DAMAGE TOLERANCE ASSESSMENT OF THE A-7D AIRCRAFT STRUCTURE

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ABSTRACT

A damage tolerance assessment of the Vought A-7D Corsair II fighter/attack aircraft structure was conducted by a team of Vought and United States Air Force (USAF) engineers (Littlefield, 1977). The purpose of the assessment was to determine structural life operational limits using fracture mechanics, establish improved structural inspection requirements, identify structural modifications needed to ensure an 8,000-flight-hour operational life and develop an individual aircraft tracking program. These tasks were performed to bring the A-7D into compliance with USAF aircraft structural integrity requirements (USAF, 1975).

KEYWORDS

Damage tolerance; fatigue; fracture mechanics; stress spectra development; fatigue damage tracking; structural integrity.

INTRODUCTION

The A-7D Damage Tolerance Assessment consisted of five primary tasks which were performed between September 1974 and January 1977. Each task is discussed with emphasis on methods which were developed during the program.

TABLE 1 A-7D Damage Tolerance Assessment Tasks

Task No.	Description
1	Preliminary Damage Tolerance Assessment
2	Initial Quality Assessment
3	Stress Spectra Development
4	Operational Limits and Inspection Requirements
5	Individual Aircraft Tracking Program

TASK 1: PRELIMINARY DAMAGE TOLERANCE ASSESSMENT

The purpose of this task was to select and evaluate potentially critical structural areas in the A-7D airframe. Candidate critical areas were systematically identified and screened. Data sources included the A-7D stress and fatigue analysis, static and fatigue test results, service experience, and non-destructive inspection data.

A review of the data sources (first screening) resulted in the identification of more than 200 potentially critical structural items. A second screening reduced this number to 100 items of primary structure which could potentially affect safety of flight. Each of these was assigned a permanent item number which it retained throughout the program. The remaining 100 items were discarded because they were not critical to flight safety.

A third screening consisted of an assessment of the degree of criticality of each candidate location. For example, areas in which a single failure would result in the loss of aircraft were considered more critical than areas which could sustain a single failure without loss of aircraft. Additional considerations included inspectability, material type and thickness, strength and fatigue design margins, crack growth rates, stress concentrations, load transfer, service experience and test data. Twenty locations which were judged sufficiently critical to warrant detailed analysis. Fifteen of these locations were in the wing center section and are shown in Fig. 1.

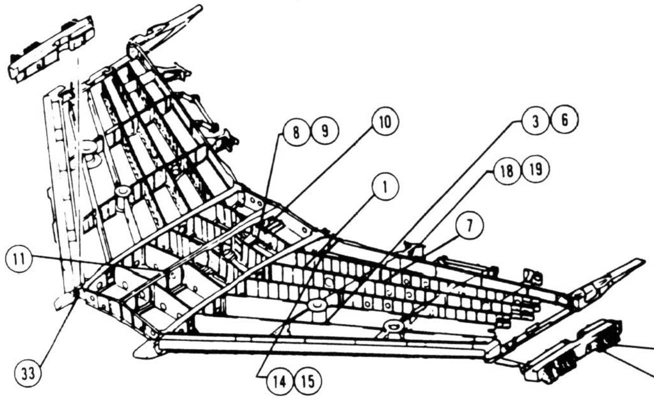

Fig. 1 Potentially critical A-7D wing center section locations.

TASK 2: INITIAL QUALITY ASSESSMENT

The purpose of this task was to assess the initial manufacturing quality of the A-7D structure and to determine the maximum size of initial flaws that could be expected in the fleet. Eight coupons containing a total of 44 holes were cut from the lower wing skin of a low-time (690 flight hours) A-7D. Each coupon was fatigue tested using a modified constant amplitude spectrum until flaws developed and propagated to failure. Each hole was broken open after failure to reveal subcritical cracks. Fractographic analysis traced the growth of each crack backwards in time to obtain its initial size. A total of 85 initial flaws were identified and fractographically measured. They resulted from both mechanical and chemical processing. After initial flaw sizes were determined, the data were statistically evaluated. The log normal probability distribution shown in Fig. 2 was estimated from the initial flaw size data. Seven hundred holes per aircraft are located in potentially critical areas, and the USAF has approximately 500 A-7D's. Therefore, the maximum flaw size in any one hole in the fleet was assumed to be that which occurred less than once in 350,000 holes. Based on the distribution in Fig. 2 and a 95% lower confidence bound, this flaw size was estimated to be 0.069 cm.

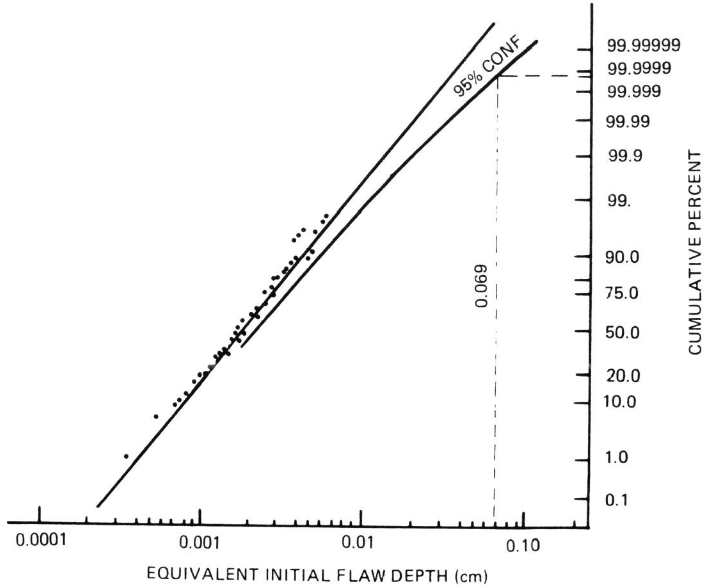

Fig. 2 Statistical distribution of initial manufacturing flaws.

TASK 3: STRESS SPECTRA DEVELOPMENT

The baseline stress spectra developed in this task were used in Task 4 to establish operational limits and inspection requirements for the 20 critical structural items identified in Task 1. The method used to develop the baseline spectra is presented in Fig. 3 (Sandlin, 1979; White 1977).

A-7D usage was based on a mix of the following USAF training missions: Transition Training, Instrument Training, Navigation Training, Weapons Delivery, Defensive Combat Maneuvering, Deployment/Refueling, and Search and Rescue.

The missions consisted of descriptive profiles defining "points in the sky" at which the pilot maneuvers the A-7D. However, to determine flight loads, more precise definitions of Mach number and altitude were needed for high activity segments. This information was obtained from recorded A-7D flight data. Initial gross weights were determined from the mission configuration descriptions; aircraft gross weights associated with Mach-altitude points were calculated by accounting for fuel consumption and stores ejections.

A-7D vertical load factor (n_z) spectra were based on specification data (USAF, 1971) modified by A-7D counting accelerometer (CA) data. The n_z spectra were defined for each mission segment: ascent, cruise, descent, loiter, air-ground, air-air, advanced transition, formation, and instruments and navigation. Each A-7D flight phase was correlated with one of the MIL-A-8866A n_z spectra.

The n_z spectra were applicable primarily to wing structure and had to be expanded to include additional airplane response parameters in order to derive stress spectra for all 20 critical items. Two-variable spectra were produced by multiplying n_z spectra by n_y (lateral load factor) probability distribution curves. The two-variable response spectrum formed the basis for expansion to an eight-variable response model. Pitching motion (q and $\dot{q}$) was correlated with n_z; rolling (p and $\dot{p}$) and yawing (r and $\dot{r}$) motions were correlated with n_y (Lauridia, 1979). Multichannel flight recorded data were used to derive cross-correlation and probability curves required to accomplish the expansion to eight variables (n_z, n_y, q, $\dot{q}$, p, $\dot{p}$, r, $\dot{r}$). Lauridia (1980) has generalized, expanded, and documented the methodology to generate multivariable spectra for any model fighter/attack aircraft.

A-7D flight-by-flight stress spectra were derived from thousands of load conditions resulting from combinations of weight, altitude, airspeed and aircraft response. A multivariable grid of loading conditions was established as a base for derivation of regression equations to predict the stress produced by each parametric combination. The A-7D data base consisted of 242 loading conditions selected to encompass the variations in the parameters which produce stress in each of the 20 critical items. The loads analysis consisted of determining net loads applied to the wing, fuselage, horizontal tail and vertical tail for each of the 242 conditions. Finite element (NASTRAN) math models were constructed for each aircraft component and maximum principal stresses were computed for the 20 critical items for each loading condition.

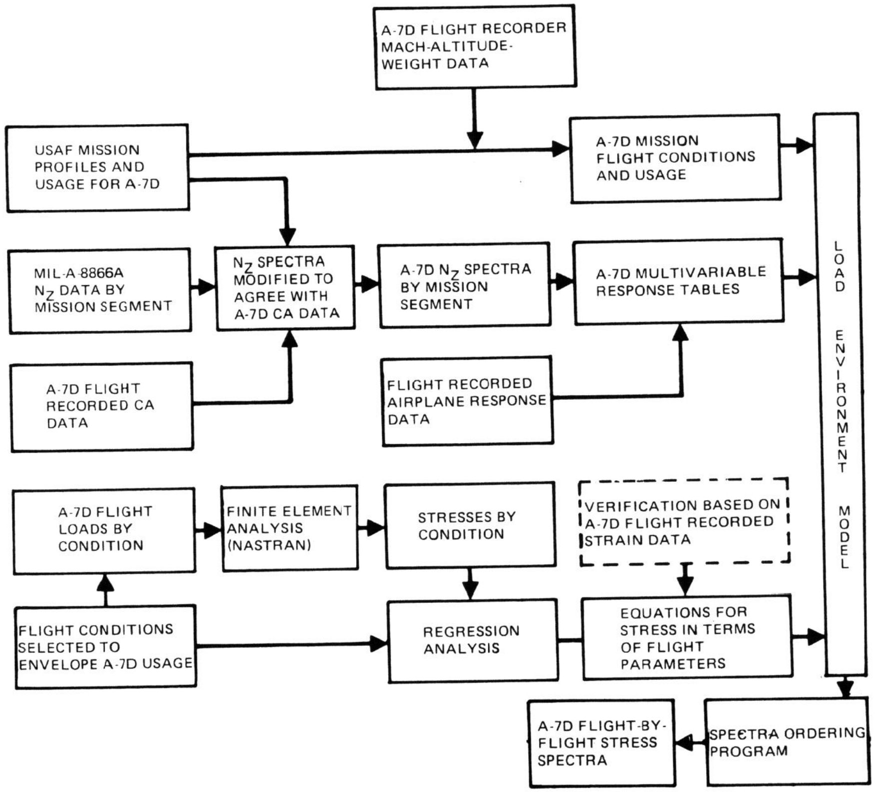

Fig. 3 Development of A-7D baseline stress spectra.

An equation for net stress, in terms of the flight parameters which produced the stress, was derived for each of the 20 critical items by use of a stepwise multiple regression analysis. The accuracy of the analytical stress equations was verified by comparisons with A-7D flight measured stresses (Lauridia, 1977).

The load environment model output consisted of spectra in the form of stress occurrences for selected stress bandwidths. These flight-by-flight, unordered spectra (based on a 1,000-hour mix of the USAF missions) were entered into a spectra ordering program which served as a link between the load environment model and the crack growth analysis model. All stresses in a flight were ordered into a low-high-low sequence and the flights were ordered into 1,000-flight-hour repeatable blocks. Stress values which occurred less than once per flight, but at least once per 1,000 hours, were added at random to the appropriate flight in the 1,000-hour block.

TASK 4: OPERATIONAL LIMITS AND INSPECTION REQUIREMENTS

The purpose of this task was to establish the operational limits and inspection requirements for the A-7D structure. This was accomplished by conducting crack growth analyses and tests for the critical areas selected in Task 1, using the results of the Task 2 initial flaw studies and the stress spectra developed in Task 3.

Operational limits were based upon conservative assumptions to prevent loss of the worst case aircraft. The operational limit was defined as the service usage interval beyond which potential failure of unrepaired structure could result in loss of aircraft. Calculation of the operational limit included the effects of initial flaw size and shape, rate of crack growth and critical crack size. A 0.127 cm quarter circular corner flaw was selected as the initial flaw. This size and shape was compatible with USAF (1974) initial flaw assumption requirements and was considered the largest flaw that would remain undetected during airframe fabrication. Since the largest flaw expected in any critical hole in the A-7D was 0.069 cm, selection of 0.127 cm as the initial flaw size provided an added measure of protection for the A-7D structure. The crack growth rate (da/dN) data selected for the analysis represented a severe environment. Although none of the A-7D critical areas were exposed to trapped water, the decision was made to use da/dN data equal to the numerical average of laboratory air environment and 100% water environment crack growth rates. Critical crack sizes were dependent on the residual strength of the structure. Residual strength was based on two requirements: (1) the structure must be capable of sustaining design limit load, and (2) the structure must be capable of sustaining the maximum load expected during the inspection period. The inspection period was defined as one-half the operational limit.

The methodology used for predicting crack growth was a cumulative crack growth approach based upon fracture mechanics. Starting with an initial crack size, shape, and orientation perpendicular to the direction of loading, crack growth behavior was predicted based on accumulation of increments caused by each load application. The computer routine EFFGRO (Szamossi, 1972) was used in this procedure.

The following crack tip stress intensity factor solutions were used for through-the-thickness and part-through cracks loaded in far-field tension:

Through-the-thickness, $$K_I = \sigma \sqrt{\pi a}\ \beta_T \qquad (1)$$

Part-through, $$K_I = 1.1\ \sigma \sqrt{\frac{\pi a}{Q}}\ \beta_T \qquad (2)$$

where Q is the flaw shape parameter which depends on crack aspect ratio, applied stress and yield strength.

The geometry correction factor β_T is the product of all individual geometric effects,

$$\beta_T = \Pi \ \beta_i \quad (i = 1, n) \tag{3}$$

where β_i are individual correction factors for geometric considerations such as finite width effects, a part-through crack approaching the back surface and a crack emanating from a fastener hole.

In most of the critical locations, load is transferred between structural components through fasteners. Therefore, load transfer and bearing stress effects must be accounted for when calculating the total stress intensity factor. The method of superposition was used to combine the effects of far-field stress and bearing stress, i.e.,

$$K_{I_{total}} = K_{I_{far\text{-}field}} + K_{I_{bearing}} \tag{4}$$

Far-field tension stress intensity was calculated as previously described. The bearing stress intensity was calculated using the solution of a surface loaded crack given by Paris (1964). Specialized stress intensity factors were developed for certain cases such as lugs or complex geometry in the presence of high load transfer. This was done through use of the cracked finite element routine FEABL (Orringer, 1972) and the experimental or "backtracking" stress intensity solution method (James, 1969).

The constant amplitude crack growth rate da/dN was obtained from previous programs and was characterized for use in crack growth analysis through the Forman (1967) equation. The Forman equation expresses da/dN as a function of ΔK while taking into account the effects of load ratio and crack instability as ΔK approaches K_c.

Since the load history used in the A-7D Damage Tolerance Assessment was a realistic, flight-by-flight spectrum with variable load amplitude, a requirement existed to account for crack growth retardation. The Wheeler (1970) crack growth model was used for this purpose. The Wheeler model uses the load interaction or plastic zone concept to characterize the crack tip residual stress state created by prior spectrum loads. If the currently applied load develops a plastic zone to or past one previously developed (greatest prior elastic-plastic interface), the growth increment associated with the current load is calculated using a steady-state (no retardation) growth rate equation. Conversely, retardation is assumed if the current load develops a plastic zone smaller than one which preceded it.

The Wheeler model accounts for retardation by operating directly on da/dN and reducing the constant amplitude crack growth rate. The crack growth rate under spectrum loading is computed using a shaping exponent, m, which is a data fitting parameter allowing the analysis to be correlated with test results, as explained by Wheeler (1970). Several analyses were conducted using various values for m until the best curve fit was obtained with specimen test data. This m value was then used in a final analysis with the proper geometric correction factors for the actual structural location and severe environment da/dN data.

The final element of the crack growth analysis was the determination of the critical stress intensity or fracture toughness. Fracture toughness data was obtained from previous programs for the materials involved. Crack length increments were accumulated with the application of stress cycles until the current crack length and minimum required residual strength load produced a stress intensity equal to the fracture toughness, which constituted the operational limit of the critical location.

Operational limits and inspection requirements were determined by applying the crack growth criteria and methodology to each of the 20 critical items. The results are summarized in Table 2. Nine items which had operational limits greater than 16,000 flight hours were discarded. The eleven critical items in Table 2 were grouped according to proximity in the structure and similarity of inspection intervals. As a result, seven critical areas which required periodic inspection and repair were identified. The inspection and repair intervals were adjusted to optimize maintenance resources. For example, inspection of Items No. 1, 9/10, and 55 required wing removal; therefore, these inspections were phased concurrently. The resulting inspection intervals, shown in Table 3, were shorter than the intervals shown in Table 2 except for Item No. 31/40 (wing fold rib lugs) and Item No. 55 (aft wing attachment lug). An engineering decision was made to waive the requirement for inspecting at one-half the operational limit due to the conservatism of the criteria used in the lug damage tolerance analysis.

TABLE 2 Operational Limits and Inspection Intervals

Item No.	Operational Limit (Flt. Hours)	Critical Crack Size (cm)	Inspection Interval (Flt. Hours)	Corresponding Crack Sizes (cm)
1	12,200	3.048	6,100	0.635
3	2,800	0.483	1,400	0.229
7	2,200	0.356	1,100	0.165
9	9,600	2.591	4,800	0.406
10	8,300	0.533	4,150	0.305
14	5,000	0.508	2,500	0.292
15	5,400	0.686	2,700	0.279
18	6,100	4.978	3,050	0.864
31/40	5,600	0.432	2,800	0.229
55	5,900	0.521	2,950	0.254

TABLE 3 Revised Inspection/Repair Intervals for Critical Items

Critical Item	Baseline Flight Hours						
	1,000	2,000	3,000	4,000	5,000	6,000	7,000
1				X			
3/7	X	X	X	X	X	X	X
9/10				X			
14/15		X		X		X	
18			X			X	
31/40			X			X	
55				X			

TASK 5: INDIVIDUAL AIRCRAFT TRACKING PROGRAM

The purpose of this task was to develop methodology (Lauridia, 1977) and a computer program (Sandlin, 1977) to monitor damage in individual A-7D aircraft. Procedures were developed which optimized the use of counting accelerometer (CA) data in scheduling inspections of the seven critical areas based upon individual A-7D aircraft usage.

In Task 4, operational limits and inspection intervals were established for baseline A-7D usage. Since usage varies from aircraft to aircraft, a method was required to relate actual aircraft usage to baseline usage. The approach selected was based upon solving an equation which directly relates individual aircraft n_z counts and flight hours to flight-by-flight crack growth damage at a reference location, translating damage at the reference location into the number of baseline spectrum flight hours required to experience the equivalent damage, and using the equivalent baseline spectrum flight hours to estimate crack growth damage at all the other critical locations in the structure (White, 1979).

Item No. 1 at Wing Station (W.S.) 32 (Fig. 1) was selected as the reference location for the damage tracking program because a large amount of flight recorded strain data for this location was available. A crack growth analysis of W.S. 32, using the baseline spectrum, resulted in a predicted operational limit of 12,200 flight hours. The operational limit was defined as the length of time it would take a crack to grow from 0.127 to 3.048 cm. Spectra were also developed for seven variations of "off-baseline" usage identified by the individual aircraft CA data. Actual A-7D flight-by-flight recorded strain data for W.S. 32 were manipulated to construct seven stress spectra which were compatible with the n_z spectra produced by the desired off-baseline usage. Order of occurrence for n_z and stress was preserved within each flight, while the flights were sequenced randomly.

Preflawed specimens representing the W.S. 32 location were tested to the off-baseline spectra. The fracture surfaces were analyzed fractographically and the crack growth curves were plotted. Test results were analytically matched using the Wheeler retardation model and the empirical shape factor, m. Final crack growth curves for W.S. 32 were obtained from EFFGRO by using the respective Wheeler-m values, assuming severe environmental da/dN data, and accounting for the actual geometry at W.S. 32.

For each crack growth curve, the value of t in flight hours at the operational limit (t_{OL}) was determined. The values of t_{OL}, in conjunction with the n_z counts at 5, 6, 7, and 8 g's that characterize the seven off-baseline test spectra, were used to determine an equation for t_{OL} in terms of normalized (to 1,000 hours) n_z counts. The equation for t_{OL} in terms of n_z counts was obtained by regression analysis and is shown below.

$$t_{OL} = 24,700 - 6.20\ N_{5g} - 2.70\ N_{6g} - 51.2\ N_{7g} + 101\ N_{8g} \qquad (5)$$

Error analysis showed that the equation predicted t_{OL} for the seven off-baseline spectra within 4% of the test values.

The regression equation for t_{OL} was used as the basis for monitoring damage in individual A-7D aircraft. Since all the crack growth curves had very similar shapes, the following approximation was assumed:

$$\frac{t_{BL}}{t_{OL_{BL}}} = \frac{t_{AP}}{t_{OL_{AP}}} \quad \text{or} \quad t_{BL} = \frac{t_{AP}\ t_{OL_{BL}}}{t_{OL_{AP}}} \qquad (6)$$

where: t_{BL} is baseline flight hours

$t_{OL_{BC}}$ is 12,200 hours (W.S. 32 operational limit for baseline usage)

t_{AP} is aircraft flight hours

$t_{OL_{AP}}$ is calculated by the regression equation

Rather than monitor equivalent baseline flight hours for individual A-7D aircraft, a decision was made to monitor a Damage Index (D.I.), where D.I. = 1 corresponds to 4,000 baseline flight hours.

$$\text{D.I.} = \frac{t_{BL}}{4,000} \qquad (7)$$

A computer program was written and is currently being used to calculate the D.I. for each A-7D aircraft periodically. The program automatically screens and edits CA data reported from USAF bases. Base averages are also used for projecting D.I.'s into the future. Operational limits are determined and inspections are scheduled for each A-7D based upon the output of the D.I. computer program.

Studies conducted by Gray (1978) further verified the approach for using cumulative n_z counts to monitor individual aircraft damage. Relating n_z counts to crack growth was based upon the ability to relate n_z to stress. A physical relationship between n_z and A-7D wing stress was identified, and a statistical relationship between n_z and stress was established for the A-7D horizontal tail and vertical tail (Lauridia, 1977). Since the current A-7D tracking method does not account for the effects of variations in unmonitored parameters such as gross weight, Mach number and altitude, an A-7D Structural Life History Recorder Program was initiated (Nichols, 1980). Its purpose is to measure strain at W.S. 32 in 20% of the A-7D fleet to sample the relationship between n_z and stress at that location, and modify the A-7D tracking program to account for variations in this relationship. Larson (1979) conducted independent studies and further validated the current method of predicting damage at the six structural areas remote from the A-7D reference structural location.

SUMMARY

The purpose of this assessment was to develop improved predictions of structural life operational limits using fracture mechanics theory and crack growth analysis, establish improved structural inspection requirements, identify structural modifications necessary to obtain an 8,000-flight-hour operational life and develop an individual aircraft structural damage tracking program. It was determined that the A-7D airframe is very damage tolerant and can be safely and economically operated for 8,000 flight hours.

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