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Aging Commuter Aeroplanes - Fatigue Evaluation and Control Methods

A.J. Emmerson, Civil Aviation Authority of Australia

Introduction

The risks associated with old aeroplanes arise in the main from the difficulty of obtaining support from the manufacturer and other skilled people, and from a general loss of reliability caused by wear and other deterioration.

The loss of reliability is caused by two broad classes of problem. There are those problems which are self evident, and hazardous rather than catastrophic. These are the problem areas where characteristically there have been multiple overhauls, repairs, and replacements, and where aging really means the results of repair ineffectiveness that accumulates.

The other class of problem is the insidious and potentially catastrophic class. It includes the progressive deterioration of items which are not maintained, and often cannot be maintained because the deterioration cannot be seen. It includes loss of physical properties in adhesives and other organic compounds, corrosion, and the response to repeated loads. This paper deals with a currently unnecessarily troublesome aspect of that response.

The present working definition of "commuter aeroplanes" encompasses a wide range of aging types, from the Fairchild F27, through deHavilland Canada Twin Otters to the smaller twin engined Piper and Cessna models. It includes such types as the Lockheed 10, the Beech 18, and the Grumman Mallard. Although we must remain concerned about those types which have been certificated under a design standard or operational rule which embodies the elementary fail-safe concept and which have not been subjected to a subsequent structural audit, this paper is directed, in the first instance, towards types for which fatigue and damage tolerance evaluation was not required as a condition of certification. It will be seen, however, that the principles are robust enough for more general application.

An Australian Historical Perspective

In Australia, the first recognised fatigue accident to what would now be called a commuter aeroplane happened on 31 January 1945. A Stinson Model A, the forerunner of the current range of twin engined Piper commuters, crashed enroute from Melbourne to Mildura. All ten people on board were killed.

The ensuing public inquiry recommended to the Australian Parliament that aircraft maintenance engineers be given special instructions covering the inspection of vital parts of aeroplanes, that measures for non-destructive inspection should be instituted, and that the operational lives of aeroplanes should be limited.

Those recommendations were adopted, although the methods used were of course quite crude by today's standards. A method of evaluating the deterioration due to cyclic loads when airborne had only just become available (1-3). The lack of sophistication in fatigue evaluation was not evident at the time. But we were soon to learn. On 15 October 1951, deHavilland Dove VH-AQO crashed killing the seven people on board when the wing suffered a fatigue failure enroute from Perth to Kalgoorlie. The Dove on which the investigators travelled to the accident site was also found to have a large fatigue crack in the wing. VH-AQO and her four sister ships were the first Doves in airline service anywhere in the world. VH-AQO was four years old at the date of the accident. She had accumulated 9000 flying hours. Had the safe life been calculated by present methods, the wing of VH-AQO would have been retired after 2500 flying hours.

In the USA, the first explicit requirement for fatigue evaluation for general aviation aeroplanes was introduced with CAR 3 in 1957. At first only pressure cabins were covered, but in September 1969 Amendment 7 to FAR 23 required the evaluation of wing structure. FAR 23-7 required the wing to be evaluated on either a safe-life or a fail-safe basis. The fail-safe residual strength capability was 75% of limit load, to which a "dynamic factor" of 1.15 was to be applied.

In making this amendment, in August 1969, the FAA said:

"Service experience and discussions with industry of designs which have sustained fatigue failures indicate that present design practices do not adequately account for fatigue. Corrosion, prior abuse, and special purpose operations may contribute to fatigue failure, but the primary reason for such failures is lack of strength. Since fatigue failures are time dependent, a higher failure rate among older airplanes is to be expected; however, fatigue problems have arisen in airplanes certificated in recent years. Neither fatigue substantiation nor better maintenance programs will eliminate all cracks. The purpose of the proposed rule is to prevent catastrophic failures. Furthermore [FAR 23] airplanes are not designed on a redundant-structure, fail-safe basis for which maintenance alone would be sufficient. Both fatigue substantiation and a good maintenance program are needed (4)."

We believe the FAA was referring to a dozen or so types which had been designed to gain performance advantages from modern materials and design techniques, but which were certificated to the standard preceding FAR 23.

A major problem from Australia's viewpoint was that the new FAR 23 only applied to applications for certification of types completely new to the USA. Under the "grandfather clause" improved models of existing aircraft can be approved on the basis of the original Type Certificate. (To illustrate, all Mooney M20 series aeroplanes are covered by the same FAA Aircraft Specification 2A3, from the wooden winged M20 of 1955 to the 173 knot 2900 lb M20 K of 1982.) Thus there was a potential for a large range of problematical US aircraft to be imported into Australia for which no fatigue substantiation had been provided to the FAA.

Accordingly, in 1970, Australian airworthiness engineers visited the USA to discuss in detail with the FAA, and the manufacturers, an Australian proposal to require fatigue evaluation of all aircraft types entering Australia for the first time. The proposal was endorsed by the FAA. The USA manufacturers cooperated freely with evaluations of both new types and of older types already in Australian service.

The Australian airworthiness authority then imposed retirement lives on several of the types at risk. Those lives were based on:

- stress analysis and measurement
- knowledge of the fatigue resistance of similar classes of structure
- measurement of the severity of Australian service
- fatigue test data and lifing recommendations from the manufacturers
- failures in service
- Miner's linear cumulative damage rule (3)
- specified risk of failure of principal elements

The general evaluation methods used in Australia were very similar to those adopted by the FAA and the general aviation manufacturers in the USA as a standard approach in 1973. They have been published in FAA report AFS 120-73-2 Fatigue Evaluation of Wing and Associated Structure on Small Airplanes.

The Australian authority carried out a programme of measuring aircraft normal accelerations resulting from gusts and manoeuvres. Some 75 aeroplanes were fitted with recording accelerometers and a total of some 150,000 hours flying was recorded and analysed. The results are available from the author.

Control of the continuing airworthiness of these aircraft in the face of fatigue rested on five points:

- Thorough fatigue evaluation of the principal bending elements at the time the type is first certificated (if the prerequisites for safety by inspection are not met, a retirement life must be set for the spars, unless the estimated safe life is greater than 40,000 hours)
- Mandatory defect reporting, close monitoring of overseas experience, and effective investigation of failures
- Spars may be changed, but primary structure which has not been evaluated may remain in service for only twice the life of the spars
- To extend the life of the aeroplane beyond two spar lifetimes there must be a structural audit and a review of the initial evaluation with a view to demonstrating damage tolerance of the principal structural elements up to some new extended retirement life

- Continuous amicable contact and free exchange of information between the operator, the manufacturer, and the authority is essential.

Retirement lives for aeroplane types vary between models and operational roles, and according to the confidence the airworthiness authority has in the evaluation. Table 1 is an indicative list of the lives now applied to some relevant types in Australia.

The anticipated outgrowth of the pre FAR 23 types did in fact occur. The general aviation twins now in commuter, air ambulance, and charter use are derivatives of earlier models and are frequently operated in a manner quite unforeseen by the manufacturer or the original certificating authority. Each year's model change of a few pounds weight, a few knots in speed and perhaps an extra two seats meant that the original basic Type Certificate could be extended. Ten or twenty years of this treatment produced aircraft having ideal performance for commuter passenger carriage with utilisation up to 2000 hours per year.

The Swearingen SA 26AT Merlin II is a good example of such a growth aeroplane - although it happens to have its own Type Certificate dating from May 1956. The Merlin II was manufactured by using wings and other components from used Beech 50 Twin Bonanza aircraft (certificated in 1949), and fitting a new pressure cabin and turbine engines. The gross weight rose from 5500lb to 10000lb, the engine power from 240HP to 972HP and the maximum cruise speed from 157 knots to 208 knots. Apart from some pressure cabin work, no fatigue evaluation was required or supplied for United States certification.

In Australia a life of 7500 hours was applied to the wing of the Merlin II. Similar action was taken with the Queen Air and King Air which were themselves growth versions of the Beech 50 using the same wing spar. A wing failure of a Beech 90 King Air in Canada and numerous spar fatigue cracks discovered in other Queen Air and King Air aeroplanes lend weight to the validity of the Australian action which had been vehemently opposed at the time. Beech aircraft are not alone of course in experiencing wing cracking. AeroCommander and Cessna and Partenavia twins, for example, have also experienced major cracks in main spars caused by fatigue. A fairly complete summary of accidents caused by fatigue in primary structure is in (5).

In an attempt to introduce some retroactivity into the requirements, Amendment 18 was added to FAR 135 in July 1970. FAR 135.169 carries a fatigue evaluation requirement for aircraft operating commercially after 1972 and having ten or more passenger seats. The FAA introduced Special Federal Aviation Regulation No. 41 in October 1979 to permit certain FAR 23 aircraft with more than 19 passenger seats to operate at higher than normal weights. A condition of the weight increase was that additional fatigue substantiation was to be provided, covering the vertical fin, horizontal stabilizer, and attaching structure. When SFAR 41 expired in 1989, FAR 23 was amended to include a commuter class with the same fatigue evaluation requirement. Australian and American certification requirements are now essentially identical for this class of aeroplane.

The current situation is that some 75 or so aircraft types have retirement lives placed on their structure in Australia. Of those, the retirement life is endorsed by the country of origin's airworthiness authority in all but eleven types. Those eleven types include eight pre 1969 twin engined types. They are no longer in production, they are getting old, they have a high and severe usage in Australia, and most individual aeroplanes are approaching their retirement life. They are nevertheless aeroplanes of some economic importance.

We have about 1250 light twins in passenger carrying operations in Australia. The Cessna 400 and the Piper PA 31 typify the types used in charter and commuter operations. There are some 200 Cessna 400 aeroplanes (and 160 Cessna 300 which share the same wing design) plus 210 Piper PA 31 in Australia. Most of them are in commercial service.

The service history of commuter aeroplanes presents a problem for regulatory authorities. There has been a significant number of fatal accidents caused by fatigue failure in the primary structure of that sort of aeroplane - aeroplanes of the same general class of design and construction, in the secondary airline operational and maintenance environment. On the other hand, there seems to have been a large number of aeroplanes of the class flying over the years that ought to have given confidence in their durability. Table 2 illustrates. But, to be truthful, we do not know how many of these aircraft are carrying cracks that have reduced their strength to less than the requirements of the design standard.

It is possible to evaluate this fleet experience. However the evaluation cannot be based on anecdotes. Hard evidence is required of aircraft that are structurally identical, and that have been flown in similar ways, and that have been free of defects.

Three final observations should be made about this Australian retrospective. Firstly, retirements due to the aging process have not just been confined to metallic aircraft structures. In the late 1950s several "commuter" types with glued wooden spars were retired from service because of progressive deterioration. They included for example the Avro Anson and some Percival types. The lesson is there for fibre composites.

Secondly, if in a fleet the cost is reckoned as the cost of replacing spars, and the benefit is reckoned as the saving of ten lives in the first accident, the benefits of Australia's policy have clearly outweighed the costs. Difficulty arises of course when the obvious cost accrues to the operator and the invisible benefit accrues to the public.

Thirdly, the application of retirement lives has been unduly prominent because of the lack of damage tolerance in many types, and of course as a result of the commercial implications of retirement. Since Australia embarked formally on the control of fatigue in civilian aircraft in 1945, the preferred method of control has been to maintain safety by inspection.

A review of the literature will show Australian contributions among the earliest papers on safety by inspection - for example Shaw's paper of 1954 (6). More recently, the CAA has encouraged and assisted local organisations to undertake damage tolerance evaluation of commuter types, and to design and fit modifications that will convey some measure of damage tolerance so as to permit safety by inspection. The Nord 262, in its Frakes 298 form, the Cessna 400 and the Piper PA 31 are the more significant types tackled successfully.

It is interesting, if not surprising, in this work how often the obvious is overlooked. Despite the commercial penalty of poor damage tolerance in some commuter types, no one in the Australian industry has yet proposed that on the expiration of safe life, 7075-T6 spar caps should be replaced with 2024-T3 caps of slightly greater thickness.

Evaluation and Control Methods

The Safe Life Concept

Before we can accept that an aircraft's structure can be kept safe by inspection we must know where to look, we must be able to look there, and there must be time to look. We need to determine how to look and when to look. The implications of accessibility and slow crack growth, the ability to tolerate damage, are clear. If these criteria cannot be satisfied, there is no airworthiness control alternative but to plan to retire the aircraft from service at some safe life.

The history of fatigue evaluation has in that way caused "safe-life" to be regarded as synonymous with "retirement life" and with retirement as a control technique; the adjective "safe-life" is also commonly used to describe a class of structures that should be controlled by retirement. None of those associations are necessary, and in hindsight we have been a little careless with our terminology.

To start with, a safe life only becomes a retirement life if it is promulgated as such by the airworthiness authority. But more generally, we should recognise that *a safe life is simply the life at which the risk of failure is predicted*, in the absence of information to the contrary, *to become unacceptable*. It is the time by which we had better be doing something.

Depending upon our control strategy, the safe life may be either the latest time by which we should have retired the aircraft, or the time by which we should have begun inspecting the structure for cracks if we are going to maintain safety by inspection.

In either case *the safe life will be the time at which* the chance that *the structure is unable to carry ultimate design load* has become unacceptable.

Methods of Estimation

Consider for a moment the way in which this loss of strength occurs. There can be said to be three phases in the generation of cracks due to repeated loads. The first is one in which so far unobservable changes are taking place on a sub-crystal-lattice scale due to an unknown process about which there have been some metallurgical postulates.

The second phase is that transition phase which starts when the micro structure changes caused by repeated loads are above lattice scale but are too small to be detected or measured by conventional means, and for which the applicability of the concepts of fracture mechanics has not been demonstrated and is theoretically dubious because the material is not a continuum on this scale.

The third phase is the growth of identifiable cracks - a phase in which our rules of stress intensity and crack growth rates have been shown to apply.

It is not at all clear at what point strength first falls below virginal strength but it is conventional to associate this with the limit of resolution of nondestructive inspection - at the end of the transition phase.

Phase one changes and phase two changes are localised and accelerated by the presence of cracklike defects left during manufacture, and, equally, by the presence of micro structure irregularities of a metallurgical scale. The description of the fabricated material for analytical purposes must include an accounting for the distribution of flaws of these various sizes. Historically this has more often been implicit than explicit.

With sufficient accuracy for the purpose, in the class of structures we are considering, the conclusion of the transition phase and the point at which the strength falls below ultimate load capability may be regarded as coincident. Few of these structures could withstand ultimate load in the presence of the stress concentrating effect of a crack. The end of the transition phase is thus reasonably regarded as coincident with the safe life.

There are in use two methods of estimating this life - the linear cumulative damage hypothesis, and the growth of initial flaws hypothesis. It may not be too adventurous to proffer the opinion that this dichotomy is impeding the development of control methods for aging commuter aircraft.

It has been the author's observation over the past fifteen or so years that the proponents of these two methods have become implacably resistant to one another's views. This seems to be because:

- The need to have resistance to damage or flaws cannot be denied and thus there must be some study of the propagation of less than obvious flaws, and such flaws could be present just after manufacture.

- The methods have in the past been associated with allegedly undesirable or inappropriate practices; and,

- The essential similarity of the two methods has not been widely recognised.

It would be remarkable if that last proposition were to go unchallenged. But consider the matter more broadly.

Through nothing more than the sequence of development in the history of fatigue evaluation, a quite unnecessary connection has been made between the initially unreliable linear cumulative damage hypothesis and the sometimes wasteful imposition of retirement lives. The two are not congruent. We can have one without the other.

Similarly a connection has been made between the less explored propagation of initial flaws hypothesis and the safety by inspection process. The two have crack growth calculations in common but they are not congruent and not necessarily conjugate concepts.

In linear cumulative damage analysis the following steps are used:

- Begin with data describing the endurance under repeated loads of structure of the general type for the member being evaluated - usually in the form of an SN curve established by pooling full scale test data.

- Scale or calibrate that curve so that when using Miner's cumulative damage rule the curve predicts the results achieved by a fatigue test of the actual component.

- Apply the anticipated load spectrum, as stress range pairs, to the calibrated curve to estimate mean life in service.

- Using standard statistical techniques, correct this estimate for a want of confidence in the mean life; and, using Impellezziri's proposition that the variance between nominally identical structures is a characteristic of the class of fabrication (7), calculate the safe life.

This technique, although derived from Miner's 1945 hypothesis, has benefited from sophistication added over some thirty years - principally by the members of the International Committee on Aeronautical Fatigue. It would be better known as the Miner/ICAF method.

The propagation of initial flaws approach uses the following steps:

- Begin with data describing, for structure of the general type of the member being evaluated, the range of initial flaw sizes and their frequency of occurrence. Choose an initial flaw size, the frequency of occurrence of which corresponds to the acceptable risk of failure.

- Develop endurance data in the form of crack growth curves for particular locations from standard curve shapes calibrated to replicate the results of crack growth measurement in tests of the member being evaluated. Correct that data for want of confidence in the test result representing the mean.

- Apply the anticipated load spectrum, as stress range pairs, to the calibrated curve to estimate safe life in service.

- Repeat this process for other crack locations in the same member.

Although this method arose out of the F111 recovery program, the crack-like initial flaws postulate was made by Griffith many years earlier. As the crack growth equations stem from the initial work of Paris in 1964 (8), the method might be called the Griffith/Paris method.

Several points should be made about these techniques. The original Miner method gained a reputation for unreliability. This was very largely a result of the pitfalls that were at the time undiscovered. The principle lessons over the years have been the extent of the innate variance between nominally identical structures; and the need for SN data that is thoroughly representative of the structure being evaluated and its propensity to develop cracks. The SN data must account for poor design geometry, built-in stresses, fretting, and the stresses due to incidental loads.

Some pitfalls have also been discovered in the Griffith/Paris method. It is fair to argue that as yet the technique is not mature, and that there has not been sufficient application to confirm any suspected unreliability. In commuter aircraft, we have seen very few failures from crack-like manufacturing faults of discernible size, other than bad welds and hydrogen embrittlement.

SN data drawn from full scale tests most often covers a fair range of failure modes in the one curve. Crack growth curves are particular to a location and this adds to the computational work.

The essential elements of the two methods are the same. Externalise the variability between nominally identical items. Calibrate an empirical deterioration curve representative of the class of structure against a full scale test. Then by piecewise integration calculate the number of flights to reduce strength to a specified value.

If we were to search for a difference in the physics implied by the two methods, an important theoretical difference would be that the endurance curves are of a different nature. In one the estimated microstructure response to load cycles is independent of what has gone before. In the other the response is dependent on what has gone before. The question that might be asked then is what are the physics during the safe life of commuter structures. The best answer is that while crack sizes remain small, and providing the curves are properly calibrated, it doesn't matter.

If there is a significant difference in the reliability of the methods, it will arise from the extent to which the final endurance data was derived from experiments in which the distribution of initial flaws, micro or macro, or propensity to crack, was the same as in the structure being investigated.

If the author were asked to express a preference for evaluating small commuter aeroplanes, it would be to use Miner/ICAF to safe life and fracture mechanics beyond that - simply on the grounds of computational simplicity and the availability of data. There can be no objection to a Griffith/Paris approach well based on measured data.

Endurance data should be calibrated against the fleet as a flying fatigue test. The desk top computer has made this perfectly practicable - along with full allowance for variability in the combined fatigue and crack growth processes. It is probably now too late to worry about the probability of early failures from the largest manufacturing flaw which might have escaped detection.

To end on a philosophical note,

> "It seems that no other scientific area has been studied so intensely and yet resulted in so little correlation between quantitative empirical data and theoretical computations based upon some hypothetical physical mechanism. The underlying reason for this lack of consistent correlation is the complexity of the basic mechanism of fatigue. The fact that the fatigue of metals takes place on a submicroscopic scale requires highly sophisticated testing techniques in order to observe the physics of the phenomenon [and in particular] what is essential to the mechanism and what is incidental (7) ."

Engineers ought not to rely on entirely self referential doctrine - doctrine which has its fixed points of reference inside itself. Such doctrine has an instability analogous to a statically indeterminate mechanism. Our theory must be pinned at least at one end on irrefutable external evidence. The position of the other end can then be located by inference. The physics of the response to repeated loads, preceding visible crack growth, have not been directly observed. The Miner/ICAF theory and the Griffith/Paris theory are in the first instance self referential theories.

The aging aircraft issue is topical but it is not a new issue. Aging aircraft have periodically caused concern throughout the seventy year history of Australian commercial aviation. While this may have been especially so in times of economic difficulty and undercapitalised airlines, Australia's good years produced utilisation rates that frequently made Australian aircraft the leaders of the world's fleet. The particular problem of structural aging or fatigue is one in which Australia has perforce had a special interest over the years. It is a particular area where regulatory interest by Government is necessary to achieve a proper balance between the pressures of commerce, and public safety in the long run.

Q-5.

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Table 1 Retirement Lives of Some Components of Representative Light Twins

Aircraft		Component	Retirement Life (hours)
Cessna	401/402	Wing main spar lower cap	8200
	402C		7700
	421C (early)		6400
	421C (late)		17300
Piper	PA 31	Wing main spar lower cap	11000
	PA 31-350		13000
	PA 42-1000	fuselage	13650 cycles
		empennage	37500
Beech	65 B80	wing main spar lower cap	12000

Table 2 Production History of Representative Light Twins

	Model	First Flown	Number Built
Cessna	411	1962	301
	401	1966	404
	402	1966	1540
	404	1975	378
	414	1968	1067
	421	1965	1909
	425	1978	232
	441	1975	360
Piper	PA 31	1964	800
	PA 31-350	1978	1850