

# F/A-18E/F FULL SCALE STRUCTURAL FATIGUE TESTING

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**ABSTRACT:** The F/A-18E/F structural airframe consisting of the wings, fuselage, and empennage has completed an extensive full scale fatigue test program. This test of the Engineering and Manufacturing Demonstration (EMD) configuration of the airframe completed three lifetimes, or 18,000 simulated flight hours (SFH), of cycling and has experienced a complete structural teardown and evaluation. This test article is referred to as FT50.

In addition, the United States Navy authorized a significant structural change to the F/A-18E/F forward fuselage and wing to support a cost reduction initiative. A separate full scale structural fatigue test was also performed on these configurations of the forward fuselage and wing airframes. These test articles are referred to as FT76 (fuselage) and FT77 (wing). The FT76 fuselage test article was cycled for two lifetimes (12,000 SFH) and the FT77 wing test article was cycled for three lifetimes (18,000 SFH).

The teardown and analysis of these test articles is complete and the paper that will be submitted will summarize all three of these test programs including the structural configuration of the test articles, test spectrum content, fatigue failures experienced during testing, and results of the structural inspection effort conducted during the teardown of the test articles. The paper will also document the process that was used by Boeing and the United States Navy to collaboratively develop the teardown procedure and extract all desired data from the test articles.



Figure 1 – F/A-18E/F

## **1.0 INTRODUCTION**

The F/A-18E/F structural airframe (consisting of the wings, fuselage, and empennage) has successfully completed an extensive full scale fatigue test program. Figure 1 presents an image of the F/A-18E/F. This test of the Engineering and Manufacturing Demonstration (EMD) configuration of the airframe started in May of 1997 and completed three lifetimes of cycling or 18,000 simulated flight hours (SFH). Two lifetimes of testing were required to certify the structure. However, the third lifetime of cycling was applied to the airframe to identify any structure that might experience a failure or to certify structure beyond its original design life. The F/A-18E/F was designed to sustain a structural life of 12,000 SFH without crack initiation so the identification of some cracks beyond this life was anticipated. Crack initiation is defined by the U.S. Navy to be the existence of a flaw that is at least 0.01" (0.254 mm) in length. However, damage tolerance analysis is performed for critical structural components of the airframe. This testing demonstrated that the F/A-18E/F structure is a very robust design that will continue to perform its mission for many years into the future. The full scale test article (referred to as FT50) recently completed a structural teardown and results are currently being evaluated.

In addition, a significant structural change was implemented relative to the F/A-18E/F forward fuselage and wing to support a cost reduction initiative. Separate full scale structural fatigue tests were also performed on these configurations of the forward fuselage and wing airframes. These test articles are referred to as FT76 (forward fuselage) and FT77 (wing). The FT76 fuselage test article was cycled for two lifetimes (12,000 SFH) and the FT77 wing test article was cycled for three lifetimes (18,000 SFH). The FT76 fatigue test and fuselage redesign were documented by this paper's author at the 2003 ICAF symposium (Reference 1). This 2007 ASIP paper will primarily address the FT50 test article since it represented the entire airframe and was one of the most extensively tested of the three F/A-18E/F full scale test programs.

The teardown and analysis of these test articles are in progress and this paper will present the status of these efforts. This paper will also discuss the technique used to execute the disassembly and non-destructive inspection of the structural components of the three test articles. This was a significant collaborative effort between The Boeing Company (contractor) and the United States Navy (customer) that resulted in a cost effective approach to identify crack indications at a minimum size that was acceptable to the customer. The paper will summarize the FT50 test program including the structural configuration of the test articles, test spectrum content, fatigue failures experienced during testing, and results of the structural inspection effort conducted during the teardown of the test articles. The paper will also document the process that was used by Boeing and the United States Navy to collaboratively develop the teardown procedure and extract all desired data from the test articles.

## **2.0 STRUCTURAL CONFIGURATION OF TEST ARTICLES**

The FT50 and FT76 full scale test articles included forward, center, and aft fuselage assemblies, left and right wing assemblies, and vertical tail assemblies. The horizontal tail stabilator assemblies were separately test. Omitted equipment that was simulated by test fixtures included landing gear components (made from partially machined landing gear forgings), horizontal stabilators, engines, gun, radar/radome, arresting hook, linear actuators in statically determinant applications, wing pylons, centerline pylon, and the ejection seat.

Figure 2 shows the FT50 test article and the components that were omitted from the structure. These items were either tested in a separate test or they were not considered to be structural items.

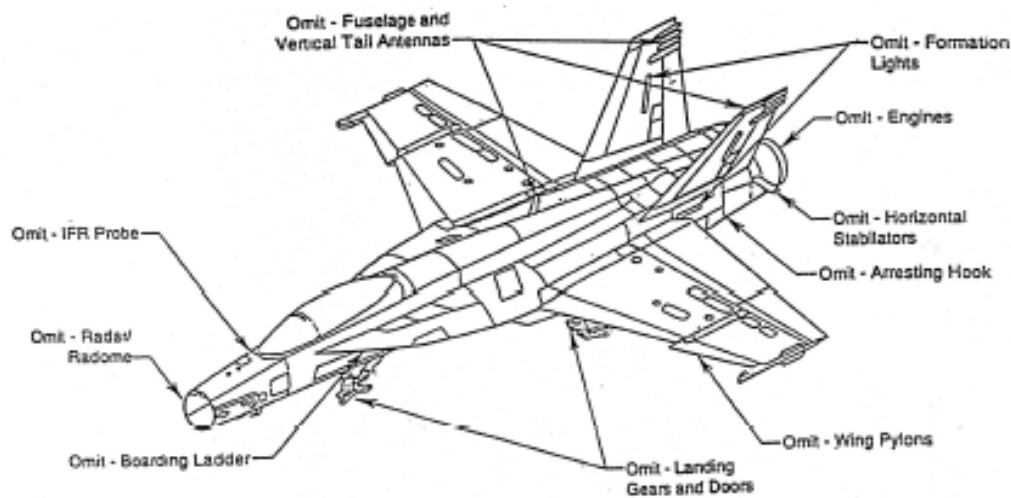


Figure 2 – FT50 test article

The FT77 full scale wing test article included inner and outer wing torque box assemblies, leading edge and trailing edge control surfaces, leading edge flap transmissions, and the wing fold transmission assembly.

### 3.0 TEST FIXTURE

Wing, control surface, leading edge extension (LEX), and vertical tail pressure loads were transmitted through neoprene pads bonded to the airframe surface. The pads were connected to a hydraulic loading cylinder through a series of turnbuckles and tension whiffletree beams or tension/compression whiffletree beams. Fuselage aerodynamic and inertia loads were introduced through both fittings and neoprene pads and whiffletrees attached to the airframe. Additional aerodynamic, ground and inertia loads were transmitted through simulated aircraft components including main and nose landing gears, horizontal stabilator, arresting hook, pylons, engines, radar/radome, gun, and seat.

The weight of the test article, simulated aircraft components, and test load application hardware were counterbalanced by either tare loads in the hydraulic loading cylinders, air cylinders, or weight baskets connected by cable via levers and pulleys. The hydraulic cylinder loads were controlled by an electro-hydraulic servo system driven by computer generated input command signals with load feedback from load cells that were mechanically in series with the hydraulic cylinders. Figure 3 presents the test article and its placement in the test fixture. Figure 4 depicts the whiffletree assembly that is capable of introducing tension and compression loads to the moldline surface of the wings.



Figure 3 – Test article in fixture



Figure 4 – Tension/compression whiffletree assembly

#### **4.0 TEST SPECTRUM, LOADS, AND INSTRUMENTATION**

The fatigue loads consist of maneuver loads, ground loads and dynamic loads which combine with the Master Event Spectrum (MES) to create design analysis spectra. The development of the MES was documented by this paper's author at the 2001 ICAF symposium (Reference 2). Symmetric maneuvers included both steady state and abrupt conditions from -2.5g to 8.5g. Abrupt maneuvers only occur for positive  $N_z$  events. Asymmetric maneuvers included 40%, 60%, 80%, and 100% lateral stick deflections and consist of -1g rolls, +1g rolls, and Rolling Pull-Outs (RPOs) up to 8.5g. Negative 1g rolls were limited to rolling through 180 degrees and positive 1g rolls continued through 360 degrees. Rolling angles (bank to bank) were dependent upon the entry g level for RPOs at  $N_z$  greater than 1g. During test operations, loads simulations represented the flight control system being operational for all maneuver loads with the exception of the g-limiter. The g-limiter was simulated as non-operational for RPOs at  $N_z$  greater than or equal to 5.5g and symmetric maneuvers greater than 7.5g to achieve the desired load levels. The test spectra used for all of the test articles represented 1000 SFH and 690 flights and was repeated to achieve the desired test life. Occurrences of dynamic cycles/loads for the vertical tails and engine mounts were evenly

distributed throughout the test spectrum following symmetric events greater than 5g. The dynamic cycles were applied with the test system hydraulic actuators

A total of 1,585 channels (including loads, reactions, and rigid body deflections) were recorded as continuous data fatigue monitor (CDFM) measurands during the testing of FT50. Various strain surveys were performed on all test articles prior to start of test, periodically during the test, and at the completion of the test. This was done to monitor the structural health of the test article and aid in the identification of any variations in load path due to failed structural components.

## **5.0 STRUCTURAL TEARDOWN AND INSPECTION TECHNIQUE**

The airframe was disassembled in the following order: aft fuselage and L/H vertical tail, forward fuselage, and then the center fuselage. The L/H wing was disassembled in parallel with the fuselage disassembly. Figure 5 presents an overview of the teardown process that was used.

The teardown inspection program is intended to find cracks which may have started within the first or second lifetime, prior to the start of the third lifetime of stretch testing. It is of high probability that small cracks (i.e., those less than 0.050 inches) initiated during the third lifetime of stretch testing in the FT50 metallic structure. This inspection program was in large part being done by using the eddy current method for conductive non-magnetic material. Other NDI methods being utilized were ultrasonic A & C scans, magnetic particle, and level II fluorescent penetrant inspection. The eddy current method was used to inspect edges, radii, and holes.

The nature of the surface condition and the large amount of surface area being inspected adds to the difficulty of the inspection process. To help ensure the desired detection level is achieved with eddy current, the test sensitivity was validated on smaller .030" x .030" corner electrical discharge machine (EDM) notches on reference standards. All crack indications were documented via photographs as well as written records of the "as detected" surface length of the crack, when possible. It is typically difficult to document crack size with eddy current for cracks less than .050" in length even with a lower limit of approximately .030" in size. Every effort was made to document surface length of cracks less than 0.050" when possible. Crack indications less than 0.050" in length were reported with their location and orientation by marking the part and photographing. Photos and tables were also used noting the hole numbers and anomalies found. All parts subjected to eddy current inspections are stored for future reference. Cracks that initiated prior to the third lifetime of testing are of primary interest as they may subsequently affect the F/A-18E/F airframe configuration to be defined for certification.

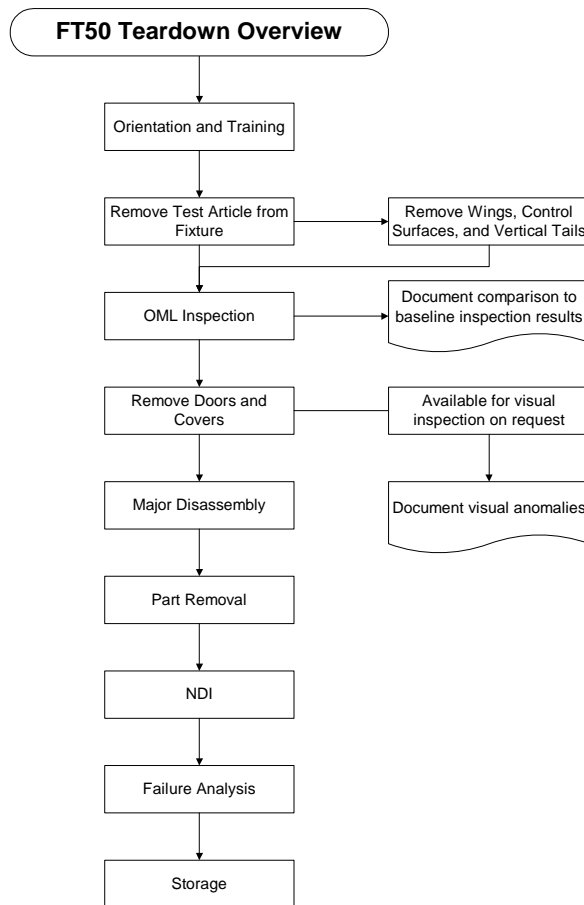


Figure 5 – Teardown Process

Paint stripping of the parts was generally not necessary. Chemical paint stripping (Turco 5351 or equivalent) was performed on selected areas as deemed necessary. The preferred method of cleaning fastener holes was to use nylon bristle “bottle” brushes with a cleaning solvent. The preferred method for cleaning other part surfaces was to use plastic scrapers. The use of a “light” hand reaming/deburr operation was only used as a final attempt to remove displaced material. A record was maintained identifying all holes that required hand reaming or deburring, along with a note on the amount of material removed. A record was also maintained identifying all holes that were considered to be un-inspectable using eddy current techniques

Eddy current equipment was standardized using the following approach for hole and surface inspections:

- Standardize for hole inspection such that response from the .030” x .030” EDM corner notch has approximate 50% screen response. Identify the signal strength from a .050” x .050” EDM notch. Typical inspection is performed with the rotating scanner eddy current gun.
- Standardize for surface inspections using the .100” x .020” EDM notch so that it has a 3 to 4 division deflection of the .050” x .025” EDM radii notch and the .030” x .030” corner edge notch on the reference standard. Draw a line around all detected cracks

## 6.0 TEST RESULTS

A few significant findings that were identified during the FT50 test program are presented in figures 6 through 9. The accumulated test life at the time of identification of the anomaly is noted in the title for the figures. Additional information regarding the analysis of the crack shown in Figure 6 is provided in Section 7.0 of this paper. The crack shown in Figure 9 required a redesign to the structure since this failure was determined to have occurred prior to 12,000 SFH which was the design life requirement for the F/A-18E/F program.

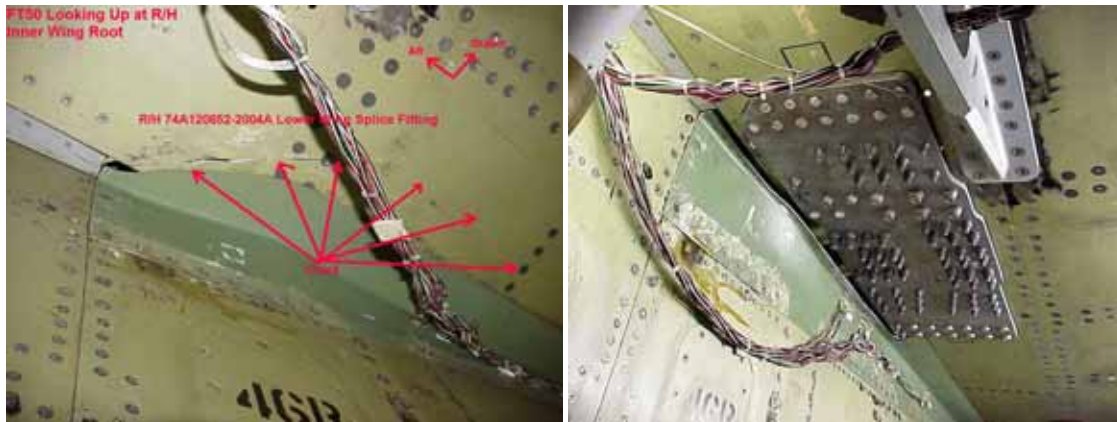


Figure 6 – Right-hand lower wing root skin 17” crack at 16,410 SFH (repair fitting shown at right)

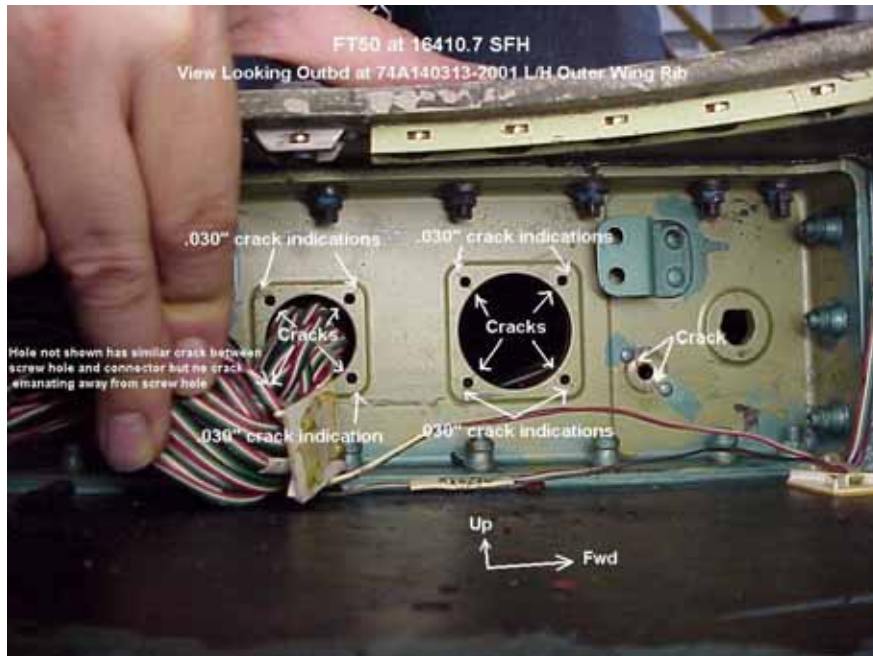


Figure 7 – Right-hand outer wing electrical cut-out cracks at 16,410 SFH



Figure 8 – Leading Edge Extension (LEX) cracks at 15,000 SFH



Figure 9 – Keel longeron Y555 cracks at 10,000 SFH

## 7.0 FAILURE ANALYSIS

The failure analysis process used during the teardown assessment included the typical approach of excising the failure zone from the structural part, exposing the failure surface, developing the fatigue spectrum that is associated with the load history at the failed detail, and then utilizing high magnification techniques to visually match the striation growth pattern to the repeated load pattern of the spectrum. This process allows the analyst to essentially work backwards from the existing crack tip and “read” the failure surface to determine the time of crack initiation for the failure. The surface is also examined under high magnification to find any anomalies and perform material verification.

One of the most challenging cracks that was analyzed was the failure of the right-hand lower wing skin titanium splice place on the FT50 fatigue test article after 16,410 SFH of testing that is shown in Figure 10. The failure created a significant change in load path and activated the error detection system causing the load actuator system to experience an automatic shut-down sequence. The decision was made to excise the crack for failure analysis and install a repair fitting to establish a structural load path so that the testing could continue toward the goal of 18,000 SFH. The excised crack and failure surface are shown in Figure 11.

The possibility of the crack being the result of improperly applied loads was considered. However, a review of the applied loads and CDFM data showed that the applied spectrum was within the acceptable tolerance. Detailed, p-version finite element models (shown in



Figures 12 through 15) were constructed to confirm the stress state at the critical fastener holes in the failed part. The FEM results contributed to a predicted crack initiation life of 20,000 SFH and a 15,000 SFH predicted crack growth life – an indication that this was not a case of poor design or insufficient analysis. As part of the failure analysis process, the failed part was examined using a scanning electron microscope (SEM) to search for manufacturing defects or other anomalies. A 'rogue flaw' was found at the failed hole as shown in Figure 16. This flaw was determined to be a manufacturing induced scratch along the bore of the fastener hole.

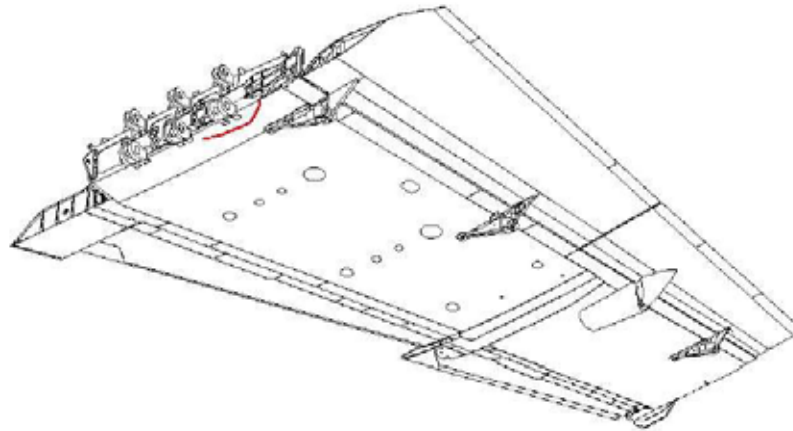


Figure 10 – FT50 lower wing root skin crack near aft lug at 16,410 SFH

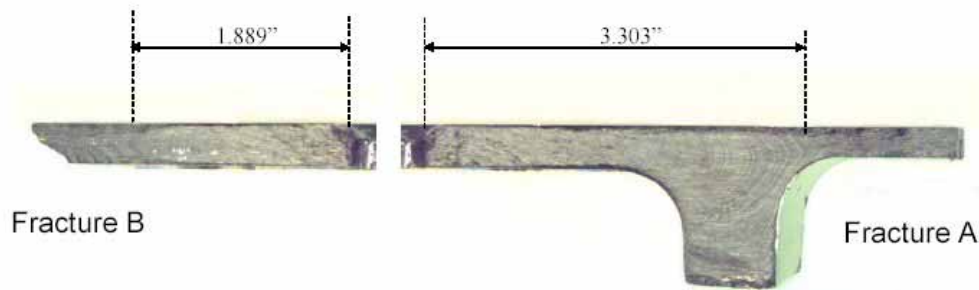


Figure 11 – Excised crack zone at fastener hole in wing root lower skin

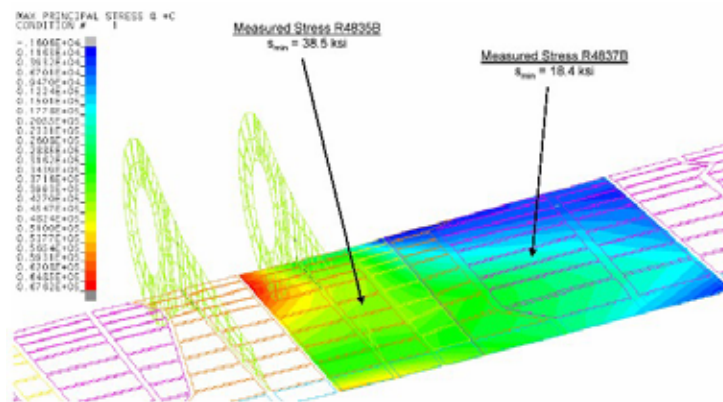


Figure 12 – Strain gages were used on other test articles to validate the finite element model results

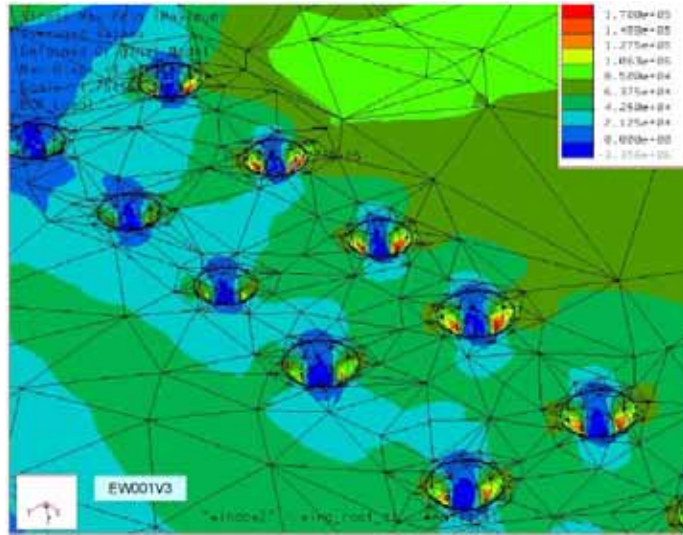


Figure 13 – Detailed models were constructed to characterize the stress field at the countersink holes

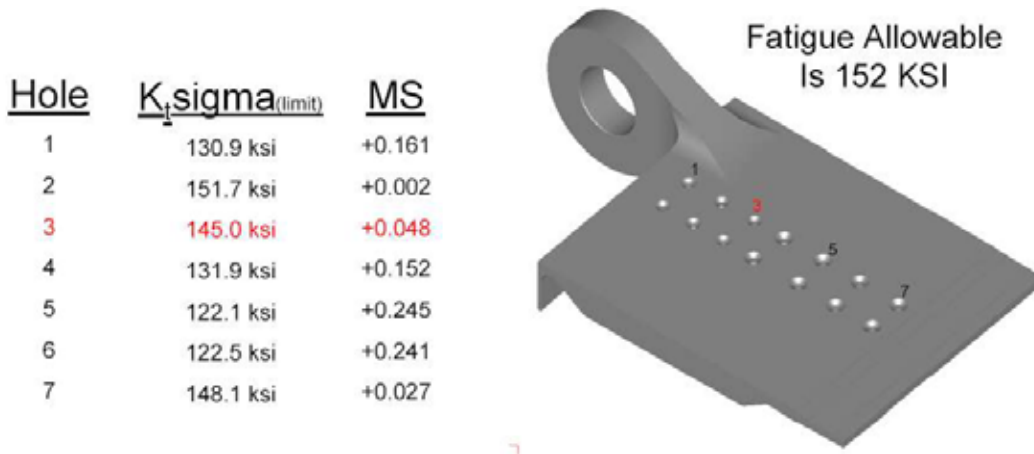
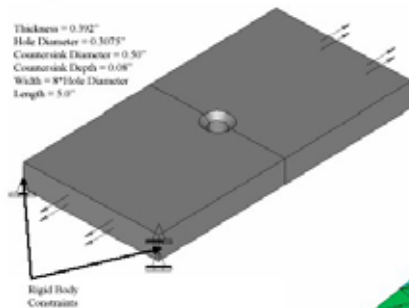


Figure 14 – The inboard adjacent hole (#2) had a higher stress than the cracked hole (#3)

### Analysis



- The Root Fitting Mechanica Model shows that the maximum stress is at the base of the countersink in the hole inboard of the hole that cracked.
- A simple model was developed to investigate the stresses through the thickness of the hole and the gradient away from the edge of the hole.

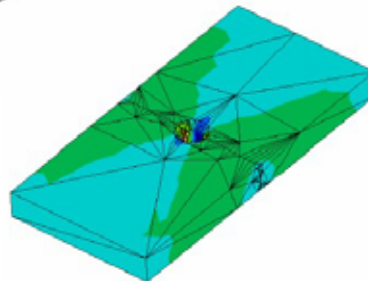


Figure 15 – Maximum stress is at base of countersink rather than in bore of hole where crack initiated

## Fracture A

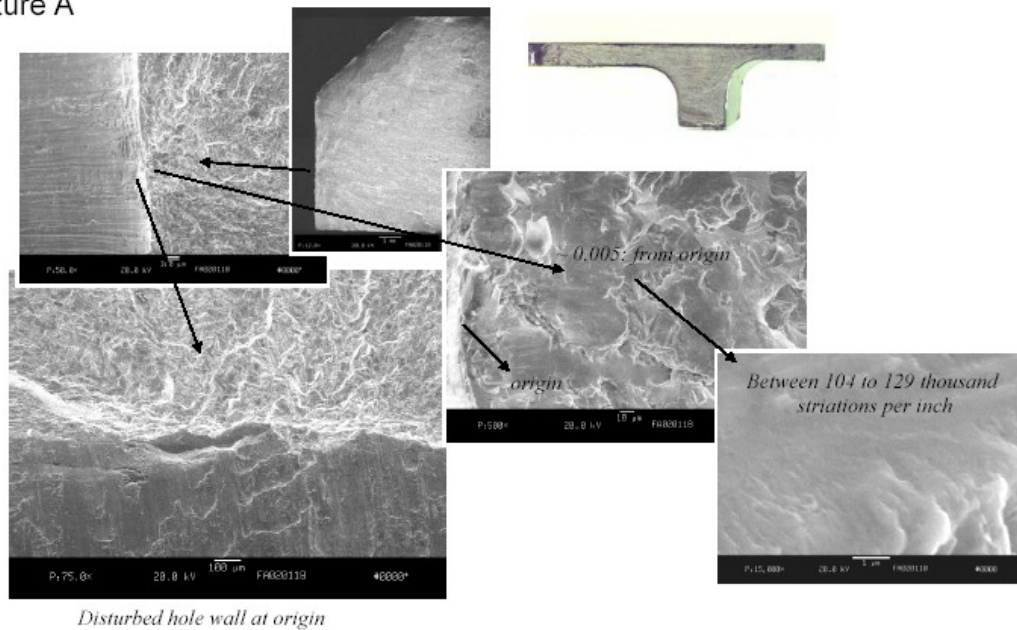


Figure 16 – Failure surface of crack emanating from flaw at fastener hole in wing root lower skin

The discovery of the manufacturing defect confirmed that a redesign was not required. The structural integrity of the original F/A-18E/F wing design was further confirmed by the following:

- The root fitting lasted more than 2.5 lifetimes before the crack reached critical length
- Even with the flaw, there is adequate crack growth life (10, 290 SFH)
- The flaw required more than one lifetime to initiate a crack (6,120 SFH)
- Three other full scale test articles were fatigue cycled with no cracks in this area
  - FT50 L/H wing successfully completed three lifetimes of testing
  - FT77 L/H wing successfully completed three lifetimes of testing
  - PT51 L/H wing successfully completed two lifetimes of testing

Table 1 provides a numerical summary of all of the parts that were identified to have cracks as well as a total summary of cracks on the FT50 full scale fatigue test.

FT50 Component	Crack Summary	
	Parts	Total
Wing	43	641
Forward Fuselage	53	492
Center Fuselage	133	1119
Aft Fuselage	56	293
Mechanisms	2	3
Vertical Tail	9	45
<b>Totals --&gt;</b>	<b>296</b>	<b>2593</b>

Table 1 – Summary of cracks on FT50 at completion of 18,000 SFH of testing

This summary of cracks listed in Table 1 is representative of cracks that were identified after three lifetimes of cycling on a test article that was designed to sustain only two lifetimes until crack initiation. A probability analysis was performed early in the F/A-18E/F program to determine the scatter factor that was to be used during fatigue analysis of structural components. A scatter factor of 1.0, i.e., design for 12,000 SFH and test for 12,000 SFH, would correspond to a 50% probability that a crack will initiate after two lifetimes (12,000 SFH) of testing. The E/F program incorporated a scatter factor of 1.33 (design for 16,000 SFH) into all fatigue design allowables which corresponds to an 18% probability that a crack will initiate after two lifetimes. Stretch testing the FT50 test article to three lifetimes (18,000 SFH) corresponds to an effective scatter factor of 0.89 since the structure was only designed for 16,000 but was tested for 18,000 SFH. This scatter factor corresponds to a 65% probability of a crack initiating. Therefore, the number of cracks identified on the FT50 test article is not unexpected.

## 8.0 CONCLUSION

The F/A-18E/F structural test program was comprehensive and comprised multiple full scale static and fatigue test articles that certified the airframe to meet the intended design life. In addition, the test program was successful in identifying structure that is capable of exceeding the intended design life. The success of the F/A-18E/F airframe continues to be realized by utilizing the E/F structural platform as the foundation of the EA-18G program as the next generation of electronic attack aircraft for the United States Navy. An image of an F/A-18E/F with the typical EA-18G store configuration is presented in Figure 17.



Figure 17 – EA-18G

## 9.0 REFERENCES

- 1) 2003 ICAF symposium paper: “Affordable Evolution: F/A-18E/F Forward Fuselage Modification and Structural Certification Incorporating Unitized Structure” - Timothy N. Callihan
- 2) 2001 ICAF symposium paper: “A Critical Point-In-The-Sky Approach to Master Event Spectrum Development” - Timothy N. Callihan