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Static and Fatigue Test Loading Development for an F-111 Bonded Composite Repair Substantiation

K. Walker and G. Swanton

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Airframes and Engines Division
Aeronautical and Maritime Research Laboratory

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ABSTRACT

Substantiation of a bonded composite repair to an F-111 lower wing skin required the development of representative loads for specimen testing. This report describes the development of representative static and fatigue spectrum loads to be applied during the testing. The spectrum was based on a known representative Wing Pivot Bending Moment spectrum. Full scale static wing testing and the manufacturer's original stress analysis reports were used to convert this spectrum to be representative of the stress at the region of interest in the outboard section of the lower wing skin. The spectrum (representing 199 flights or 499.1 flying hours) was expressed in both blocked and cycle by cycle form. Truncation was performed to reduce the total number of cycles to a manageable level and this was evaluated both analytically (for the blocked spectrum) and experimentally (for the cycle by cycle spectrum) to ensure that damaging cycles were retained. The blocked spectrum was found to give highly conservative results and this approach is recommended only where no further information about the cycle history is known. The implications of the present results for realistic spectrum truncation are briefly discussed.

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Executive Summary

Following the discovery of a crack in the outboard section of a RAAF F-111C wing in 1994, a bonded composite repair scheme was devised by RAAF and DSTO personnel. The repair was challenging and critical because it was applied to primary structure and the damage had reduced the residual strength of the wing to below Design Limit Load. A comprehensive validation and substantiation program was required to assure the airworthiness of the repair. The substantiation included testing of representative specimens. This report details the derivation of the static and fatigue loads that were applied during the testing.

Strain gauge sensors were installed in the repair region on an uncracked, unpatched test wing which was then subjected to loading equivalent to that experienced during Cold Proof Load Testing. The results from this test were used to calibrate a detailed three dimensional finite element model of the repair region. A nominal section stress under a given load was thereby established and was related to the Design Limit Load.

The spectrum was based on a known representative Wing Pivot Bending Moment Spectrum. Full scale static wing testing and the manufacturer's original stress analysis reports were used to convert this spectrum to be representative of the stress at the region of interest in the outboard section of the lower wing skin. The spectrum was truncated to reduce the total number of cycles to a manageable level and this was validated both analytically and experimentally. The static and fatigue loads to be applied to representative test articles were therefore accurately determined.

The present work has provided valuable new insights into the development of static and fatigue test loads, including an improved understanding of the effects of cycle matching techniques and spectrum truncation.

Contents

1. Introduction

In February 1994, a 48 mm (tip to tip) through thickness chordwise fatigue crack was discovered at a Fuel Flow Passage (FFP) in the port lower wing skin of a Royal Australian Air Force (RAAF) F-111C aircraft (Tail Number A8-145). The crack was located in the outboard section of the wing, approximately mid way between the inboard and outboard fixed pylon at Forward Auxiliary Spar Station (FASS) 281. The wing skin is manufactured from 2024-T851 Aluminium Alloy. The crack location is shown in Figure 1. Fracture mechanics calculations indicated that the presence of this crack had degraded the residual strength to 150 MPa (21.7 ksi) which is considerably less than the predicted Design Limit Stress for the area of 240 MPa (34.6 ksi, Reference 1).

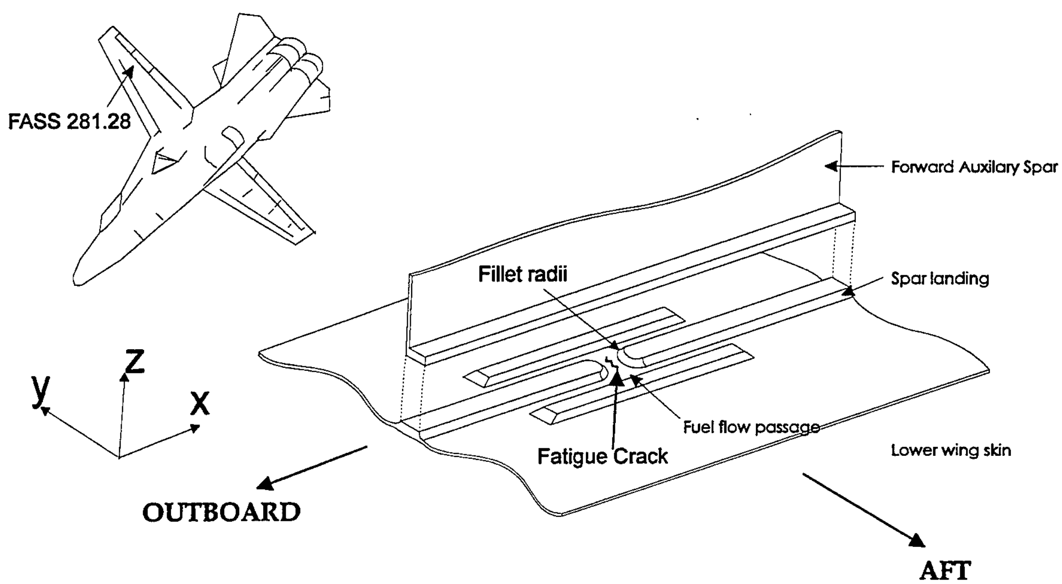


Figure 1. Location of Cracking in F-111 Lower Wing Skin

A mechanically fastened metallic doubler repair option was initially considered but was rejected for a range of reasons including undesirable aerodynamic implications and because the crack and the underlying structure would then be uninspectable. The only alternative to scrapping the wing was a bonded composite repair. A bonded boron epoxy repair was designed and implemented by RAAF personnel, on the basis of design principles and application procedures developed by Defence Science and Technology Organisation (DSTO, References 2 and 3) and now incorporated by the RAAF in an Engineering Standard (Reference 4). Details of the repair design are given in Reference 5. The repair was granted interim approval, pending the results of a comprehensive validation and substantiation program to be undertaken by DSTO, Aeronautical and Maritime Research Laboratory (AMRL). The validation and substantiation program requirements can be summarised as follows:

a. To validate the repair design procedures by an independent method. In this case a detailed three dimensional Finite Element Analysis (FEA) was used.

b. To substantiate the repair through testing of representative test specimens.

The testing of representative specimens required the determination of suitable loads, both static and fatigue, to be applied. This report describes how the test loads were derived. The results of the FEA validation are detailed in Reference 6, and the substantiation testing results are found in Reference 7. The repair design was found to be sound and the test results demonstrated that the repair was acceptable in terms of static strength, durability and damage tolerance.

2. Static Loading

2.1 Design Limit Load

The original stress analysis for the F-111 wing box (Reference 1) produced a Design Limit Stress for the FASS 281 region of 240 MPa (34.6 ksi). A finite element model of the area was created for two main purposes; to perform the validation exercise, and to determine an accurate nominal stress for the FASS 281 region. A detailed three dimensional FE model was created (see Reference 8) and calibrated against data from a strain survey on a full scale test wing.

The test wing was subjected to loads representing those applied during Cold Proof Load Testing (CPLT) where the flight loads under design limit conditions are approximated by applying loads via a series of jacks along the wing. The jacks provide a load distribution along the wing which approximates the Design Limit Load (DLL) case (as per Reference 1) but is not exact. DLL conditions in terms of vertical shear and bending moment at the root of the wing (at the pivot point) closely approximate design limit, but the conditions at FASS 281 are less than DLL. The axial spanwise stresses at FASS 281 are the primary stresses driving the fatigue cracking and are caused by wing bending. The local bending moment at FASS 281 is the determining parameter and the axial spanwise stress in the lower skin is directly proportional to it (see Reference 9).

Under CPLT conditions, the local bending moment at FASS 281 is 1.61×10^6 inch pounds or 1.61 Million Inch Pounds (MIPS, see Reference 8). The Reference 8 Finite Element Model was subjected to a nominal axial stress of 193 MPa (28.0 ksi) in order to achieve a strain calibration under CPLT conditions. The bending moment at FASS 281 under design limit conditions (Reference 1) is approximately 1.83 MIPS. The nominal stress at DLL is therefore calculated as follows:

$$\sigma_{FASS281} \text{ (at DLL for 4.2 mm skin thickness)} = \frac{1.83}{1.61} \times 193 = 219 \text{ MPa (31.8 ksi)}$$

This compares well with the original estimate from Reference 1 of 240 MPa (34.6 ksi).

For the purposes of conducting a conservative substantiation, it was decided to also account for possible variation in wing skin thickness. As discussed in Reference 8, the minimum skin thickness specified for the area is 3.6 mm. This compares with a measured skin thickness on the test wing (used for the calibration) of 4.2 mm. The DLL stress was therefore further factored as follows:

$$\sigma_{FASS281} = \frac{4.2}{3.6} \times 219 = 256 \text{ Mpa (37.1 ksi)}$$

This was the stress used to calculate the load to be applied to the test specimens when conducting static tests under DLL conditions. As described in Reference 7, the loads to be applied to each specimen were determined by multiplying the DLL stress by the cross sectional area of the specimen, ignoring any crack present and the addition of a boron patch. For certification purposes, demonstration that the repair restored static strength to at least Design Ultimate Load (DUL) which is 1.5 times DLL was required.

2.2 Cold Proof Load Test Loads

From Reference 10, the CPLT cycle is a sequence of four loads; -2.4 g, +7.33 g, -3.0 g, +7.33 g.

As stated previously, the FE model calibrated stress at CPLT was 193 MPa (28.0 ksi) for the maximum positive load case of +7.33 g. However, this was for a 4.2 mm skin thickness, so once again this was factored as follows:

$$\sigma_{FASS281} \text{ (at CPLT for 3.6 mm skin thickness)} = \frac{4.2}{3.6} \times 193 = 225 \text{ MPa (32.6 ksi)}$$

An early run of the FE model had produced a stress figure for the CPLT calibration of 200 MPa (29.0 ksi) and this figure had already been used to derive the loading for early panel specimen tests. Using this figure produced the following:

$$\sigma_{FASS281} \text{ (at CPLT for 3.6 mm skin thickness)} = \frac{4.2}{3.6} \times 200 = 233 \text{ MPa (33.9 ksi)}$$

This is a slightly more conservative figure and since it had been used in the early tests it was decided to retain it for the whole testing program. The CPLT sequence stresses were scaled according to the bending moments produced in the test (see Reference 9) and the resulting sequence was as follows:

-93.8, +233, -116, +233 (MPa) or -13.6, +33.9, -16.8, +33.9 (ksi)

The CPLT loads are experienced by each aircraft at a nominal 2,000 hour interval. These loads were included during the fatigue testing at 2,000 hour intervals to ensure that a representative load history was applied.

3. Fatigue Stress Spectrum

3.1 Wing Bending Moment Ratio

As discussed in the previous section, the axial stresses driving the cracking at FASS 281 are a function of the local applied bending moment. Since a representative load spectrum for wing pivot bending moment at the root of the wing (the pivot point) was available (Reference 11), it was decided that derivation of a spectrum for FASS 281 may be feasible. This required a relationship between bending moment at the pivot and local bending moment at FASS 281.

The ratio of bending moment at FASS 281 to bending moment at the pivot is not expected to be constant under all conditions. However, the load spectrum data from Reference 11 was not connected to individual load cases or flight conditions. To produce a spectrum therefore required the use of a constant ratio. Investigation was required to determine if this was a reasonable thing to do, and if so, to quantify the ratio to be used.

From References 1 and 12, the spanwise bending moment distributions for a range of design conditions was determined. These are plotted in Figure 2 for the F-111A (short wing aircraft) and Figure 3 for the FB-111 (long wing aircraft). Bending moment is plotted as a function of spanwise distance from the pivot. FASS 281 is 211 inches from the pivot (FASS is measured from the centre line of the aircraft, and the pivot pin is approximately 70 inches outboard from the centre line). The worst case design condition from each case is shown in a composite plot in Figure 4. These plots demonstrate that the relationship between bending moment at FASS 281 and at the pivot is similar for a range of design conditions. Using the worst case design condition for the long wing FB-111 aircraft (equivalent to the RAAF F-111C) gives the following relationship:

Wing Pivot Bending Moment (WPBM) = 19.3 MIPS

Bending Moment at FASS 281 ($BM_{FASS281}$) = 1.83 MIPS

$$\frac{BM_{FASS281}}{WPBM} = \frac{1.83}{19.3} = 0.0948$$

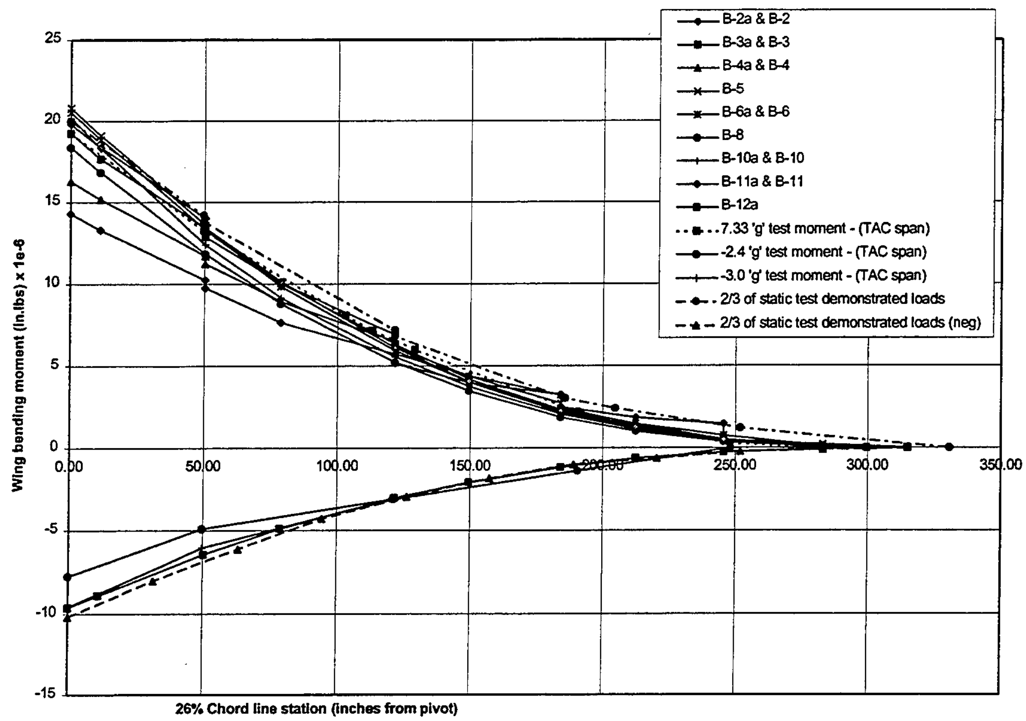
3.2 Stress at FASS281 per WPBM

From Section 2, the nominal stress at FASS 281 for a 3.6 mm skin thickness under DLL conditions ($BM_{FASS281}$ = 1.83 MIPS) using the figure from the early run of the FE model is as follows:

$$\sigma_{FASS281}\,\text{(at DLL for 3.6 mm skin thickness)} = \frac{4.2}{3.6} \times \frac{1.83}{1.61} \times 200 = 256 \text{ MPa (38.5 ksi)}$$

The ratio for stress at FASS 281 to WPBM can therefore be calculated as follows:

$$\frac{\sigma_{FASS281}}{WPBM} = \frac{38.5}{19.3} = 2.0 \text{ ksi per MIP}$$



Notes:

The curves on this plot were produced from the following sources:

a. The static test demonstrated loads are from the aircraft certification tests whereby the 2/3 loads are the same as limit load (ie limit load = 2/3 x ultimate load). The data required to produce these curves was not available in tabulated form. The values were scaled off plots in Reference 11 and are reproduced here.

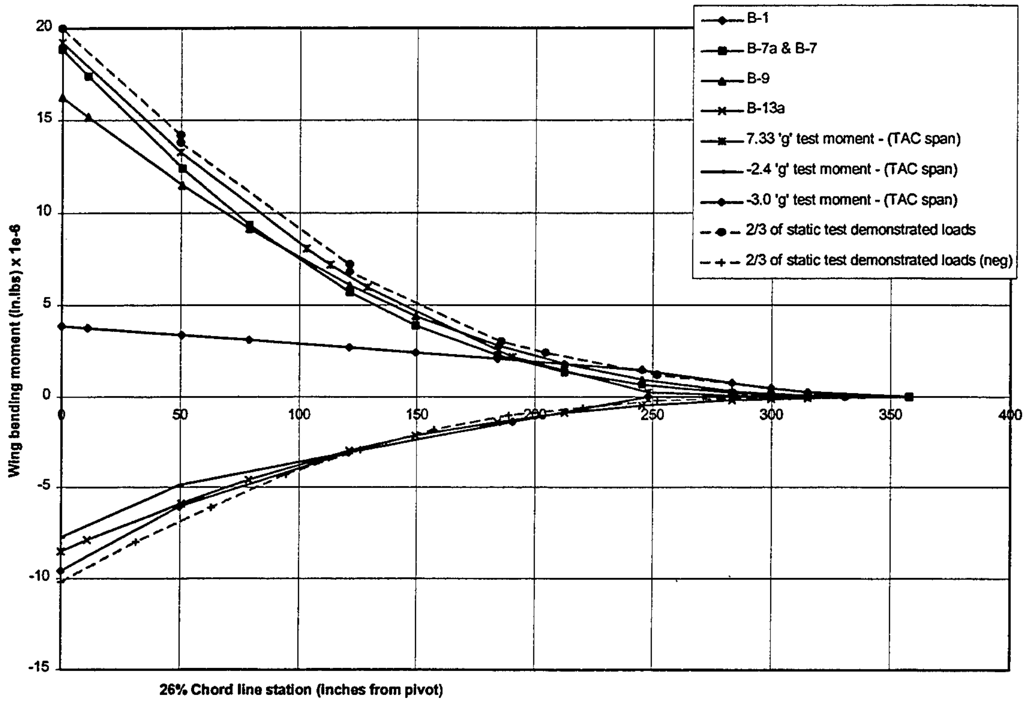
b. The 7.33g, -2.4g and -3.0g test moment cases are from CPLT data whereby wing loads are applied using several jacks at specific loading points (Reference 9)

c. Conditions B-2 through B-12a are from Reference 1. Details are as follows:

CONDITION	CONFIGURATION	SWEEP ANGLE	Nz	Mach No	T_L °F	T_{AW} °F
B-2a, B-2	RPOI/ 4 - 3000# stores/wing	26°	5.86	0.88	110	110
B-3a, B-3	RPOI/ 2 - 3000# stores/wing	35°	5.86	0.94	100	100
B-4a, B-4	RPOI/ 3 - 3000# stores/wing	26°	5.86	0.88	110	110
B-5	Bal. Sym:+ve maneouvre/4- light wgt. stores/wing	26°	7.33	0.88	140	140
B-6a, B-6	Bal. Sym:+ve maneouvre/ clean wing	35°	7.33	0.94	100	100
B-8	Bal. Sym:+ve maneouvre/ clean wing	50°	7.33	1.40	205	205
B-10a, B-10	Bal. Sym:+ve maneouvre/ clean wing	26°	7.33	0.88	85	85
B-11a, B-11	RPOI/ 3 - 3000# stores/wing	35°	5.86	0.94	100	100
B-12a	Bal. Sym.-ve maneouvre/ clean wing	35°	-3.0	0.96	85	85

Some of the conditions above have an "a" suffix, for example B-2a. This refers to flight conditions that have the Centre of Pressure (CP) forward. Conditions without the suffix have an aft CP.

Figure 2. Wing Bending Moment at Limit Load Along the 26% Chord Line for the F-111 A (Wingspan 314.8 in)



Notes:

The curves on this plot were produced from the following sources:

a. The static test demonstrated loads are from the aircraft certification tests whereby the 2/3 loads are the same as limit load (ie limit load = 2/3 x ultimate load). The data required to produce these curves was not available in tabulated form. The values were scaled off plots in Reference 11 and are reproduced here.

b. The 7.33g, -2.4g and -3.0g test moment cases are from CPLT data whereby wing loads are applied using several jacks at specific loading points (Reference 9)

c. Conditions B-1 through B- 13a are from Reference 1. Details are as follows:

CONDITION	CONFIGURATION	SWEEP ANGLE	N_Z	Mach No	T_L °F	T_{AW} °F
B-1	Balanced symmetrical +ve manoeuvre for tip design	–	–	–	250	270
B-7a, B-7	Balanced symmetrical +ve manoeuvre/ clean wing	35°	6.5	1.05	85	85
B-9	Balanced symmetrical +ve manoeuvre/ clean wing	50°	6.5	1.40	167	167
B-13a	Balanced symmetrical -ve manoeuvre/ clean wing	35°	-3.0	1.00	50	50

Some of the conditions above have an "a" suffix, for example B-7a. This refers to flight conditions that have the Centre of Pressure (CP) forward. Conditions without the suffix have an aft CP.

Figure 3. Wing Bending Moment at Limit Load Along the 26% Chord Line for the F-111B (Wingspan 357.86 in)

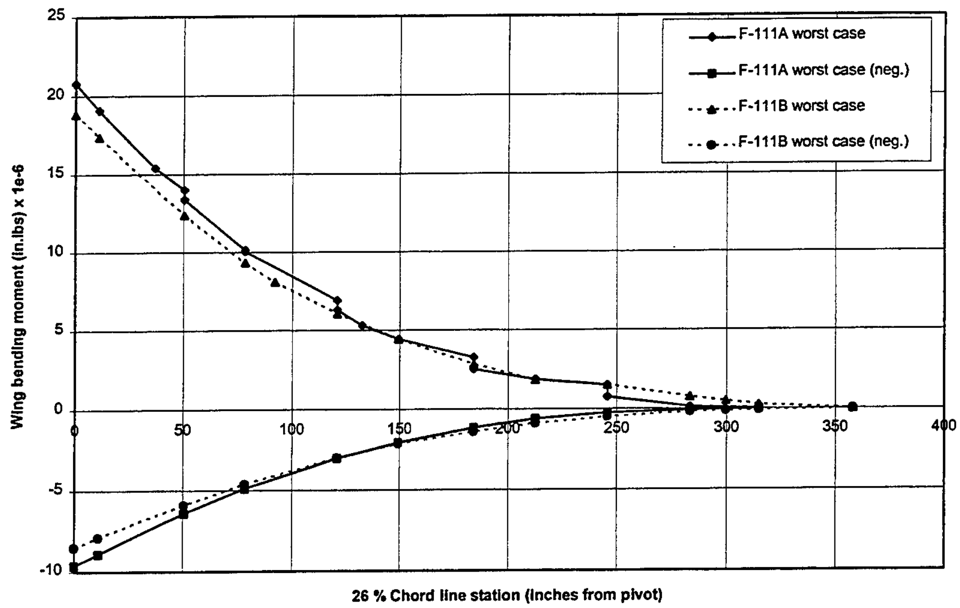


Figure 4. Comparison of "Worst Case" Bending Moment Diagrams for F-111A (Wingspan 314.88 in) and FB-111 (Wingspan 357.86 in) Models at DLL (Reference 1)

3.3 Stress Exceedance Spectrum at FASS 281

From Reference 11, the WPBM exceedance spectrum was available and is shown in Figure 5. This spectrum was developed by the RAAF using data gathered from four specially instrumented F-111C aircraft performing operational missions during the 1980s. The spectrum is considered by the RAAF to be the most representative spectrum currently available and is based on a block of 199 flights/499.1 flight hours. Because it is in exceedance diagram form in Figure 5, it is shown as the distribution of maxima and minima against exceedances per 1,000 hours. This shows that the spectrum contains approximately 3.5×10^6 peaks and the same number of troughs in 1,000 flying hours.

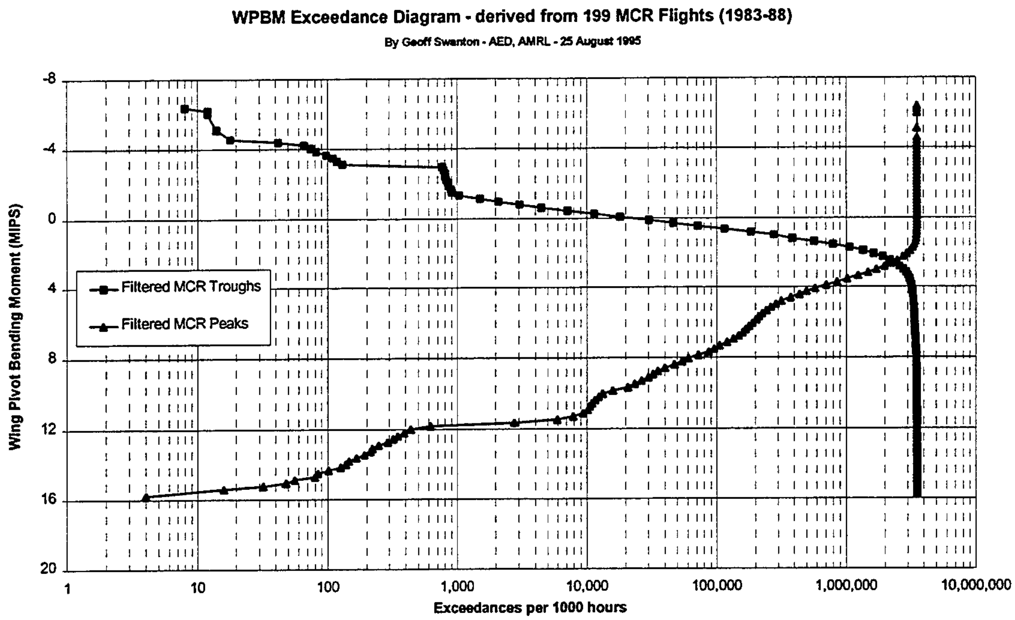


Figure 5. Wing Pivot Bending Moment Exceedance Diagram

Using the relationship developed in Section 3.2, the WPBM exceedance spectrum was converted to a stress exceedance spectrum by a simple scaling process. The stress spectrum at FASS 281 is shown in Figure 6.

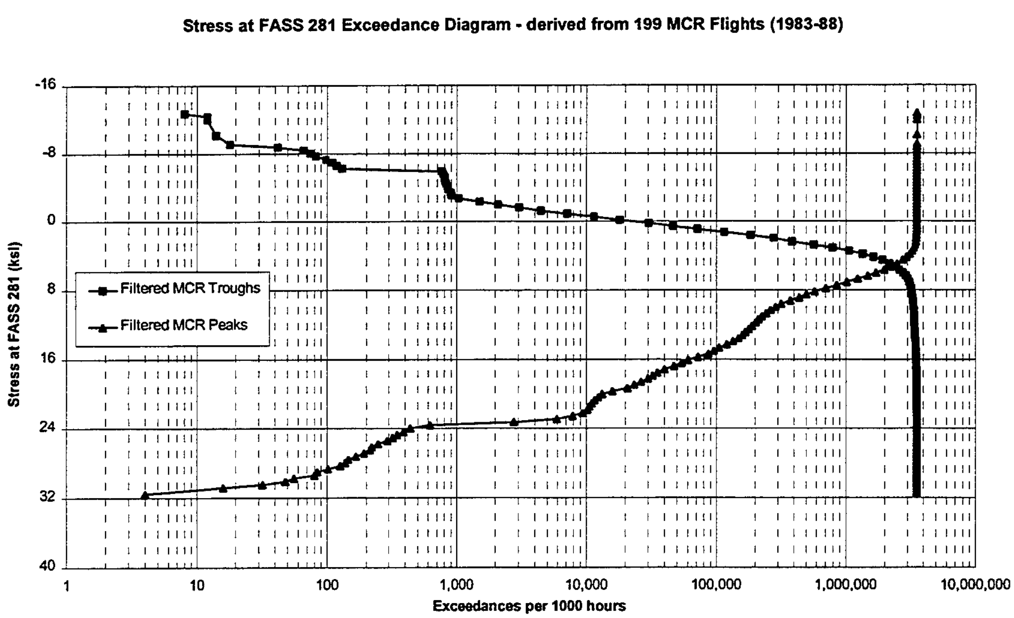


Figure 6. Stresss Exceedance Diagram at FASS 281

3.4 Blocked Spectrum

The spectrum shown in exceedance diagram form earlier was available in a flight by flight, turning point by turning point form. However, to simplify the testing and in particular to establish initial spectrum loading test results quickly, it was decided to devise a simplified, truncated blocked spectrum based on the exceedance diagram shown in Figure 5. This was done as follows:

a. The spectrum was converted to exceedances per 100 hours, to create a block representing 100 hours. In this way a smaller block of loading could be repeated many times and therefore avoid the problem of unrepresentative load sequence effects.

b. The spectrum was "blocked" using the Reference 13 software. The first step of this was to define the spectrum in terms of the peak and trough stress values at five exceedence levels, the steady stress and the total number of exceedences. The points are shown connected by straight lines in comparison with the full spectrum in Figure 7. The shape of the spectrum is therefore "approximated" by straight lines connecting the six points.

c. The Reference 13 software was used to create a 16 level "blocked" spectrum. The software is set up to enable the user to select the number of discrete levels the spectrum is blocked to, with 16 being the maximum available.

d. Since it was desired to truncate the spectrum at a reasonable level, it was decided to delete the seven smallest blocks and perform a true truncation exercise as per References 13 and 14.

e. The true truncation exercise involved increasing the number of cycles left in the smallest block left after step (d) above, and running a simplified crack growth analysis and comparing the result with an analysis including the full 16 blocks. The number of cycles in the smallest block was then adjusted and the process repeated until the predicted crack growth lives matched. In this way, the truncated spectrum produces the equivalent "damage" as the full spectrum. The spectrum was therefore reduced to 9 blocks and a total of 8,628 exceedences per 100 hours as shown in Figure 8.

f. The load spectrum as shown in Table 1 was therefore derived.

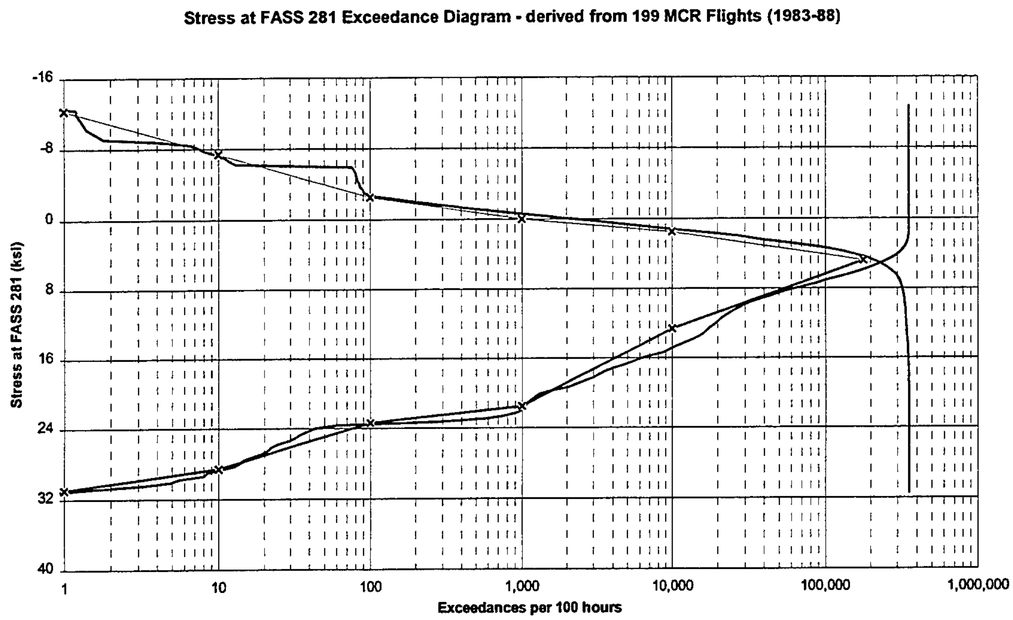


Figure 7. Stress Exceedance Spectrum per 100 Hours With Linear Approximation

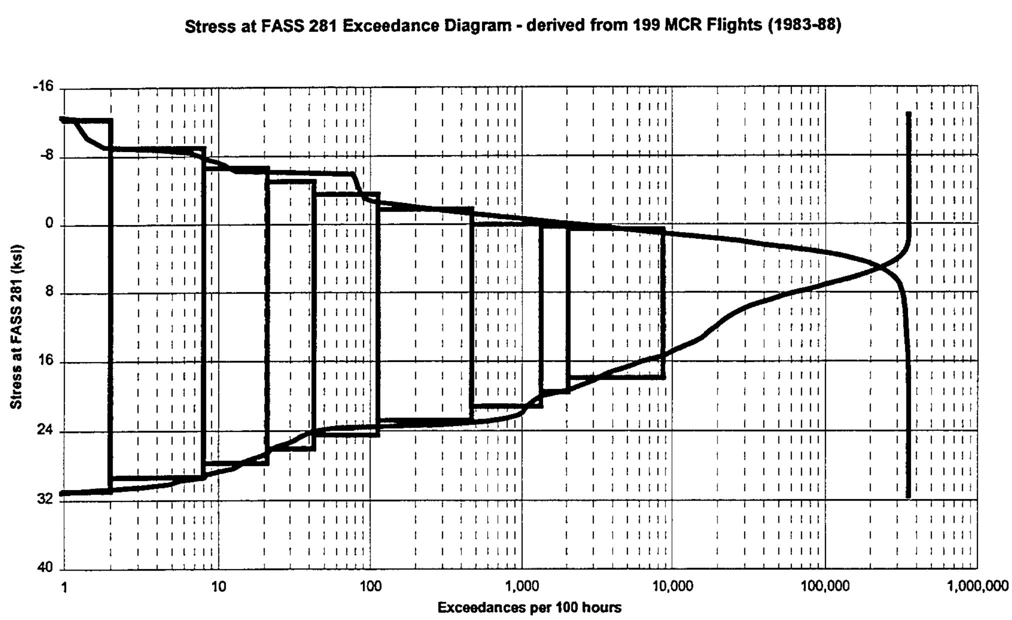


Figure 8. Stress Exceedence Spectrum per 100 Hours With Blocked Approximation

Table 1. "Blocked" Stress Spectrum Representing 100 Flight Hours

Total Occurrences	Exceedences	Minimum Stress (ksi)	Maximum Stress (ksi)
2	2	-12.3	31.0
6	8	-8.957	29.362
13	21	-6.575	27.725
22	43	-5.032	26.088
70	113	-3.489	24.45
357	470	-1.727	22.813
874	1344	0.055	21.175
709	2053	0.331	19.538
6575	8628	0.607	17.9

As detailed in the Reference 6 report, the blocked spectrum was applied to several specimens and was found to produce rapid crack growth. The rapid crack growth was considered to be caused by the blocked spectrum being overly conservative. This is because the blocking approach matches the highest peaks with the lowest troughs and pairs them to form load cycles. In practice however, the cycle matching is far less severe and the range mean pair or rain flow method (Reference 15) is far more representative. In this case the spectrum was available in flight by flight, turning point sequence format. The only problem was that the 199 flight/ 499.1 flight hour block contained approximately 3.5 x 10^6 turning points, so truncation would be required. This was carried out as discussed in the next section.

3.5 Cycle By Cycle Spectrum

The flight by flight, cycle by cycle spectrum was subjected to a discrimination or truncation process. It was found that by applying a discrimination level of 5 ksi, the number of turning points in the 499.1 hour block was reduced from 3.5 x 10^6 to 35,273. Truncating the spectrum at this level was later validated by testing patched and cracked specimens at constant amplitude loading of 5 ksi. There was no crack growth observed at this level. This effectively validated the truncation process, at least in terms of crack growth under the patch. The validity of truncating the spectrum at 5 ksi in terms of potential damage to the adhesive and the total bonded repair system is being further investigated at this time (Reference 16). Initial indications are that truncation at 5 ksi does not remove any damaging loads. The cycle by cycle spectrum was used for the majority of the repair substantiation testing as reported in Reference 7. The cycle by cycle spectrum is considered to be more representative than the "blocked" spectrum which was only used for a limited number of early tests.

4. Conclusion

The loads to be applied to a series of static and fatigue test specimens representing a structural detail in the F-111 lower wing skin were successfully derived. The static loads were based on a calibrated finite element model analysis. The fatigue spectrum loads were derived from a wing bending moment spectrum which is known to be representative for the RAAF F-111C fleet. Adjustments to the spectrum utilised information from the manufacturer's original stress analysis reports and the results from full scale static wing tests at AMRL. The spectrum was expressed in exceedence diagram and cycle by cycle forms, and the validity of truncating the spectrum was ensured through both analysis and test. The blocked version of the spectrum, based on the exceedence diagram, was found to produce an extremely conservative result when applied to the testing. The use of this approach is therefore only recommended when no further information other than an exceedence type diagram is known.

5. Acknowledgments

The authors wish to acknowledge the assistance and advice provided by Dr Francis Rose and Messrs L. Molent, R. Boykett and K. Watters of the Airframes and Engines Division, AMRL.

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No limitations

17. CASUAL ANNOUNCEMENT Yes

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Cyclic loads, fatigue test, fiber laminates, repair, F-111 aircraft

19. ABSTRACT
Substantiation of a bonded composite repair to an F-111 lower wing skin required the development of representative loads for specimen testing. This report describes the development of representative static and fatigue spectrum loads to be applied during the testing. The spectrum was based on a known representative Wing Pivot Bending Moment spectrum. Full scale static wing testing and the manufacturer's original stress analysis reports were used to convert this spectrum to be representative of the stress at the region of interest in the outboard section of the lower wing skin. The spectrum (representing 199 flights or 499.1 flying hours) was expressed in both blocked and cycle by cycle form. Truncation was performed to reduce the total number of cycles to a manageable level and this was evaluated both analytically (for the blocked spectrum) and experimentally (for the cycle by cycle spectrum) to ensure that damaging cycles were retained. The blocked spectrum was found to give highly conservative results and this approach is recommended only where no further information about the cycle history is known. The implications of the present results for realistic spectrum truncation are briefly discussed.