of failure in the order of 1 in 10 million with scheduled maintenance. Condition-based maintenance is found to improve the safety of an airplane over scheduled maintenance and will also lead to savings in lifecycle cost. The main factor of the savings stems from the reduced net revenue lost due to shortened downtime for maintenance. There are also other factors such as work saved on inspection and removing/installing surrounding structures for manual inspection. In addition to cost savings, some potential advantages of condition-based maintenance are discussed such as avoiding damage caused by removing/installing surrounding structures, more predictable maintenance, and improving the safety issues of same aircraft model by posting the frequently occurred damages into Airworthiness Directives, Service Bulletins, or Service Letters.

Keywords: condition-based maintenance, structural health monitoring, damage tolerance, lifecycle cost

1. Introduction

Traditionally, aircraft structures have been designed using the damage tolerance concept (Hoffman, [1], Simpson et al. [2]), which refers to the ability of structure to sustain anticipated loads in the presence of certain damage until such damage is detected through inspections or malfunctions and repaired [3]. More specifically, as cracks on fuselage skin are the damage this chapter is focusing on, it means that structure is designed to withstand small cracks and large cracks are repaired through scheduled inspection and maintenance. In damage tolerance design, an airframe is regularly inspected so that potential damages are early identified and repaired. As such, scheduled maintenance is the primary tool in aircraft maintenance philosophy where inspections and repair works are performed at fixed scheduled intervals in order to maintain a desired level of safety.

Historically, the risk due to fatigue cracks in fuselage has been identified early in civil aviation due to the three accidents of Comet aircraft (BOAC Flight 783 (1953), BOAC Flight 781 (1954), South African Airways Flight 201 (1954)). In addition, the accident of Aloha Airlines Flight 243 (1988) revealed that multiple-site fatigue cracking caused the failure of the lap joint. Fatigue cracks also caused accidents in other parts of the aircraft, such as the wing spar failure in Northwest Airlines Flight 421 (1948). Since then, inspection and scheduled maintenance have been conducted to detect fatigue cracks and repair them before they cause structural failure. However, deficiency and mishap during the inspection and maintenance often caused accidents. For example, the accident of Aloha Airlines was partly caused by the fact that the inspection was conducted at night. Japan Airlines Flight 123 (1985) crashed due to incorrect splice plate installation during the corrective maintenance, which reduces the part's resistance to fatigue cracking to about 70%.

Scheduled maintenance can be categorized into transit check, 24 h of check, and A/B/C/D checks with increasing intensity and interval. For a Boeing 737-300/400/500, the typical C check is carried out at about 2800 flight cycles (4000 flight hours with an average flight length of 1.4 h) [4]. This inspection schedule is chosen such that the probability of an undetected crack growing beyond the critical size before the next scheduled maintenance is less than 1 in 10 million [5].

In CBM, a damage parameter is continuously monitored by a structural health monitoring (SHM) system, whereby maintenance is requested when the value of damage parameter exceeds a certain threshold [6]. Such an SHM system uses onboard sensors and actuators, enabling the damage assessment to be performed as frequently as needed.

This chapter presents an estimate of cost savings using condition-based maintenance over scheduled maintenance. The effect on cost and safety of condition-based maintenance using SHM system over scheduled maintenance is demonstrated for fuselage skin subject to fatigue crack growth. In scheduled maintenance, maintenance is scheduled at predetermined intervals. Since these inspection intervals are relatively large, all detectable cracks must be repaired. In condition-based maintenance, however, crack assessment can be performed as frequently as needed; repair work is then requested only when the size of detected crack exceeds a certain threshold that can threaten the safety of fuselage skin. This leads to condition-based maintenance using SHM to be an effective approach to reduce lifecycle cost. Boller [7] observed that using SHM for condition-based maintenance would lead to lower downtime and inspection cost. Sandborn and Wilkinson [8] and Scanff et al. [9] studied the cost estimation of electronic and helicopter systems, respectively, using health monitoring systems. In order to facilitate a progressive transition from scheduled maintenance to condition-based maintenance, a hybrid approach is also considered where scheduled maintenance is used for critical structures and condition-based maintenance for noncritical structures.

Several simplifications are made in this chapter in order to make the cost calculation simple:

Firstly, although three types of crack detection approaches have been used in scheduled maintenance, general visual inspection (GVI) is considered as the only detection approach in this chapter because it is the most commonly used inspection method. The three detection approaches are:

- General visual inspection (GVI)
- Detailed visual inspection (DVI)
- Nondestructive test (NDT) with increasing resolution

NDT can be subcategorized into eddy current, ultrasonic, X-ray, magnetic particle, and penetrant [10]. For the most part of fuselage skins, GVI is used. As areas that require DVI and NDT are extremely small compared to those that require GVI, it is assumed that GVI is the only detection approach herein.

Secondly, repair of fuselage skin is considered to be the only maintenance in this chapter. In scheduled maintenance, the maintenance of fuselage skin includes repair and replacement. However, replacement of fuselage skin is only performed when unexpected damage in fuselage skin occurs because of incidents, such as the aircraft bumping into a ground vehicle when taxiing or when widespread fatigue damage occurs on aged aircraft. The latter refers to the simultaneous presence of cracks at multiple locations that are of sufficient size and density resulting in the structure not being able to meet any longer required damage tolerance limits; thus, it will not maintain required residual strength after partial structural failure. Under normal circumstances, for a single crack on fuselage skin, the probability of replacing fuselage skin is extremely low based on the first author's experience and can be negligible. Therefore, this chapter discusses only the repair of fuselage skin.

Lastly, the loading condition for every aircraft structural component is complicated, and variable amplitude loadings and repeated hard landing, for example, should be considered. In this study, however, the discussion is focused on crack propagation on fuselage skin. The most dominant loadings are repeated pressurizations during takeoff and landing. Therefore, the pressurization difference is assumed to be the only loading condition herein.

The structure of the chapter is as follows:

In Section 2, the literature on SHM sensor technologies are reviewed. In Section 3, the processes of damage detection and repair are explained. Section 4 quantifies the parameters for scheduled and condition-based maintenance to maintain a specific level of safety. Section 5 compares the cost savings of condition-based maintenance over scheduled maintenance. Section 6 discusses some potential advantages of condition-based maintenance, followed by conclusions in Section 7.

2. Literature review on structural health monitoring technologies

In CBM, the inspection is performed using sensors installed on the aircraft structure, called a SHM system. Therefore, it is important to review the current sensor technologies to evaluate their performance in detecting cracks. In general, the sensors used in SHM systems are either active or passive sensors. Passive sensors detect signals generated by damage due to the evolution of the damage, which does not require an external excitation. Acoustic emission belongs to this category [11]. If damage is detected during flight, this can be a useful method. As mentioned earlier, however, since the inspection is performed on the ground, it would be difficult to use passive sensors to detect damage. Therefore, passive sensors will not be discussed in this chapter.

Active sensors detect damage by sending a signal to the damage. Since the purpose is to use them for SHM, the review in this section focuses on the smallest size of detectable damage, the detection range, the weight of SHM systems, and the possibility of detecting closed cracks. It would be desirable that the SHM systems can detect at least the same damage size with the NDT. The detection range will determine the total number of sensors required to inspect the entire fuselage panels. In order to reduce the payload loss, it is important to reduce the weight of the SHM system. Since the inspection is performed on the ground, it is required to detect closed cracks.

The most widely used active sensor is the piezoelectric wafer active sensor (PWAS), which uses ultrasonic lamb waves. As an actuator, it converts the electric signal to mechanical motion to generate a longitudinal or transverse wave, which propagates on the panel and is reflected at a crack. As a sensor, it receives a wave reflected from a crack and converts it to electric signals. The location and size of damage are estimated by measuring the time, amplitude, or frequency of the reflected wave.

In general, two methods are used to detect damage [6]. In the pulse-echo method, one PWAS sends waves and receives waves reflected at a crack. In the pitch-catch method, one PWAS sends waves, and the other PWAS receives the waves. In addition, several PWAS, called a phase array, are used simultaneously to improve detection capability [12]. Although the abovementioned two methods require undamaged (pristine) state, the time reversal method [13] does not require it. Since the mechanism of detecting damage using PWAS is similar to conventional NDT ultrasonics, the detectable damage size is also similar to NDT. The most preferable feature of PWAS is its capability of detecting a remote damage from the sensor. Giurgiutiu [14] showed a lamb wave tuning method to detect a remote damage effectively. It has been shown that PWAS can be used for both metallic and composite panels [15]. In order to reduce the excessive number of wires to connect sensors, SMART layer [16] is developed by printing circuits of 30 sensors into a thin dielectric film.

Fiber Bragg grating (FBG) uses a series of parallel lines of optical fiber with different refractive indices [17]. When a local strain is produced due to the presence of a crack, it will change the spacing between gratings, which shifts the wavelength of the reflected wave. FBG sensors detect damage by measuring the shift of reflected wavelength. It is small and lightweight. It was shown that a single optical fiber could incorporate up to 2000 FBG sensors [18]. The literature also showed that it could detect barely visible impact damage in a composite panel [19]. However, FBG sensors have a very short detection range because the local strain diminishes quickly as the distance increases. It would perform better for hotspot damage monitoring, where the damage location is already known. Since cracks in fuselage are opened during flight and closed on the ground, FBG is not appropriate for on-ground SHM. Lastly, since FBG measures the change in strains, it requires strains at the undamaged (pristine) state. If there is pre-existing damage, it can only measure the change from the previous damage.

Comparative vacuum monitoring (CVM) sensors are composed of alternating vacuum and atmospheric pressure galleries and detect cracks using pressure leakage between galleries. The testbed in Sandia National Laboratory showed that CVM could detect cracks in the size of 0.02 in [20]. Airbus [21] and Delta Airlines [22] also tested the feasibility of CVM on SHM. CVM sensors are lightweight made of polymer, and the gallery can be as small as 10 μm [23]. Even if CVM sensors do not require undamaged (pristine) state, it can only detect damage underneath the sensor. Therefore, CVM is appropriate for hotspot monitoring. For fuselage damage monitoring, it would require a sensor layout with a very high density.

There are other kinds of sensors, such as carbon nanotube sensors [24], printed sensors [25], and microelectromechanical systems sensors [26]. These sensors are, however, still in the research or development stage and take more time to be commercially available.

As a summary, among different sensor technologies, it turned out that PWAS is the most appropriate for an SHM system for airplane fuselage monitoring as it can detect cracks that are relatively small and far away from the sensors.

3. Maintenance process for fuselage structures

3.1 Corrective maintenance procedure

Repeated pressurization during takeoff and landing of an airplane can cause existing cracks on a fuselage skin to grow, for example, Aloha Airlines Flight 243. The rate of crack growth is controlled by, among other factors:

- The size of initial cracks due to manufacturing or previous maintenance
- The pressure differential between the cabin and the outside atmosphere
- The thickness of the fuselage skin

If left unattended, the cracks may grow to cause fatigue failure of the fuselage skin. In damage tolerance design, the less frequent the inspection, the lower the damage size threshold for repairing cracks in order to maintain a desired level of safety. The action of repairing cracks on fuselage skin to maintain a desired level of safety until the next scheduled maintenance is termed corrective maintenance. This section explains the modeling of the corrective maintenance procedure undertaken to prevent fatigue failure due to excessive crack growth.

The size of cracks in fuselage structures in a fleet of airplanes is modeled as a random variable characterized by a probability distribution that depends on manufacturing and the loading history of the airplane. The corrective maintenance procedure changes this distribution by repairing large-sized cracks as illustrated in **Figure 1**. **Figure 1** is presented as a probability density function (PDF) versus crack length. The solid curve represents the crack size distribution of an airplane entering the maintenance hangar. Different cracks grow at different rates because of random distribution of the Paris-Erdogan model parameters. The maintenance process is designed to repair fuselage skin with cracks larger than a repair threshold. Since crack detection is not perfect due to inspector's capability [27], maintenance only partially truncates the upper tail of the distribution, as represented by the dashed curve in **Figure 1**. It is noted that while there is uncertainty in damage detection, it is assumed that the size of the detected damage is known without any error/noise.

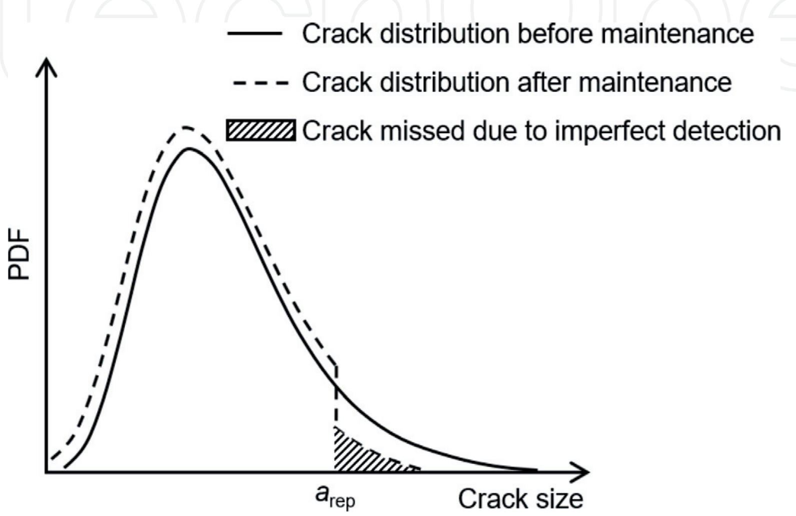

Figure 1.
The effect of inspection and repair process on crack size distribution.

The shaded area represents the fraction of cracks missed during maintenance because of detection imperfection. The cracks that are missed during maintenance and happen to grow beyond the critical crack size before the next maintenance affects the safety of the aircraft.

3.2 Scheduled maintenance

The flowchart in **Figure 2** depicts the scheduled manual maintenance, in which maintenance is programmed at specific predetermined intervals (every N_{man} flight cycles) and corrective action is taken to ensure the airworthiness of the airplane until the next scheduled maintenance.

As all detected cracks on fuselage skins are repaired, the desired level of safety is determined by detection resolution/capability of GVI, a_{gvi}. It is expected that trained inspectors are able to detect cracks larger than 0.5 in (12.7 mm) in GVI. This is also the threshold for repair in scheduled maintenance.

Three parameters affect the lifecycle cost and safety of an aircraft undergoing scheduled maintenance: the maintenance interval, N_{man}; the threshold for repair (detection capability), a_{gvi}; and the thickness of the fuselage skin, t. To achieve a certain desired level of safety, N_{man} and a_{gvi} are correlated with each other. These three parameters together determine the number of maintenance trips and the number of cracks needed to be repaired on fuselage skins.

3.3 Condition-based maintenance

The condition-based maintenance process tracks crack growth continuously and requests maintenance when the crack threatens safety. In this chapter, the condition-based maintenance is considered to be performed using SHM technique. This technique employs onboard sensors and actuators, which are embedded in the structure, to monitor existing crack condition. In doing so, they detect cracks in metallic structures using guided waves transmitted from one location and received

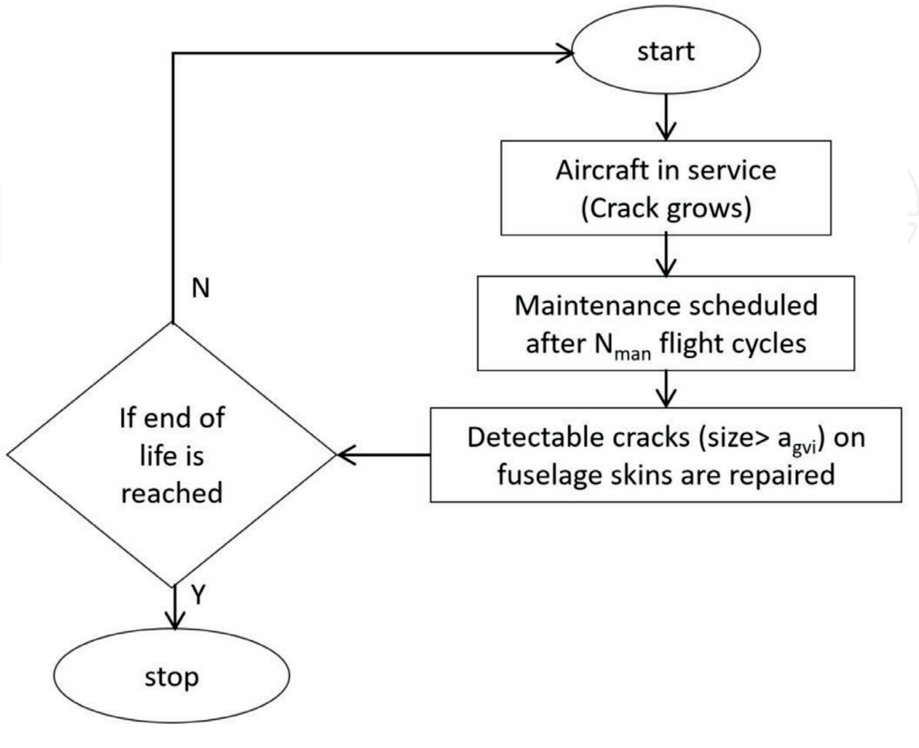

Figure 2.
Flowchart of the scheduled maintenance.

at a different one. The analysis of the change in a guided wave's shape, phase, and amplitude yields indications about crack presence and extension. The probability of detection of the SHM method is comparable with that of conventional ultrasonic and eddy current methods [28]. Crack size and location can be displayed on ground equipment when connecting to onboard sensors and actuators after landing. On-ground equipment can reduce the flying weight and thus may lower the lifecycle fuel cost.

The abovementioned process is called herein maintenance assessment. SHM-based maintenance assessment can be performed as frequently as every flight. However, as the crack increases by only a small amount in each flight cycle, it is unnecessary to perform this assessment after every flight. Also, maintenance assessment is not completely cost-free but requires a small amount of time and personnel. Typically, this assessment frequency (N_{shm}) is assumed to coincide with the A check of scheduled maintenance, which is about 180 flight cycles (250 flight hours with average flight length of 1.4 h [4]).

Figure 3 delineates the condition-based maintenance process. During the assessment, maintenance is requested if the crack size on a fuselage skin exceeds a specified threshold (a_{th}). This threshold is performed, so as to repair all detected cracks on fuselage skins with threatening crack sizes. Additionally, the threshold for threatening crack size $(a_{rep-shm})$ is set substantially lower than the threshold for requesting maintenance (a_{th}) to prevent too-frequent maintenance trips for that airplane.

Condition-based maintenance is controlled by the following parameters:

- The thickness of fuselage skin (t), which affects the crack growth rate
- The thickness (t), along with the frequency of assessment (N_{shm}), and the threshold for requesting maintenance (a_{th}) affect the safety of the airplane
- The threshold for repair $(a_{rep-shm})$ determines the number of cracks needed to be repaired on fuselage skin. It is also set to prevent frequent maintenance trips

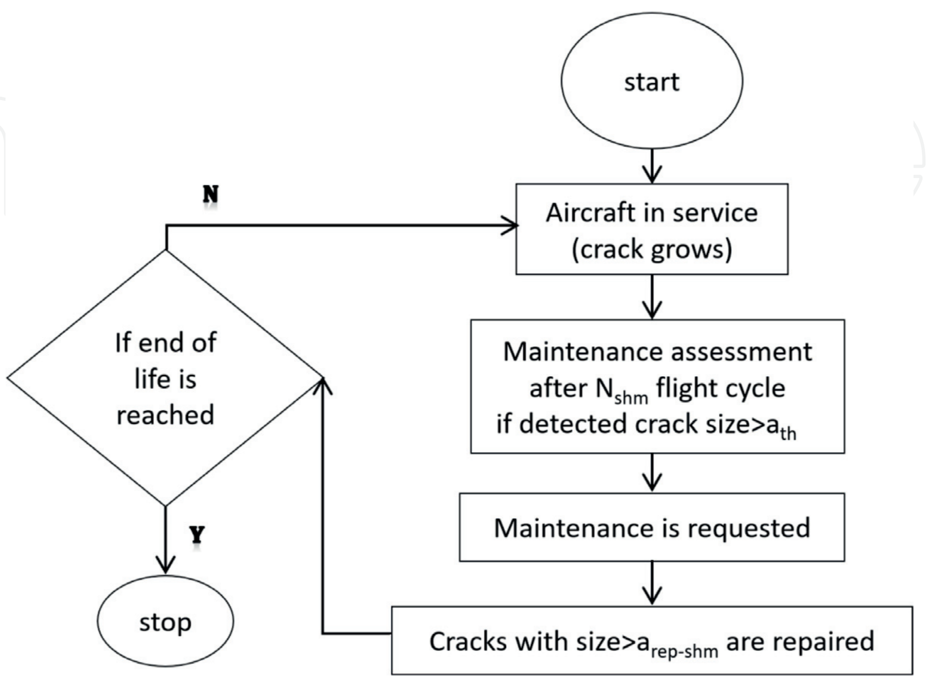

Figure 3.
Flowchart of the condition-based maintenance.

4. Parameters assumed for scheduled and condition-based maintenance

Cracks that are missed or intentionally left unattended during maintenance and grow to critical size before the next maintenance interval affect the safety of the aircraft structure. In the case of scheduled maintenance, the thickness of the fuselage skin (t), the interval of scheduled maintenance (N_{man}), and the threshold for repair (a_{gvi}) affect the aircraft's safety, which is influenced by the thickness of the fuselage skin (t), the frequency of maintenance assessment (N_{shm}), and the threshold for requesting maintenance (a_{th}).

This section deals with quantifying the range of parameters for scheduled and condition-based maintenance. As such, each damage instance is modeled as a through-the-thickness center crack in an infinite plate subject to Mode-I fatigue loading, as shown in Appendix A. The uncertainty in the loading condition and material parameters are summarized in **Table 4**. A crack grows due to pressure differential between the cabin and atmosphere, which is modeled by the Paris-Erdogan model, as shown in Appendix A. From fracture mechanics, the critical crack size (Eq. (3)) to cause failure of a fuselage skin depends on the pressure load and, hence, may also be modeled as a probability distribution. This chapter considers a fuselage skin to be failed if the crack grows undetected beyond the 10^{-7} percentile of critical crack size distribution.

In the scheduled maintenance of a B737-300/400/500, the C check is carried out at about every 2800 flight cycles $(N_{man} = 2,800)$ [4] for an airplane life of 50,000 flights. The threshold for repair is equal to the detection capability of GVI, $a_{gvi} = 0.5$ in (12.7 mm). The fraction of cracks which cause failure of fuselage skins due to excessive crack propagation until the end of life is computed by Monte Carlo

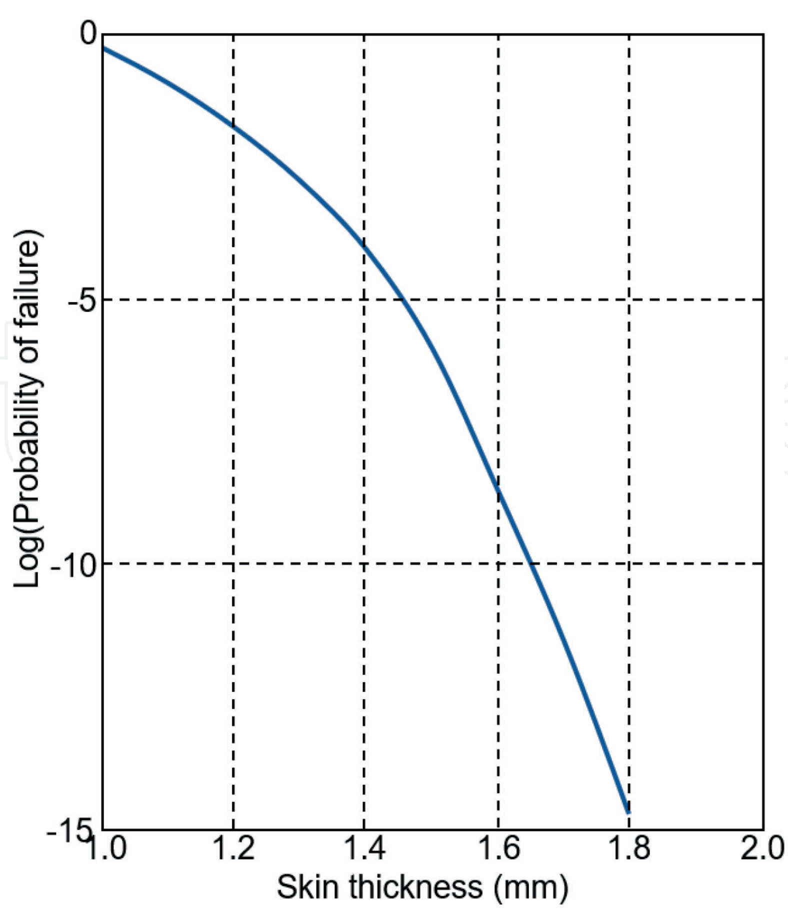

Figure 4.
Variation of lifetime (50,000 flight cycles) probability of failure as a function of fuselage skin thickness for scheduled maintenance at every 2800 flight cycles.

simulations. A fleet of 20,000 airplanes with 500 initial cracks per airplane due to manufacturing or previous maintenance are considered. These cracks are distributed on fuselage skins. The initial crack size and crack growth parameters (m, a_{th}) are randomly sampled for each crack. Pressure is also assumed to vary in each flight. The fraction of cracks that cause fuselage skins to fail is computed for different values of skin thickness, and the variation is plotted in **Figure 4**. Based on **Figure 4**, a fuselage skin with a minimum thickness of 0.06 in (1.53 mm) is required to achieve the target probability of failure of 10^{-7}. Considering that 0.063 in (1.6 mm) is the most common thickness of a typical fuselage skin, this calculation provides a reasonable estimate.

In condition-based maintenance, the threshold for scheduling aircraft to maintenance must be chosen in such a way so as to satisfy the reliability constraint until the next maintenance assessment(N_{shm}). The latter has been chosen as 180 flight cycles, which is equivalent to the current A check interval. If say the threshold for requesting maintenance (a_{th}) is fixed at 1.57 in (40 mm), the reliability for the given value of a_{th} and N_{shm} can be computed using a direct integration procedure, detailed in Appendix C, and is proven to satisfy the desired level of safety.

5. Cost comparison between two maintenance processes

In this chapter, the lifecycle cost of an airplane is considered to be the sum of manufacturing cost, fuel cost incurred during lifecycle, and maintenance cost. Other costs that remain constant for two different approaches are not considered. Cost comparison of two maintenance approaches is discussed in two aspects: cost increase and cost decrease. **Table 1** summarizes the parameters that are used for cost calculation for the two maintenance processes based on Boeing 737-300 Structural Repair Manual and estimated the cost in the maintenance field.

Based on the Structure Repair Manual of a Boeing 737-300, the fuselage skin in the pressurized area is not a regular cylinder. However, it was assumed to be a cylinder to simplify calculation, by using the average diameter $D = 148\text{in}$. In addition, the length of the cylinder can be calculated as $L = 977\text{in}$. As already stated, the thickness of the fuselage skin varies from station to station; however, the most common thickness of $t = 0.063\text{in}$ is used herein. In addition, the density of fuselage skin, which is made of aluminum alloy 2024-T3, is about $\rho = 0.1\text{lb}/\text{in}^3$. Therefore, the total weight of fuselage skin in the pressurized area is $W = \pi DLt\rho = 2957\text{lb}$.

5.1 Cost increased

(1) Manufacturing cost.

Manufacturing cost with SHM system: \$600/lb.
Manufacturing cost without SHM system: \$500/lb.

Weight of fuselage skins	2957 lb. [10]
Interval of C check	2800 flight cycles
Life cycles	50,000 flight cycles
Net revenue lost due to downtime	\$27,000/airplane/day
Labor cost in hangar	\$60/h

Table 1.
Parameters for maintenance cost calculation.

$$\text{Cost increased} : (600 - 500) \times 2,957 = 3 \times 10^5 (\$)$$

(2) Cost on replacing SHM equipment.

A finite life of 12,000 flight cycles for SHM equipment is assumed so that the system will need to be replaced four times during 50,000 flight cycles. The lifetime cost for replacing the SHM system after manufacturing is as follows:

$$\text{Cost increased} : 3 \times 10^5 \times 4 = 1.2 \times 10^6 (\$)$$

(3) Fuel cost.

Weight penalty: lifetime fuel consumption cost per aircraft weight. Kaufmann et al. [29] used $1000 per pound as the lifetime fuel cost for 1 pound of gross weight of aircraft. About 5% extra weight is considered for fuselage skin with SHM equipment. Therefore, the cost increase due to SHM equipment weight increased is as follows:

$$\text{Cost increased} : 2957 \times 5\% \times 1000 = 1.5 \times 10^5 (\$)$$

5.2 Cost decreased

As damage assessment intervals in condition-based maintenance are much smaller than that of the scheduled maintenance, the threshold a_{th} for requesting condition-based maintenance to be much larger than a_{gvi} in scheduled maintenance. This high damage tolerance reduces the number of maintenance trips. In addition, because the threshold for repair $a_{rep-shm}$ is larger than a_{gvi}, the number of cracks that are repaired is reduced in condition-based maintenance. It is assumed that these are two factors that would cause savings in aircraft lifecycle maintenance costs.

Monte Carlo simulation (MCS) is performed to compute the number of maintenance trips and the number of cracks repaired on fuselage skins for scheduled and condition-based maintenance. It is assumed that 500 initial cracks on a B733 are distributed on fuselage skins, showing a typical thickness of 0.063 in (1.6 mm).

The damage detection process is governed by the Palmberg expression (Appendix B) with different parameters for scheduled and condition-based maintenance. The parameters computed are listed in **Table 2**. Values in parentheses are MCS standard deviations based on 20,000 airplanes. It is considered that SHM equipment is replaced every 12,000 flight cycles.

It is noted that for the same fuselage skin thickness, condition-based maintenance leads to better reliability and lower number of maintenance trips and cracks repaired. The reason is that scheduled maintenance repairs all the cracks that might grow to threaten safety until the next maintenance, while the condition-based maintenance repairs only those that actually grow to threaten safety.

Based on the results computed above, the cost saved can be calculated as follows:

(1) Net revenue saved due to shortened downtime.

Types of maintenance	Probability of failure	Avg. no. of maintenance trips per airplane	Avg. no. of cracks repaired/airplane
Scheduled	1E-8	18	10 (0.6)
Condition-based	1E-13	7.6 (0.3)	5.8 (0.2)

Table 2.
Comparison between scheduled and condition-based maintenance.

The downtime for C checks of B737 CL varies from several days to 2 months as the age of the aircraft increases. This chapter regards 30 days as a typical downtime for a C check. Usually, the inspection procedure takes up about 1/3–1/4 of the whole downtime in scheduled maintenance. In condition-based maintenance, however, it is assumed that the assessment process can be completed in 1 day using the SHM system. Therefore, about 7 days can be saved on inspection.

Downtime is shortened not only because of the efficient assessment process in condition-based maintenance but also due to the time spent on removing/installing the surrounding structures for GVI in scheduled maintenance. In the latter case, the general visual inspection can only be carried out when surrounding structures are removed. For example, if general visual inspection is performed on fuselage skins in the cargo area, all floor panels, sidewalls, insulation blankets, etc. have to be removed. Downtime of CBM can be reduced by about 5 days by skipping this procedure.

From the analysis above, the downtime can be shortened by 12 days for each maintenance trip in condition-based maintenance. Therefore, the downtime for condition-based maintenance is assumed as 18 days for each maintenance trip:

$$\text{Cost saved} : 27,428 \times (18 \times 30 - 7.8 \times 18) = 1.1 \times 10^7(\$)$$

(2) Inspection cost.

As stated above, the time shortened on inspection by using SHM system is 7 days. Assume that 100 h of labor is needed on inspection per day at $60/h:

$$\text{Cost saved} : 7 \times 100 \times 60 \times 18 = 7.56 \times 10^5(\$)$$

(3) Cost for removing/installing surrounding structures.

The time spent on removing/installing surrounding structures for easy access of GVI is about 5 days with 300 h of labor per day:

$$\text{Cost saved} : 5 \times 300 \times 60 \times 18 = 1.62 \times 10^6(\$)$$

(4) Crack repair cost.

As calculated above, the number of cracks that need to be repaired is 10 in scheduled maintenance and 5.8 in condition-based maintenance for each maintenance trip. Fuselage skin with cracks detected is repaired by different methods depending on the size of the crack [10]. In the case of fuselage skin, the doubler repair is the most common method. Although different repair methods are adopted according to the size of the crack, in this chapter, it is assumed that the typical doubler repair be implemented.

For a doubler of 10 × 10 in, 60 h of labor is needed. The cost for this doubler repair is about $360 with $60 labor cost per hour:

$$\text{Cost saved} : 360 \times (18 \times 10 - 7.6 \times 5.8) = 4.9 \times 10^4(\$)$$

Table 3 summarizes cost increase and decrease for the two maintenance strategies. It can be concluded from the table that total cost saved is about $\$1.18 \times 10^7$, which is about 10% of the lifecycle cost, by using SHM system on condition-based maintenance over scheduled maintenance. The main factor leading to this cost savings is the reduced net revenue lost due to shortened downtime. The effect of cost saved on inspection and removing/installing the surrounding structures is

Cost increased ($)	Manufacturing cost	SHM replacement		Fuel cost	Total
	3×10^5	1.2×10^6		1.5×10^5	1.65×10^6
Cost decreased ($)	Net revenue saved	Inspection cost	Removing/installing cost	Crack repair cost	Total
	1.1×10^7	7.56×10^5	1.62×10^6	4.9×10^4	1.34×10^7
Total cost saved ($)			1.18×10^7		

Table 3.
Summary of cost increased and decreased for two maintenance approaches.

relatively small (20% of the total cost saved). It is also noted that cost saved by the reduced number of cracks repaired is negligible.

6. Potential advantages of condition-based maintenance

In addition to the cost savings calculated, some further potential benefits may be gained by using SHM system on condition-based maintenance. Firstly, skipping the removing/installing surrounding structure procedures in SHM systems not only saves time and labor but also prevents potential damage to structures caused by the removing/installing process. Although the fasteners can be replaced after each removing/installing, the fastener holes, taking rivet holes, for example, will be enlarged after each repair work. This is an irreversible damage and might also be the source of new cracks. Even worse, some accidents might occur during the removing/installing, such as drilling through unrelated structures. All these are troublesome issues reported by MRO companies and airlines frequently. With the introduction of an SHM system, this problem may be eliminated.

Secondly, maintenance is more predictable with the SHM system. In scheduled maintenance, damages are detected by manual inspection in the hangar. For some unexpected damages, several days are wasted on preparing special equipment, tools, and/or materials. Sometimes, it even takes a week or so to confirm a repair plan by consulting the manufacturer of the aircraft.

In condition-based maintenance, however, by monitoring the cracks continuously using the SHM system combined with the Paris-Erdogan model and the MCS to model the growth, crack growth and size are more predictable, thus stepping up maintenance and repair work.

Furthermore, with the ongoing research on sensors and actuators, its detection ability will not only be confined on cracks; it can also be used for detecting other typical structural damages such as corrosion, dents, holes, delamination, etc. By collecting and analyzing all the data from the SHM system, the structures on which certain damage frequently occurred affecting the safety of aircraft could be found. These can be posted in Airworthiness Directives (AD), Service Bulletins (SB), or Service Letters (SL), to help eliminate the potential safety issues in the whole fleet of same aircraft model.

7. Conclusions

Two maintenance approaches are discussed in this chapter. Traditionally, scheduled maintenance is carried out at predetermined intervals to maintain a desired level of safety. Recently, with the development of SHM techniques, condition-based maintenance uses onboard SHM sensors and actuators to detect damage on fuselage skins, which, in turn, may be performed as frequently as needed. Hence, maintenance is requested only when a particular condition is met.

The improved reliability and cost savings of condition-based maintenance over scheduled one are discussed. As the usage of onboard SHM system, downtime for each maintenance trip is shortened significantly in condition-based maintenance, leading to considerable cost saving of net revenue. This SHM system also avoids removing/installing the surrounding structures. All these factors may lead to significant cost savings in CBM. In addition, some potential advantages of condition-based maintenance are discussed in this chapter, which includes reducing the possibility of human error during the maintenance process, preparing maintenance equipment in advance, and using the same sensors to detect other types of damages.

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List of abbreviations

CBM	condition-based maintenance
CVM	comparative vacuum monitoring
DVI	detailed visual inspection
FBG	fiber Bragg grating
GVI	general visual inspection
MCS	Monte Carlo simulation
MRO	maintenance, repair, and overhaul
NDT	nondestructive test
PDF	probability density function
PWAS	piezoelectric wafer active sensor
SHM	structural health monitoring

Appendices

A. Fatigue damage growth due to fuselage pressurization

Fatigue crack growth can be modeled in a number of ways. Beden et al. [30] provided an extensive review of crack growth models. Mohanty et al. [31] used an exponential model to model fatigue crack growth. Scarf [32] advocated the use of simple models, when the objective was to demonstrate the predictability of crack growth. In this chapter, a simple Paris-Erdogan model [33] is considered to describe the crack growth behavior. However, other advanced models can also be used.

Damage in the fuselage skin of an airplane is modeled as a through-the-thickness center crack in an infinite plate. The life of an airplane can be viewed as consisting of damage growth cycles, interspersed with inspection and repair. The cycles of pressure difference between the interior and the exterior of the cabin during each flight is instrumental in fatigue damage growth. The crack growth behavior is modeled using the Paris-Erdogan model, which gives the rate of damage size growth as a function of half damage size (a), pressure differential (p), thickness of fuselage skin (t), fuselage radius (r), and Paris-Erdogan model parameters, C and m:

$$\frac{da}{dx} = C(\Delta K)^m \tag{1}$$

where the range of stress intensity factor is approximated with the stress $\Delta\sigma = pr/t$ as

$$\Delta K = \Delta\sigma\sqrt{\pi a} \tag{2}$$

The following critical crack size can cause failure of the panel and is approximated as

$$\sqrt{a_{cr}} = \frac{K_{IC}}{\Delta\sigma\sqrt{\pi}} \tag{3}$$

where K_{IC} is the fracture toughness of an infinite plate with a through-the-thickness center crack loaded in the Mode-I direction.

In the above damage growth process, the following uncertainty is considered: uncertainty in the Paris-Erdogan model parameters, pressure differential, and initial crack size. The damage size after N flight cycles depends on the aforementioned parameters and is also uncertain. The values of uncertain parameters are tabulated in **Table 4**.

It is approximated that all fuselage skins are made of aluminum alloy 2024-T3 with dimensions of $57' \times 57' \times 0.063$ (17.4 m $\times$ 17.4 m $\times$ 1.6 mm). Newmann et al. (Pg 113, **Figure 3**) [34] showed the experimental data plot between the damage growth rate and the intercept and slope, respectively, of the region corresponding to stable damage growth. As the region of the stable damage growth can be bounded by a parallelogram, the estimates of the bounds of the parameters, C *and* m, are obtained from **Figure 3** of Newmann et al. [34].

For a given value of intercept C, there is only a range of slope (m) permissible in the estimated parallelogram. To parameterize the bounds, the left and right edges of the parallelogram were discretized by uniformly distributed points. Each point on

Parameter	Type	Value
Initial crack size (a_0)	Random	LN(0.2, 0.07)mm
Pressure (p)	Random	LN(0.06, 0.003)MPa
Radius of fuselage (r)	Deterministic	2 m (76.5 in)
Thickness of fuselage skin (t)	Deterministic	1.6 mm (0.063 in)
Mode-I fracture toughness (K_{IC})	Deterministic	$36.58 MPa\sqrt{m}$
Paris-Erdogan law constant (C)	Random	U[$\log_{10}$(5E-11), $\log_{10}$(5E-10)]
Paris-Erdogan law exponent (m)	Random	U[3, 4.3]
Palmberg parameter for scheduled maintenance (a_{h-man})	Deterministic	12.7 mm (0.5 in)
Palmberg parameter for scheduled maintenance (β_{man})	Deterministic	0.5
Palmberg parameter for SHM based inspection (a_{h-shm})	Deterministic	5 mm (0.2 in)
Palmberg parameter for SHM based inspection (β_{shm})	Deterministic	5.0

Table 4.
Parameters for crack growth and inspection.

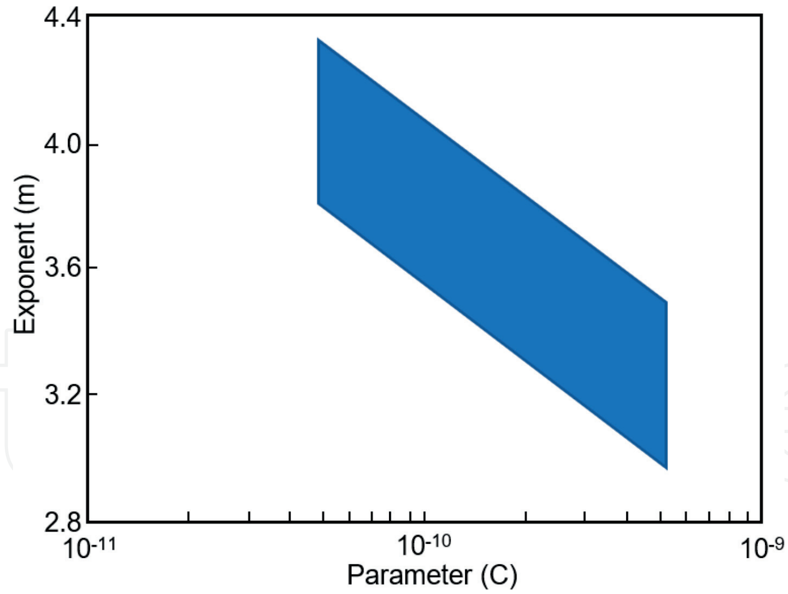

Figure 5.
Possible region of Paris-Erdogan model parameters.

the left and right corresponds to a value of C. For a given value of C, there are only certain possible values of the slope, m. **Figure 5** plots those permissible ranges of slop (m), for a given value of intercept (C). It can be seen from **Figure 5** that the slope and $\log(C)$ are negatively correlated; the correlation coefficient is found to be about -0.8.

B. Inspection model

Kim et al. [35], Packman et al. [36], Berens and Hovey [37], Madsen et al. [38], Mori and Ellingwood [39], and Chung et al. [40] have modeled the damage detection probability as a function of damage size. In this chapter, the inspection of fuselage skins for damage is modeled using the Palmberg equation.

In scheduled maintenance and in SHM-based maintenance assessment, the detection probability can be modeled using the Palmberg Equation [41] given by

$$P_d(a) = \frac{\left(\frac{a}{a_h}\right)^{\beta}}{1 + \left(\frac{a}{a_h}\right)^{\beta}} \tag{4}$$

The expression gives the probability of detecting damage with size $2a$. In Eq. (4), a_h is the half damage size corresponding to 50% probability of detection, and β is the randomness parameter. Parameter a_h represents average capability of the inspection method, while β represents the variability in the process. Different values of the parameter, a_h and β, are considered to model the inspection for scheduled maintenance and also for SHM-based maintenance assessment. **Table 4** shows the parameters used in the damage growth model, as well as the inspection model.

C. Direct integration procedure

The direct integration procedure is a method used to compute the probability of an output variable with random input variables. In general, Monte Carlo simulation can be used to calculate the probability, but it requires many samples, and the results have sampling error. In this chapter, the direct integration process is used to compute the probability of having a specific crack size. The damage size distribution is a function of initial crack size, pressure differential, and Paris-Erdogan model parameter (C, m), which are all random:

$$f_N(a) = h(a_0, f(p), J(C, m)) \tag{5}$$

where a_0, $f_N(a)$, and $f(p)$ represent the initial crack size, the probability density function of crack size after N cycles, and the pressure differential, respectively. $J(C, m)$ is the joint probability density of the Paris-Erdogan model parameters (C, m). The probability of crack size being less than a_N after N cycles is the integration of the joint probability density of input parameters over the region that results in a crack size being less than or equal to a_N, that is,

$$\Pr(a \le a_N) = \int \dots \int_R a_0 J(C, m) f(p) \mathrm{d}R \tag{6}$$

where R represents the region of (a_0, C, m, p) which will give $a \le a_N$.

Based on preliminary analysis performed by the authors, the effect of random pressure differential was averaged out over a large number of flight cycles. Therefore, the average of the pressure differential is used in the following calculation. Hence, Eq. (6) reduces to be a function of m and C, as

$$F_N(40) = \iint_A J(C, m)\ \mathrm{d}C\mathrm{d}m \tag{7}$$

where A represents the region of $\{C, m\}$ that would give $a_N \le 40mm$ for a given initial crack size, a_0. The parallelogram in **Figure 6** is the region of all possible combinations of Paris-Erdogan model parameters, $\{C, m\}$. For the initial crack size, $a_0 = 1mm$, cracks in the gray triangular region will grow beyond 40 mm after $N = 50,000$ cycles. If the initial crack size is distributed, then the integrand is evaluated at different values in the range of the initial crack size, and the trapezoidal rule is used to compute the probability at the desired crack size.

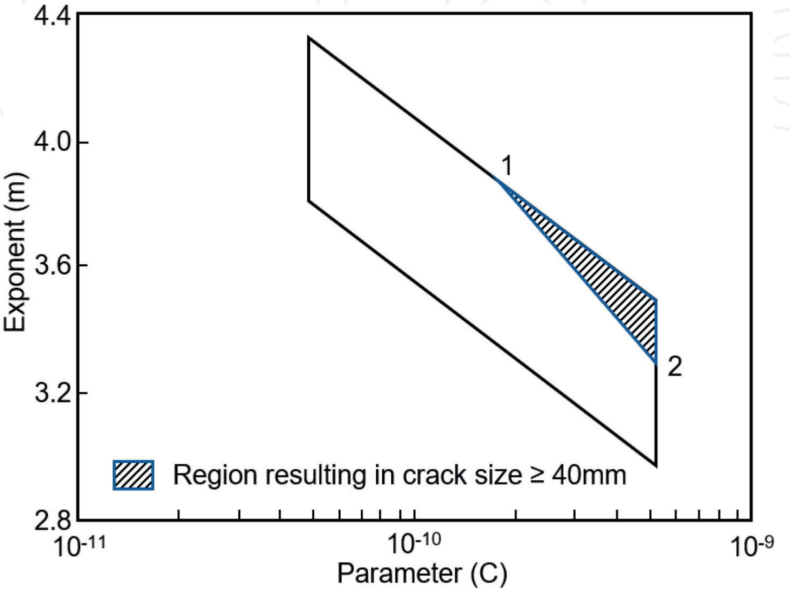

Figure 6.
Regions of $\{C, m\}$ for $N = 50,000$ and $a_0 = 1mm$.

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