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To: 9-NPRM-CMTS
cc: Frederick Sobeck; BOB SEXTON
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Ladies and Gentlemen

My letter:

<<NPRM Comments.doc>>

My ICAF Paper:

<<Gnats and Camels (8.5x11 version).doc>>

Thank you for the opportunity to comment.

Steve Swift
Principal Engineer
Fatigue Evaluation
Civil Aviation Safety Authority
Australia

NPRM Comments.doc   RFC822.TXT   Gnats and Camels

GNATS AND CAMELS

30 Years of Regulating Structural Fatigue in Light Aircraft

Steve Swift[12183]

This paper reviews thirty years of regulating structural fatigue in light aircraft from an Australian perspective. Australia has one of the world's largest and hardest-working fleets. Australia's regulator (CASA) has been an active international participant.

The paper looks at the history and effectiveness of FAR 23.572, and the issues and controversies along the way. Undoubtedly safety has improved. Wings and things still break, but less often.

Suggestions for further improvement include better targeting of regulatory effort and more international cooperation. We still have much to learn. We can only assess the effectiveness of new policies and methods by revisiting them after ten. twenty or thirty years service experience.

INTRODUCTION

Exactly thirty years have passed since the United States Federal Aviation Administration (FAA) issued FAR 23.572, its first fatigue rule for light aircraft wings. It faced stiff industry opposition. Fatigue rules often do.

It is now worth asking whether FAR 23.572 has been effective and whether there is room for improvement.

Australia is a good vantage point for such a review. It has enforced the rule for more light aircraft than any other country. Some are now the world's oldest in terms of hours flown.

GNATS AND CAMELS

The title of this paper is not inspired by Follands and Sopwiths,[0] but by the biblical text, 'You strain out a gnat but swallow a camel'.[15689] It was spoken as a rebuke to the morals regulators of that day for their obsession with the finer points of religious law while ignoring basic principles of morality and justice. The history of FAR 23.572 has shown a similar tendency to focus on minute details at the expense of basic principles. The author confesses to his own guilt in this.

When approving light aircraft, regulators spend more time disputing the finer points of scatter factors and crack retardation than assessing locations and loads.

When approving test plans for composites, regulators argue the load enhancement factors which affect test duration. Then they ignore the duration.

More than any other area of aviation safety regulation, structural fatigue demands wisdom and experience. After fifteen years, the author barely feels competent. He is thankful for good mentors.

[12183] Principal Engineer, Fatigue Evaluation, Civil Aviation Safety Authority (CASA), Canberra, Australia. The views in this paper are the author's and not necessarily CASA's.

[0] Do you remember the Folland Gnat and the Sopwith Camel?

[15689] Jesus Christ, Matthew 23:24

RULES AND MORE RULES

Through the years, the FAA has used several rules to regulate structural fatigue in light aircraft:

Rule	Effective	Applicability
CAR 3.270	1957	Pressurised fuselages only.
FAR 23.57 1	1964	Pressurised fuselages only.
SFAR 23.28	1969	Commercial operations only. 11 or more occupants only.
FAR 23.572	1969	Wings only until 1989 when empennage added.
FAR 135 Appendix A.28	1978	Commercial operations only. 10 or more passengers only.
SFAR 41.5	1980	10- 19 passengers only.
FAR 23.573	1993	Composites only.
FAR 23.574	1996	Commuter category only.
FAR 135.168	1999? (see footnote 9)	Scheduled operations only. Multi-engine only.

Broadly, they require the designer to predict the fatigue behaviour of the structure, then use the prediction to develop actions and schedules that will keep the structure airworthy.

At first, the rules allowed two maintenance options:

- Remove parts from service at a fixed time, *before* they crack (called the 'safe life' method); or
- *Leave* parts in service *until* they crack, but only if the structure is 'fail safe' (which means that cracks are expected to be obvious before they are dangerous).

Neither requires inspections. 'Damage tolerance', a third and later option, now does.

LIGHT OR HEAVY?

The word 'light' can be misleading when describing aircraft. For a long time the regulatory dividing line was 12,500 pounds maximum take off weight. But now, FAR 23[0] applies to aircraft as heavy as 19,000 pounds (Commuter Category). Hence light aircraft are Beech 1900s as well as Cessna 150s. They carry fare-paying passengers as well as 'weekend warriors'.

[0] FAR 23 is the United States design standard for light aircraft. For heavy aircraft, it is FAR 25.

Firstly, there are regulatory differences between light and heavy aircraft:

	FAR 23 (light)	FAR 25 (heavy)
Safe Life	Allowable Option	Allowed only if damage tolerance is 'impractical',
Fail Safe	Allowable Option	No
Damage Tolerance	Allowable Option	Required for most structure.

Why does FAR 23 still allow 'fail safe' when, for good reason, FAR 25 has not allowed it for heavy aircraft since 1978.[34097] 'Fail-safe' is un-safe for *all* aircraft, whether light or heavy. Moreover, a bit more analysis and testing can usually produce an inspection program to achieve 'damage tolerance'. Such programs might soon be as common for light aircraft as they are now for heavy aircraft.

Like 'fail-safe', FAR 25 has not allowed 'safe life' for heavy aircraft for a long time. But unlike 'fail-safe', 'safe life' is still safe for light aircraft and is rightly still an option in FAR 23.572 (see 'Safe Life and Damage Tolerance').

Secondly, there are structural differences between light and heavy aircraft. In light aircraft it is harder to duplicate major structural parts such as wing spars. For this reason, it was once thought impractical to design 'damage tolerant' light aircraft. However, light aircraft *can* achieve damage tolerance. Some do have two spars, like the Cessna 404 Titan. [0] Some have a multi-element spar, like the Piper PA-3 I-350 Chieftain. Some have a monolithic spar with low stresses in a tough material, like the Piper PA-32.[35210] Some are made from composites, such as the Grob 115.

CHOOSE YOUR PARENTS CAREFULLY

The FAA has approved 88 models of American light twins since 14 September 1972 - three years after FAR 23.572 came into force. Therefore, FAR 23.572 should have been part of their design standard.' But FAA waived it for 50 of the 88. Why?

The reason is that few of the 'new' models have been genuinely new. Most have been derivative descendants of earlier models. As descendants, they kept the right to retain the old design rules their 'parents' and 'grandparents' had. Thus they avoided newer, safer, rules such as FAR 23.572. Typical is the Mooney M20 series. FAA has not yet enforced FAR 23.572 despite 45 years of continuing evolution from a wooden 'puddle-jumper' to an all-metal, 200 knot 'hot rod'. This is what is known as 'grandfathering'.

[34097] Amendment 45

[0] Although only certificated 'failsafe', Cessna probably did enough work to satisfy the damage tolerance option.

[35210] The damage tolerance of this aircraft is being explored in Australia at the time of writing.

[0] According to FAR 2 1.17(c), three years is the maximum time allowed between allocation of a design standard and final approval of the design.

Grandfathering has a legitimate place where the application of a new rule would require the expense of a major modification to a proven design. But there are good reasons why grandfathering for fatigue is neither necessary nor appropriate:

- Fatigue rules are easy to comply with retrospectively and don't require re-design.
- Old designs are never proven for fatigue simply by virtue of their longevity. Fatigue is wear-out. There is no guarantee that future failures will be confined to those seen in the past.
- Fatigue is unavoidable and is better dealt with sooner rather than later.

The FAA now worries about the safety consequences of grandfathering. It is using its operating rules to retrospectively require a fatigue evaluation for grandfathered aircraft. It started with 10+ passenger commercial aircraft. Now FAA proposes to extend this to *all* multi-engined commercial aircraft.[6617] In its Notice of Proposed Rulemaking (NPRM), the FAA said:

> *Left unchecked, it is not a question of whether the repeated loadings on aircraft will produce a major structural failure, but rather, when.*

Moreover:

> *Under existing procedures (mostly no fatigue evaluation at ail), the FAA cannot assure the continuing airworthiness of these airplanes, and that constitutes an unacceptable risk to air transportation.*

For this reason, hopefully it won't take the FAA too many more years to address the tens of thousands of other grandfathered light aircraft which continue to grow older, unsafely, in the United States.

Overcoming grandfathering will improve consistency as well as safety. For example, the Cessna 421C and 425 both share the same wing, but only the 425 has a wing life on the Type Certificate Data Sheet.

BARK AND BITE

Regulators enforce fatigue rules at type certification with more vigour than they later enforce their maintenance consequences. They turn a blind eye to life limits in Type Certificates. They welcome excuses to extend life limits and delay inspections when the time comes.

When it comes to grounding aircraft, regulatory managers face enormous industry and political pressure. Senior technical staff need to be well prepared to defend the doctrine.

Quoting from an earlier report by the author (Swift 1995):

> *It seems that there are attributes offdigue and corrosion - scatter is one - which mean that engineers must work very hard to convince their managers in the face of commercial and political pressures. Fatigue engineers need more than technical skills.*

AUSTRALIA AND THE UNITED STATES

In 1945, in Australia, fatigue caused the left wing to break off a Stinson Model A airliner. The accident investigators concluded that the crack could not have been found by normal maintenance. A few months later, a public inquiry recommended 'special equipment' to find cracks.

[6617] NPRM Docket No. FAA- 1999-5401; Notice No. 99-02, 2 April 1999

However, even with the special equipment of the day, the inquiry could not be confident of reliably finding every crack, so it also recommended life limits. These recommendations formed the basis of Australia's fatigue policy. The first formal published rule was Air Navigation Order 101.22.6.17, issued 1 July 1967 (two years before FAR 23.572):

6. I 7 - Fatigue Strength

The design of the aeroplane shall be such as to ensure that the possibility of disastrous fatigue failure of the primary structure is extremely remote under the action of the repeated loads of variable magnitude expected in service.

This had a very clear statement of the objective - high structural reliability - but concentrated on design to the exclusion of maintenance. This problem was fixed by the next amendment, published as AN0 10 1.22.6.62 on 27 May 1974:

6.61 *- Fatigue*

(1) A fatigue strength substantiation shall be provided justifying the adequacy of the primary structure either on a safe life or on a fail safe basis.

(2) To establish a safe fatigue life an operational life shall be determined, in a manner acceptable to the Director-General, during which the possibility offatigue failure of any principal structural element under the repeated loads to be expected in service is extremely remote.

(3) To establish fail safe characteristics, it shall be determined to the satisfaction of the Director-General that the primary structure will remain capable of safely supporting critical design limit loads and also the ***repeated loads*** *to be expected in service during the period following the complete or partial fatigue failure of any principal structural element until detection of the damage by proposed inspection procedures.*

These words, which have remained unchanged for 25 years, still clearly express the requirement. Even the 'failsafe option' is really modern damage tolerance!

Grandfathering was a particular concern to Australia because grandfathered aircraft were expected to be widely used to carry fare-paying passengers. Australia could not risk doing nothing to control fatigue. So, in 1970, a delegation from the Australian Department of Civil Aviation[0] visited the United States. The delegation told the FAA and the light aircraft manufacturers that Australia would henceforth enforce its fatigue rule for all 'new' aircraft, including derivatives. The American light aircraft manufacturers cooperated to develop Australia's life limits. The United Kingdom Civil Aviation Authority adopted a similar stance.

PREDICTION AND PRACTICE

For more than fifteen years, the author has had access to:

- the manufacturers' fatigue evaluations for every aircraft type on the Australian register; and
- their Australian and international defect and accident history.

How well has practice matched prediction? For some aircraft types, it is too early to tell. For many, they match quite well. But for some, the cracking in service:

- occurs at a different location;
- is caused by different loads; or

[0] CASA's predecessor.

- starts much earlier than predicted (even with scatter factors of 4 to S).

Some examples:‘- . .

Aircraft	Part	Prediction	Practice
Ayres Thrush	Wing main spar	Analysis only.	Service shows that the wing spar cracks in several places, but most dangerously in the steel spar cap, a location not predicted by the analysis.
Cessna singles	Wing lift strut	Analysis and test predicted a virtually unlimited fatigue life.	Service shows widespread cracking where the lower end of the strut attaches to the fuselage.
Cessna 402B	Wing spar	Analysis and test predicted cracking at the junction with the auxiliary spar	Service shows that the wing spar cracks near the engine beams from the interaction of engine and wing loads.
Commander 112/114	Wing main spar	Analysis predicted cracking at the wing root from wing bending.	Service shows that the wing spar cracks in a different place in one tenth the time predicted by the analysis because of loads from the landing gear actuator.
GAF Nomad	Tailplane spar	Analysis only.	Service shows that the tailplane spar cracks earlier than predicted by analysis. Wake turbulence during ground running is far more damaging than expected.
Partenavia P68	Wing spar	Analysis only.	Service shows that the wing spar cracks in one tenth of the time predicted by the analysis because of an extraordinarily fatigue-prone design detail.
Piper PA-3 1	Wing centre-line splice plate	A simple resonance test of the spar assembly predicted cracking at the wing root.	Service shows that the splice plate cracks first, at half the time predicted for the wing root. Was the test unable to replicate the fretting which contributed to the splice plate cracking in service because it omitted the surrounding fuselage structure? Or *did* the test replicate the fretting and the splice plate *did* crack, but the cracks were so small and tight that the testers missed them?

[0] No criticism is implied of any of the companies. There are other examples that could not be given because they involve proprietary information that is not so public. Fatigue is hard to predict for even the most skilled and conscientious.

Considering all the evidence (not just these examples), there are some lessons for us:

- Consider *all* conceivable loads and locations. Regulators should review this critical first step with the designer before they start into the details of an analysis or test. They should look at production-standard structure, as well as drawings and stress reports. Regulatory fatigue specialists can assist designers because they get to see far more examples of design features and their fatigue consequences - both good *and* bad.
- Consider as much as possible of the structure *surrounding* critical joints.
- For wings, be very wary of omitting engine and landing gear loads.
- Tests are far more reliable than analyses.
- The standard scatter factors are not over-conservative as some argue. Sometimes they are not large enough. Uncertainty and variability are considerable, whether one uses fracture mechanics or Miner's rule.

SAFETY AND SERENDIPITY

Sometimes we are lucky and a prediction is right for the wrong reason. Even though Cessna and Ayres failed to predict the right fatigue-critical *location* on the wing spars of the 402B and the S2R respectively, fortuitously they got the *timing* right.

If the spar is lifed, timing is all that matters and safety is not compromised. In this way, Australia avoided the potentially catastrophic Cessna 402B spar failures which occurred in the United States.

If instead the spar is inspected, but in the wrong location, we get tragedies such as the wing separations which happened to the Ayres S2R.

Serendipity is an often unappreciated advantage 'safe life' has over 'damage tolerance'. Brot (1997) describes retirement as 'a "secret weapon" in our battle for structural integrity'.' We should use the 'secret weapon' more often than we do.

SAFETY AND COST

Firstly, are fatigue rules cost-effective for society as a whole?

Air crashes cost dollars as well as grief. Each air fatality has been estimated to cost US$2.7 million in the United States,' 0.8 million pounds sterling in the United Kingdom,' and A$1.5 million in Australia.' Since wing separation is fatal, it is not surprising that cost-benefit analyses consistently show fatigue rules to be good value.

[0] Comparatively, safety-by-retirement is even better than Brot suggests, His probabilistic analysis of damage tolerance overlooks the chance that cracking will occur somewhere unexpected. This happens. In April, the wing of a ten-passenger twin broke at a location not predicted by a brand new damage tolerance evaluation. The evaluation, which used the most modem methods of analysis, had the benefit of a fatigue test and more than twenty years of service experience to draw on.

[0] The United States government's Value of a Statistical Life (VSL), 1999.

[0] DETR 1998

[1] BTE 1998

Secondly, are fatigue rules cost-effective for the aircraft companies?

For simple aircraft, with conventional structure, the cost of a fatigue analysis is minimal. Simple methods, such as AFS- 120-73-2 (FAA 1973), have been available since FAR 23.572 took effect. Now, computerised, they take hours rather than days.

Even the most basic fatigue evaluation forces the designer to consider fatigue: keep stresses low; choose fatigue-resistant materials; avoid sharp changes in section and improve inspectability. Such features improve durability, which in turn improves saleability and minimises warranty claims and litigation.

Fatigue tests of components, sub-assemblies and complete aircraft get progressively more expensive but give progressively more reliable predictions.

Some companies *voluntarily* perform elaborate fatigue tests. They view them as a valuable design tool, not a regulatory nuisance. Some test twice: first to identify design flaws; second to prove and certify the corrections. Cessna's first test of the Citation III business jet showed up design flaws in details that cracked very early. Cessna was able to fix them before production started. In this way, a fatigue test can pay for itself.

The fatigue test of the Beech King Air 300 paid off for Raytheon. It exposed that a new design of wing attachment fitting had a very short fatigue life because of fretting. Finding the fault early enough limited field modification to the first few aircraft.

Thirdly, are fatigue rules cost-effective for the aircraft operators?

Many operators don't think so. They exploit the uncertainty. Often in ignorance, they challenge the present ability to predict an aircraft's fatigue behaviour and use that as a means of questioning the effectiveness of the control measures. They question that effectiveness because they don't like the costs. The challenging and questioning continues until cracks occur in service, and frequently well after.

The truth is that an aircraft type that has complied with the fatigue rules should need fewer unexpected major repairs. Moreover, it should be less likely to suffer a catastrophic structural failure - which would be very bad for business.

Another issue for operators is their perception of the relative costs of life limits versus inspections. They hope that inspections won't find anything and they won't have to pay. Many seem more willing to pay for continual inspections that are more costly than one time replacements. Most ignore the extraordinary costs of the *unanticipated* replacement of cracked parts found by inspection (Emmerson 1992).

BENEFITS AND RISKS

While a fatigue evaluation has undoubted benefits, could there be potential risks? Possibly, by engendering unwarranted confidence. If there is a life limit, is there a risk that mechanics won't inspect *anywhere*? If there is an inspection program, is there a risk that they won't inspect *elsewhere?* Regulators have the tricky job of fostering respect for approved fatigue control measures without discouraging the healthy suspicion and curiosity of the good mechanic.

MINER AND PARIS

There has been a lot of argument about the relative merits of Miner (cumulative damage) and Paris (fracture mechanics from an initial flaw) when setting life limits or inspection thresholds. This is a 'gnat' because both methods, when calibrated, give the same answer.

The 'camel' is that both methods get the same *wrong* answer if not enough effort and testing are devoted to getting the loads and the locations right.

ANALYSIS AND TEST

These days, every aircraft designer thinks *their* aircraft will never fatigue because of good design and low stresses. But which stresses? The troublesome loads are not always the ones the part was designed to carry. More often they are incidental loads, which are harder to foresee. This is one reason for analytical errors.

Tests are far more reliable, which is why FAR 23.572, has increasingly insisted on them. A problem, however, has been lenient interpretation of the exemption provisions.

Lower scatter factors are another incentive to test.

WINGS AND THE REST

Although fatigue usually strikes the highly stressed wing spar first, eventually the whole airframe succumbs. According to the rules, fatigue should be evaluated for the whole airframe right from the start. But Australia has tried to ease the initial design burden by allowing fatigue evaluation of most of the airframe to be deferred for up to two spar life-times. So what might seem an impenetrable barrier is merely a trigger for further evaluation.

SAFE LIFE AND DAMAGE TOLERANCE

Safe life and damage tolerance are two common regulatory terms. But are they clear and unambiguous? Think how people commonly differentiate the two:

Area	**Safe Life**	**Damage Tolerance**	**Comments**
By design	Single load path	Multiple load paths Low Stresses Tough materials	However, single load paths can be analysed by fracture mechanics and can often become damage tolerant.
By method of analysis	Miner's cumulative damage using S-N curves	Fracture mechanics from initial flaws	Both methods, when calibrated, give the same answer.
By allowance for manufacturing flaws and accidental damage	None	Yes	This is what the proponents of damage tolerance say. But safe life *does* make allowance in the scatter factor and directed NDI can easily *miss* cracks if they don't occur where expected - which is usually the case for manufacturing flaws and accidental damage.
By maintenance control action	Retire parts at fixed time	Inspect. Start at threshold and repeat at regular intervals.	The only dichotomy and the only useful differentiator.

Only one is useful. Does it matter? It does for clear thinking on a complex subject. Think of the confusion surrounding the damage tolerance of composites. Yes, they can tolerate damage. But how safely and for how long?

The question is npt just academic. FAR 23.573(a)(2) allows a 'no-growth' method for certificating composite structure: By this method, the composite is damaged then tested. If the damage *doesn't* grow, the composite is considered 'damage tolerant'. But for how long? Only as long as the test. No one knows whether or not longer service *could* cause damage growth. No one knows whether or not such growth *could* escape detection and become dangerous. Yet regulators don't limit the life of the composite in service. Has the terminology duped them?

Since it is the maintenance control action that is the useful differentiator, Emmerson and others prefer to use the terms 'safety-by-retirement' and 'safety-by-inspection' (Emmerson 1992 - a very good paper on airworthiness control methods).

Using the clearer terminology, which is better?

Consideration	**Safety-by-Retirement**	**Safety-By-Inspection**
Structure that is inaccessible, highly stressed or has low fracture toughness.	Often the only safe option.	Impractical.
Lack of certainty about failure modes and locations.	Can still be safe if the timing is right for the wrong reason. Today's techniques are such that this is an intended outcome.	A serious problem for designs that require highly directed NDI - while inspecting one hole with eddy currents, cracks could be growing in the next. Really need area inspections. The whole history, from the Lusaka Boeing 707 to the Goldsby Cessna 402C, shows that you can't second guess the structure.
Manufacturing flaws and accidental damage.	Allowed for in the scatter factor.	As above.
Statistical variability in the load history and in the fatigue properties of nominally identical structure.	Both must include more than just specifying a life limit or an inspection interval. Must include monitoring usage - not necessarily in minute detail.	
Maintenance skill required	Both part replacement and NDI require skills of a high order.	
Certification cost	Safety-by-inspection costs at least three times as much as safety-by-retirement.	
Operator cost	Similar in the long run if the same operator for the life of the aircraft. Not a simple issue (Emmerson 1990).	

One is not better than the other. There is a place for both, especially for light aircraft. It is not clear then why safety-by-retirement is no longer fashionable and FAA's new NPRM proposes safety-by-inspection exclusively.'

Safety-by-inspection is only safe if we know *where* to look, *when* to look and *how* to look - and there must be *time* to look. If we don't confidently know *all* these things, the only safe option is safety-by-retirement (see Swift 1992). Overemphasising safety-by-inspection could force

[0] Except during the early transition period.

designers to try to inspect the uninspectable.

In the United States, many light aircraft have neither. *Something* is better than *nothing*.

METALS AND COMPOSITES

Predicting the fatigue behaviour of composites is an order of magnitude harder than for metals. Degradation mechanisms are more complex, harder to analyse, harder to test, and there are more of them. Failures caused by secondary stresses weaken the material's resistance much more to primary stresses. The fatigue resistance of nominally identical components is much more variable.

If prediction is difficult for metals, it is even more so for composites. It is not surprising then that composite light aircraft are turning up unexpected structural problems in service.

PRIVATE AND COMMERCIAL

The FAA seems in two minds about controlling fatigue in private aircraft. On the one hand it has been toughening its design rules, which apply whether the aircraft type will fly privately or commercially. On the other hand, so far it is only applying its operational fatigue rules to aircraft that fly commercially.

Regulators often allow private fliers to accept higher risks than those who pay fares. For example, private fliers don't have to carry as much communication, navigation and emergency equipment in their aircraft. The extra risk is constant:

But is such a policy appropriate for wear-out phenomena such as fatigue, where the extra risk is not constant, but diverges? If you do nothing, the risk eventually becomes a certainty.

There are also practical questions. What duty does the regulator have to warn the public? Should aircraft that have not had a fatigue evaluation carry warnings, as do packets of cigarettes? Do private flyers need education about the risks, to help them make informed choices? #at would happen if the same aircraft type were to have mandatory life limits or inspections in one role but not in another? What would happen to aircraft that regularly swap between private and commercial operations? What would be the attitude of the market and aircraft insurers?

This is not a simple issue and warrants more debate.

WEIGHTS AND FLIGHT TIMES

The progress of fatigue is highly sensitive to many variables. For example, the time to crack nucleation and the rate of subsequent crack growth are both exponential functions of stress. Therefore small changes in 'stress per g' produce very large changes in life limits and inspection thresholds and intervals. How then does one account for loading differences between individual aircraft?

Similarly, the 'ground-air-ground cycle' can cause half the fatigue damage. How then does one account for flight time differences between individual aircraft?

The 'simple' solution would be to monitor 'g' or strain in every aircraft. Unfortunately this still seems too expensive.

In Australia and the UK, operators must tell the regulator if their aircraft fly unusually heavy or short flights. The regulator can then arrange for a special determination. As regulatory resources get scarcer and fewer in industry are able to make these special determinations, CASA is beginning to find this solution unmanageable. If CASA is forced to abandon this level of refinement, the result will be a lowering of safety standards.

MODIFICATIONS AND WORK AROUNDS

An increasing number of performance-enhancing modifications are appearing on the market. Two that are popular are winglets and vortex generators. Winglets increase the stress per g in the wing by increasing the proportion of lift generated by the tips. Vortex generators increase the stress per g by increasing the allowable take off weight.

The problem is that their designers rarely consider their effect on the basic aircraft's life limits and inspections. For example, winglets can halve times to crack nucleation and double crack growth rates. Australia is one of few countries trying to control this problem, but the increasing number of fatigue-affecting modifications is making regulation difficult. Again, if CASA is forced to abandon this level of refinement, the result will be a lowering of safety standards.

It is important that the FAA's new NPRM addresses this problem as it proposes.

NATIONAL DIFFERENCES AND HARMONISATION

A frustration for both regulators and operators has been national differences in the regulation of light aircraft fatigue. Why does an aircraft have a life limit in one country, but not in another? Why does fitting winglets halve the wing life in one country, but not in another? Why does fitting a spar reinforcement eliminate the wing life in one country, but not in another? Why does an Airworthiness Directive require urgent inspections in one country, but not in another? If the world's fatigue rules are all the same, where do the differences come from?

Firstly, certification policy. Some countries more liberally grandfather than others. Therefore some countries control fatigue for aircraft which others don't.

Secondly, there are differences in enforcement. The Australian wing life for the Bandeirante is shorter than elsewhere because CASA rightly only counted the time until the test wing lost ultimate strength, not until it finally broke under a much lesser load.

Regulators of light aircraft fatigue should cooperate more, as do their heavy aircraft counterparts. The aim should be standardisation of world's best practice, not the most lenient which some industry groups lobby for. The FAA's new NPRM is a good opportunity to start.

DESIGN AND OPERATIONAL RULES

The FAA is already regulating fatigue more by its operational rules than its design rules. Transferring the regulation of fatigue from the design rules to the operational rules would offer these advantages:

- It would better match safety to usage (aircraft other than transport category end up in transport operations).
- It would allow entry into service without having to wait for lengthy tests.
- Manufacturers could defer fatigue compliance to spread their certification costs.

ENDS AND MEANS

Fatigue and damage tolerance evaluations are only a means to an end. The 'end' is a maintenance program that should assure high structural reliability.

Means of compliance have cluttered FARs 23.571 to 23.575. One danger is losing sight of the principles, as has happened with composites. Another is the stifling of creative safety solutions. The penalty for industry is less flexibility.

Accordingly, the author would like to propose the following alternative for discussion:

The aircraft's primary structure shall be:

- *designed;*
- *manufactured; and*
- *have instructions for its:*
  - *operation;*
  - *maintenance; and*
  - *repair,*

such that the risk of structuralfailure will be extremely remote.

The following shall be accounted for rationally:

- *uncertainty in the-demonstration of compliance;*
- *variability in crack nucleation, crack propagation and cracked strength of nominally identical aircraft;*
- *variability in the loads expected in service for the full rang? of permitted operations; and*
- *the probability of detection if inspections are prescribed.*

This is the requirement. Acceptable means of compliance should be published separately.

TYPE CERTIFICATION AND CONTINUING AIRWORTHINESS

The two are complementary because an important purpose of type certification is to lay the foundations for maintaining continuing airworthiness. Even the most thorough type certification will not prevent surprises in service, but, a good product can be quickly and safely recovered.

Historically, CASA has put a lot of effort into continuing airworthiness. It has had to, even more so than the Americans and Europeans whose practices have not always suited Australia's circumstances. Continuing airworthiness is very different when you are half the world away from the manufacturer.

Torkington and Emmerson (1991) put it this way:

> *Structural fatigue continues to be a particular source of difficulty in general aviation aircraft. About 75% of that fleet in Australia is over fifteen years old. Some are more than 50 years old. Limited mailable data suggest that aircraft in the Australian charter operators 'fleets have seen considerably more service than those in the corresponding American fleets for example.*
>
> *It is essential that Australia's commercial general aviation aircraft, a valuable resource, are durable and reliable. With a diverse operating industry remote from the centres of manufacture, the CAA must insist on compliance with current standardsand on retaining the expertise and authority to ensure continuing airworthiness.*

In-country control of continuing airworthiness has been indispensable for Australia and has been the cornerstone of Australian air safety for the last fifty years.

World-wide, safety regulators are now trying to shift the onus for continuing airworthiness to the type certificate holder. But there are difficulties:

- Since many of Australia's aircraft were built outside Australia, Australia would be dependent on other regulators regulating the type certificate holder's ability to monitor and investigate structural defects. Few do this effectively now.
- While one would expect the type certificate holder to be familiar with the aircraft and hold the necessary design data, this is not always true. Some were not the original designers. Some are technically moribund. Some have lost the design data.'
- Few keep the records and few have the procedures to systematically investigate service problems. On the whole, the author is very disappointed with what he has seen so far of fatigue investigations by light aircraft manufacturers. In some companies, the only time Product Support and Engineering see each other is in the lunchroom or *after* a crash.
- Type certificate holders face commercial pressures which inhibit honest and thorough investigation. Regulators will have to maintain a very close watch.

Therefore regulators should be wary of a completely 'hands off approach. When the chips are down, one cannot rely on the type certificate holder. This is a matter of recorded fact. They are bound to act out of self-interest. No matter what the company's design engineer may think, its lawyers, accountants and salesmen all get in the way of cooperation for safety.

[0] The author is aware of two such cases in recent years.

However, while not relinquishing their autonomy, regulators could shift some of the burden. ICAO's Continuing Airworthiness Manual has some advice (Section VI-l-4):

> 3.2.3 *It is worthwhile for the organisation responsible for the type design[0] to systematically and periodically review and analyse service data obtained from all operators. Summarised data should be reported to the State of Design. Use should be made of appropriate statistical methods and comparison of service data with predictions made for type certification.*
>
> *3.2, I One State uses and recommends a time interval between such reports of 20 to 25 per cent of design life goal, or three to five years of service. This aspect may be controlled by the State of Design specifically for each case.*

While trend reviews and other ways of looking at past defects are no way to see what will happen in the future, such stocktakes are an opportunity for the type certificate holder and the regulator to agree on a strategy that will. It is an opportunity for the regulator to assess the company's ability to control the continuing airworthiness of their types.

Inability would warrant a threat to cancel the affected type certificates. While some type certificate holders would be pleased to rid themselves of responsibility for some models, owners might want to fund technical support elsewhere rather than lose the use of their aircraft.

TRENDS AND ISOLATED OCCURRENCES

It has always been important to quickly, rationally and systematically respond to the unexpected consequences of fatigue which inevitably arise in service. When we don't, we get 'multiple tombstones' as happened with the Aero Commander, Beech 18, Piper PA-25, Ayres S2R and others.

These are just two of the traps we fall into:

The Isolated Occurrence Syndrome

Every fleet-wide problem starts with the first report. One should assume every problem could be fleet-wide unless there is sound evidence of a peculiarity. It is too easy to dismiss a reported fatigue crack as a 'one-off if it is caused by a manufacturing flaw, a bad modification, a corrosion pit or an unusual aircraft role. All are solid indications of the reality of structural fatigue which will eventually affect all aircraft.

Trend Monitoring

Service difficulty reports are increasingly the domain of statisticians more than engineers. This is a worrying development for fatigue. While it *can* be useful to monitor reliability trends of vacuum pumps or airspeed indicators, such an approach is clearly unsafe for wing fatigue. The accident record amply demonstrates this. Sometimes we only get one warning. If we don't make good use of it, the next report comes from the crash site.

There are advisory circulars for fatigue compliance at type certification, but little for defect investigation afterwards. A good paper on the principles and the process is A *Measured Response to Structural Defects* (Emmerson 1995). Two analytical tools applicable to light aircraft are the *Maximum Likelihood Method* (Emmerson 1976) and Walker's *Trend Analysis* (199 1). The author would be pleased to hear of others.

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[0] ICAO-speak for the type certificate holder.

LOOKING FORWARD AND LOOKING BACK

FAR 23.572 and the like have undoubtedly improved safety. Wings and things don't break so often. But still they do. There is room for improvement.

As this report has shown, there are still 'gnats' and 'camels'. There should be:

- Less 'grandfathering'.
- More continuing airworthiness, not just type certification.
- Less prescriptive rules, more wisely enforced.
- Less nit-picking, more attention to locations and loads.
- Less analysis, more testing.
- More safety-by-retirement, less trying to inspect the uninspectable..
- More responsibility to go with the privileges of holding a type certificate.
- More defect investigation, not just trend monitoring.
- For composites, less talk of 'damage tolerance', more of inspectability.

It would improve our perspective of what are 'gnats' and what are 'camels' if we more often revisited old predictions. ICAF papers herald technological advances. But how can we assess their effectiveness? Only by reviewing the service experience in ten, twenty or thirty years. The same is true of regulatory policy.

Perhaps a theme for a future ICAF meeting could be:

Looking forward by looking back.

It might be humbling, but it would certainly be worthwhile.

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THE HISTORY OF FAR 23.572

Amendment 23-7, Effective 14 September 1969

The FAA had a lot of opposition to its first issue of the rule. Its own words are telling:

> *Numerous comments were received objecting to this proposal. In this connection, it was stated tha (1) the current strength requirements and design practices are conservative and adequate to prevent serious fatigue problems; (2) corrosion, prior abuse, and special purpose operations cause fatigue failure, rather than lack of design fatigue strength; (3) fatigue substantiation will not eliminate cracks, and a better maintenance program would be more effective; and (4) sufficient data is not available to establish load spectra for fatigue substantiation. The FAA does not agree. 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 independent, a higher failure rate among older airplanes is to be expected; however, fatigue problems have arisen in airplanes certificated in recent years. Neither fatigue substantidion nor better maintenance programs will eliminate all cracks. The purpose of the proposed rule is to prevent catastrophic failures. Furthermore, Part 23 airlanes are not designed on a redundant-structure, fail-safe basis for which maintenance alone wouldbe sufficient. Both fatigue substantiation and a good maintenance program are needed. Reasonable load spectra* can and *have been established from the extensive gust data which is available, and reasonable and acceptable methods of compliance have been established and widely publicised. However, while it has been determined that fatigue substantiation is necessary, the FAA'* **does** *not agree with one comment which suggested that the proposed requirement should be expanded-to a full limit loadfail-safe requiremat with a design objective of at least 10,000 hours and a test life of 30,000 flights. Such a requirement would be more severe than that requiredfor transport category airlanes and cannot be justified on the basis of service experience. The amendment is adopted as proposed.. .*

23.572 Wing and associated structure

The strength, detail design, and fabrication of those parts of the wing, wing carrythrough, and attaching structure whose failure would be catastrophic must be evaluated under either of the following unles it is shown that the structure, operating stress level, materials, and expected use are comparable, from a fatigue standpoint, to a similar design that has had extensive satisfactory service experience:

A fatigue strength investigation, in which the stucture is shown by analysis, tests, or both, to be able to withstand the repeated loads of variable magnitude expected in service.

A fail safe strength investigation in which it is shown by analysis, tests or both, that catastrophic failure of the structure is not probable after fatigue failure, or obvious partial failure, of a principal structural element, and that the remaining structure is able to withstand a static ultimate load of 75 percent of the critical limit load factor at Vc. These loads must be multiplied by a factor of 1.15 unless the dynamic effects of failure under static load are otherwise considered.

Amendment 23-14. Effective 20 December 1973

In this amendment, FAA added the sentence, *'Analysis alone is acceptable only when it is conservative and applied to simple structures'*, to 'make it clear that, for a fatigue evaluation, the use of analysis alone is acceptable only under certain specified circumstances'.

Amendment 23-34. Effective 17 Februarv 1987

In this amendment, FAA added Commuter Category to FAR 23. Commuter Category caters for those aircraft that formerly straddled the boundary between FAR 23 and FAR 25. FAA had the opportunity to require these aircraft to meet the standards of FAR 25 and be damage tolerant. Instead it extended the choice of safe life, fail safe or damage tolerance to aircraft as heavy as 19,000 pounds carrying as many as 19 passengers.

In replying to a specific recommendation for damage tolerance, the FAA said:

> *The FAA recognises the merit of a damage tolerant design; however, the service experience with airplanes recertificated to SFAR No. 41 with their corresponding high utilisation does not support the need for a mandatory damage tolerant design philosophy for commuter category airplanes.*

However, by 1999, the FAA had changed its mind. Its NPRM of 2 April proposes requiring 'damage tolerance-based inspections' exclusively, even for aircraft that have previously been evaluated under the safe life and fail safe provisions of FAR 23.

Amendment 23-38. Effective 26 October 1989

In this amendment, FAA extended the scope of the rule to require fatigue evaluation of empennages, canards, tandem wings and winglets.

Amendment 23-45, Effective 7 September 1993

In this amendment, FAA extended the scope of the rule by adding a new § 23.573 specifically to require fatigue evaluation of composite structure (even though composite structure had never been excluded from 23.572).

Amendment 23-48, Effective 11 March 1996

In this amendment, FAA rearranged its fatigue rules to harmonise with the JAA. The only substantive change was the addition of § *23.575, Inspections and Other Procedures,* to explicitly require airplane manufacturers to publish the maintenance that flows from their fatigue evaluations. ANNEXE 2 is the current version (as at June 1999) of the FAA's light aircraft design rules, including pressure cabins.

ANNEXE 2

FAA DESIGN RULES FOR LIGHT AIRCRAFT FATIGUE (@ June 1999)

§ 23.571 Metallic pressurized cabin structures.

For normal, utility, and aaobatic category airplanes, the strength, detail design, and fabrication of the metallic structure of the pressure cabin must be evaluated under one of the following:

(a) A fatigue strength investigation in which the structure is shown by tests, or by analysis supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected in service; or

(b) A fail safe strength investigation, in which it is shown by analysis, tests, or both that catastrophic failure of the structure is not probable after fatigue failure, or obvious partial failure, of a principal structural element, and that the remaining structures are able to withstand a static ultimate load factor of 75 percent of the limit load factor at Vc, considering the combinedeffects of normal operating pressures, expected external aerodynamic pressures, and flight loads. These loads must be multiplied by a factor of 1.15 unless the dynamic effect of failure under static load are otherwise considered.

(c) The damage tolerance evaluation of § 23.573(b).

§ 23.572 Metallic wing, empennage, and associated structures.

(a) For normal, utility, and acrobatic category airplanes, the strength, detail design, and fabrication of those parts of the airframe structure whose failure would be catastrophic must be evaluated under one of the following unless it is shown that the structure, operating stress level, materials and expected uses are comparable, from a fatigue standpoint, to a similar design that has had extensive satisfactory service experience:

(1) A fatigue strength investigation in which the structure is shown by tests, or by analysis supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected in service; or

(2)A fail safe strength investigation in which it is shown by analysis, tests, or both, that catastrophic failure of the structure is not probable after fatigue failure, or obvious partial failure, of a principal structural element, and that the remaining structure is able to withstand a static ultimate load factor of 75 percent of the critical limit load at Vc. These loads must be multiplied by a factor of 1.15 unless the dynamic effects of failure under static load are otherwise considered.

(3) The damage tolerance evaluation of § 23.573(b).

(b) Each evaluation required by this section must-

(1) Include typical loading spectra (e.g., taxi, ground-air- ground cycles, maneuver, gust);

(2) Account for any significant effects due to the mutual influence of aerodynamic surfaces; and

(3) Consider any significant effects from propeller slipstream loading, and buffet from vortex impingements.

§ 23.573 Damage tolerance and fatigue evaluation of structure.

a) Composite airframe structure. Composite airframe structure must be evaluated under this paragraph instead of §§ 23.57 1 and 23.572. The applicant must evaluate the composite airframe structure, the failure of which would result in catastrophic loss of the airplane, in each wing (including canards, tandem wings, and winglets), empennage, their carrythrough and attaching structure, moveable control surfaces and their attaching structure, fuselage, and pressure cabin using the damage- tolerance criteria prescribed in paragraphs (a)(1) through (a)(4) of this section unless shown to be impractical. If the applicant establishes that damage-tolerance criteria is impractical for particular structure, the structure must be evaluated in accordance with paragraphs (a)(1) and (a)(6) of this section. Where bonded joints are used, the structure must also be evaluated in accordance with paragraph (a)(5) of this section. The effects of material variability and environmental conditions on the strength and durability properties of the composite material must be accounted for in the evaluations required by this section.

(1) It must be demonstrated by tests, or by analysis supported by tests, that the structure is capable of carrying ultimate load with damage up to the threshold of detectability considering the inspection procedures employed.

(2 The growth rate or no-growth of damage that may occur from fatigue, corrosion, manufacturing flaws or impact damage, under repeated loads expected in service, must be established by tests or analysis supported by tests.

(3) The structure must be shown by residual strength tests, or analysis supported by residual strength tests, to be able to withstand critical limit flight loads, considered as ultimate loads, with the extent of detectable damage consistent with the results of the damage tolerance evaluations. For pressurized cabins, the following loads must be withstood:

(i) Critical limit flight loads with the combined effects of normal operation pressure and expected external aerodynamic pressures.

(ii) The expected external aerodynamic pressures in lg flight combined with a cabin differential pressure equal to 1.1 times the normal operating differential pressure without any other load.

(4) The damage growth, between initial detectability and the value selected for residual strength demonstration, factored to obtain inspection intervals, must allow development of an inspection program suitable for application by operation and maintenance personnel.

(5) For any bonded joint, the failure of which would result in catastrophic loss of the airplane, the limit load capacity must be substantiated by one of the following methods-

(i) The maximum disbonds of each bonded joint consistent with the capability to withstand the loads in paragraph (a)(3) of this section must be determined by analysis, tests, or both. Disbonds of each bonded joint greater than this must be prevented by design features: or

(ii) Proof testing must be conducted on each production article that will apply the critical limit design load to each critical bonded joint; or

(iii) Repeatable and reliable non-destructive inspection techniques must be established that ensure the strength of each joint.

(6) Structural components for which the damage tolerance method is shown to be impractical must be shown by component fatigue tests, or analysis supported by tests, to be able to withstand the repeated loads of variable magnitude expected in service. Sufficient component, subcomponent, element, or coupon tests must be done to establish the fatigue scatter factor and the environmental effects. Damage up to the threshold of detectability and ultimate load residual strength capability must be considered in the demonstration.

(b) Metallic airframe structure. If the applicant elects to use §23.571(a)(3) or § 23.572(a)(3), then the damage tolerance evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. The determination must be by analysis supported by test evidence and, if available, service experience. Damage at multiple sites due to fatigue must be included where the design is such that this type of damage can be expected to occur. The evaluation must incorporate repeated load and static analyses supported by test evidence. The extent of damage for residual strength evaluation at any time within the operational life of the airplane must be consistent with the initial detectability andsubsequent growth under repeated loads. The residual strength evaluation must show that the remaining structure is able to withstand critical limit flight loads, considered as ultimate, with the extent of detectable damage consistent with the results of the damage tolerance evaluations. For pressurising cabins, the following load must be withstood:

(1) The normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditims specified in this part, and

(2) The expected external aerodynamic pressures in 1 g flight combined with a cabin differential pressure equal to 1.1 times the normal operating differential pressure without any other load.

(c) Removed

§ 23.574 Metallic damage tolerance and fatigue evaluation of commuter category airplanes.

For commuter category airplanes-

(a) Metallic damage tolerance. An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion defects, or damage will be avoided throughout the operational life of the airplane. This evaluation must be conducted in accordance with the provisions of § 23.573, except as specified in paragraph (b) of this section, for each part of the structure thatcould contribute to a catastrophic failure.

(b) Fatigue (safe-life) evaluation. Compliance with the damage tolerance requirements of paragraph (a) of this section is not required if the applicant establishes that the application of those requirements is impractical for a particular structure. This structure -must be shown, by analysis supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks. Appropriate safe-lifescatter factors must be applied.

§ 23.575 Inspections and other procedures.

Each inspection or other procedure, based on an evaluation required by §§ 23.571, 23.572, 23.573 or 23.574, must be established to prevent catastrophic failure and must be included in the Limitations Section of the Instructions for Continued Airworthiness required by § 23.1529.

ANNEXE 3

'LIGHT-' AIRCRAFT CRASHES CAUSED BY STRUCTURAL FATIGUE

1945	Stinson A2 W	Australia
1947	Beech D18	West Virginia
1952	De Havilland Dove	Australia
1953	Bristol 170	Australia
1955	Bristol 170	Nigeria
1957	Bristol 170	New Zealand
1957	Scottish Aviation Twin Pioneer	New Guinea
1957	Scottish Aviation Twin Pioneer	Libya
1958	Noorduyn Norseman	Canada
1960	Curtiss C46 Commando	Utah
1960	Beech D 18	Texas
1961	Aero Commander 680	New Zealand
1963	Noorduyn Norseman	Canada
1963	De Havilland DHC-2 Beaver	Australia
1964	De Havilland DHC-2 Beaver	Australia
1964	Beech G18	New Mexico
1964	Aero Commander 680E	Canada
1965	Handley Page Herald	Canada
1965	Handley Page Herald	Syria
1966	Beech Cl8	Iowa
1967	Beech D18	?
1967	Beech E18	?
1967	Aero Commander 560E	Texas
1968	Airspeed Ambassador	London
1971	De Havilland Dove	Arizona
1971	Cessna 206B	Alaska
1972	Beech E 18	Ohio
1973	Beech E18	Indiana
1973	Beech E18 (C45H)	Manitoba
1979	Grumman Goose	California
1979	Beech King Air 90	Canada
1981	Beech King Air E90	Texas
1987	Piper PA-28	Texas
1990	GAF Nomad	Australia
1990	Aero Commander 680E	Sweden
1991	Ayres S2R	South Africa
1991	Piper PA-25-235	Alabama
1993	Piper PA-25- 150	Iowa
1993	Ayres S2R	Holland
1997	Ayres S2R	Arkansas
1999	Beech T-34A	Georgia
1999	Beech T-34A	Venezuala
1999	Cessna 402C	Oklahoma

The author would be interested to hear of others that should be on this list.

Some got very close:

- One Beech 99 and five Cessna 402Bs which suffered complete main spar failures in flight, yet made it home ‘on a wing and a prayer’.

One Beech Queen Air, one Partenavia P68 and one Aero Commander which had almost-severed spar caps. The Beech Queenair and the Aero Commander would not have survived to the next 100-hourly inspection.