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AIRCRAFT FATIGUE AND CRACK GROWTH CONSIDERING LOADS BY STRUCTURAL COMPONENT

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SUMMARY

The indisputable 1968 C-130 fatigue/crack growth data is reviewed to obtain additional useful information on fatigue and crack growth. The proven Load Environment Model concept derived empirically from F-105D multichannel recorder data is refined to a simpler method by going from 8 to 5 variables in the spectra without a decrease in accuracy. This approach provides the true fatigue/crack growth and load environment by structural component for both fatigue and strength design. Methods are presented for defining fatigue scatter and damage at crack initiation. These design tools and criteria may be used for both metal and composite aircraft structure.

INTRODUCTION

The main objective is to provide the best methods for fatigue/crack growth and load environment prediction. In most cases only one full scale fatigue test is conducted per aircraft type. The local loading on component elements can not be simulated even though shear, torsion and bending moment at a station may be in good agreement with theoretical or flight test values.

In 1967 many C-130 aircraft were flying in Southeast Asia with fatigue cracks (Yost, Ref. 1) at low risk due to their damage tolerant structural design. Aircraft were sent to be repaired when the cumulative cracks' length at a surface wing station was 10 inches or more. In about 1970 the aluminum 7075T6 center wing sections on the C-130s were replaced with a new structure due to many fatigue cracks in fleet aircraft. Wing station (WS) 120 center section crack data are used to give a better understanding of structural fatigue in aircraft.

The primary elements of the load environment may be broken down into ground-air-ground (GAG), gust, maneuver, buffet, taxi, jump takeoff and landing. Buffet loads may be superimposed on gust or maneuver and sometimes on both. Load spectra of most concern for fighter aircraft (Yost and Johnson, Ref. 2) are maneuver loads. Both gust and maneuver are very important for transport design for the flight domain. Current civil transport aircraft fly at higher speeds than earlier prop aircraft, thus they respond to a lower frequency portion of the gust Power Spectral Density or longer wavelength higher magnitude gusts (Yost et al., Ref. 3). Comparison of current versus past NASA *VGH* data shows this trend. Also, severe turbulence is avoided and the normal load factor (*Nz*) spectra contain buffet, gust, and maneuver. Military specifications in the MIL-A-8860 series have been updated as new data became available from test programs and fleet aircraft load recorder programs. Even though these data are Nz spectra which include buffet, gust, and maneuver, the breakdown to mission segment is of great value.

Ground operations are centered on landing gear design for sink rate, taxi response for usage and gear dynamic magnification factor (DMF). DMF for spin-up and spring-back loads may be greater in the Drop-Tower certification than during aircraft landings, so fatigue and strength design is conservative. Fatigue is accumulated on other aircraft components besides the landing gear and backup structure during ground operations.

A new method is presented for defining scatter in fatigue failures (Yost, Ref. 4). It is believed that the methods for prediction of loads (Ref. 2 and Yost, Ref. 5) and fatigue failures in metal structure are the same for composite structure.

SYMBOLS USED

Aa = angle of attack (deg)

AC = aerodynamic center (in)

Ax = roll acceleration (deg/s^2)

Ay = pitch acceleration (deg/s^2)

Az = yaw acceleration (deg/s^2)

B = sideslip angle (deg)

CG = center of gravity (in)

CL = crack length (in)

D = Miner's damage equation: $\Sigma(n/N) = 1$

Da = aileron position (deg)

De = elevator position (deg)

Di = damage at crack initiation

Dr = rudder position (deg)

FW = fuel weight (lb)

g = acceleration due to gravity (ft/s^2)

Ixx = roll moment of inertia (lb-s^2-ft)

Iyy = pitch moment of inertia (lb-s^2-ft)

Izz = yaw moment of inertia (lb-s^2-ft)

Kt = stress concentration factor

n = structure stress cycles

N = cycles read on S/N curve

Ny = lateral load factor (g)

Nz = normal load factor (g)

Q = dynamic pressure (psf)

Sa = stress amplitude (ksi)

Sm = mean stress (ksi)

S/N = stress versus cycles curve

STW = store weight (lb)

V = variation coefficient

VGH = velocity, g, altitude

W = gross weight (lb)

C-130 CRACK GROWTH DATA BASE

The C-130 data (Ref. 1) from 1967 and 1968 is the best, most complete large statistical sample source for fatigue and crack growth by mission of documented data for upper and lower wing surface. The in-flight fatigue test data cover nine different missions. These included a mild load environment long range cargo mission. One of the most severe load environment missions was low altitude with gust loads superimposed on the maneuvers. A very high percentage of these fleet aircraft had fatigue cracks as shown in Table 1.

The crack growth was monitored by measuring the crack length every two weeks. Accumulated crack growth data was added to the data base. The past history of individual aircraft was defined by taking records of missions flown per base and the flight hours accumulated at assigned bases.

Table 1. C-130 Fleet Aircraft Cracks in 1968

Total aircraft	619
Aircraft with fatigue cracks	345
Aircraft with fatigue cracks at WS 120	289
Aircraft repaired	81

A crack growth curve was derived from documented data with crack length on the ordinate and fatigue damage on the abscissa. This crack growth curve was used to convert the total crack length per upper or lower surface at WS 120 to fatigue damage on the left or right side. A new crack growth curve was obtained by least squares curve fit of fleet crack growth data and is presented in Figure 1 for upper and lower surfaces.

The actual fatigue damage on individual aircraft was obtained by taking the crack growth curve in Figure 1 and cumulative crack length per surface for converting to actual damage. Next a regression analysis was conducted using actual damage as the dependent variable and flight hours for each of the nine missions as independent variables. These analyses produced coefficients that were damage per flight hour for upper and lower surfaces at WS 120 for each of the nine missions. Validation of these coefficients was accomplished by using crack growth data from aircraft which had most or all of their usage on only one mission.

The actual damage per mission flight hour was used as a dependent variable to interrogate the fleet data with regression analysis for the damage source. Statistical sound parameters were used as independent variables in this analysis. These variables were: (% time in turbulence) (gust intensity factor)/ (mission/hour) for gust, (GAG/mission/hour) for GAG, (ratio of paved to unpaved runway roughness standard deviation) (number of landings/ mission/hour) for taxi. To define maneuver damage per hour, the missions having very low time in turbulence gave maneuver contribution as the equation constant, so this one unknown resulted in some iteration.

These empirical relations allow a definitive look at aircraft structural fatigue that is not available for any other aircraft.

AIRCRAFT FATIGUE LIFE

In order to define what portion of the aircraft life that is available after crack initiation we selected a 0.5-inch crack as the actual total life for our analysis. The two crack growth equations in Figure 1 were used in computing the actual damage per 0.03-inch and 0.5-inch crack lengths for the crack growth portion. This produces a crack growth portion of 26.7% for lower surface and 18.6% for upper surface. The average portion of fatigue life that is crack growth is about 23%. Since this statistical sample is so large, this information provides a better understanding of structural fatigue. One point we must make is that these aircraft had flaws which required the use of stress concentration factors (Kt) in fatigue analysis of 8 and 10.

C-130 SCATTER FACTORS

The Long Range Cargo was selected as the mildest and Sky-Hook as one of the most severe load environment missions to study scatter. Data was available on these two missions for crack initiation where the aircraft were almost single mission usage aircraft. Then 10,000 flight hours were computed as the mean flight hours to crack initiation from Figure 1 and the damage per flight hour, for the Long Range Cargo mission at WS 120 on the lower surface. The first crack initiations were about 5,000 flight hours, which allowed the calculation of 2 for the scatter factor. Using this same procedure on Sky-Hook the mean flight hours to crack initiation were 1,400 and first crack initiation was about 950 flight hours, so the computed scatter factor is 1.47. The crack initiation and crack growth curve for the first aircraft on each of these two missions are shown in Figure 2.

The more severe load/stress cyclic environment has less scatter, which is the same trend in stress versus cycles (S/N) curve raw data points. In an earlier paper (Ref. 4), we derived an equation to represent the scatter in S/N raw data points as a function of stress amplitude, mean stress and stress concentration factors. The natural log of the variation coefficient (Ln V) as the dependent variable in Equation 1 represents the scatter.

$$\mathrm{Ln}V = 1.56 + \frac{9.57}{Sa} + \frac{2.53}{Kt} - 0.0255Sm \qquad (1)$$

Data for this equation derivation was a very large number of 2024T3 and 7075T6 aluminum specimens. This coefficient is computed from Miner's Damage Equation (SUM $[n/N] = 1.0$) values.

THEORETICAL FATIGUE SCATTER ($\Sigma[n/N]$)

To help keep scatter to a minimum relative to the S/N curve effects, it is suggested that good statistical coverage of mean stress (Sm), stress amplitude(Sa), and stress concentration factors is needed for small specimens. The next step we propose is to derive an equation for these type data to represent the complete family of S/N curves for each material, as we have done. Other variables should be added to the equation for component specimens, loading complexity, and specimen size. Thus six component specimens per helicopter S/N curve and a large scatter factor would not be needed. Accurate load spectra are needed to reduce scatter in fatigue predictions. These spectra should be per mission segment for structural component location.

For each analysis location, the design stress concentration factor must be defined from use of finite element model and/or detailed stress analysis.

THEORETICAL CRACK GROWTH ANALYSIS SCATTER

The same accurate load spectra for fatigue analysis are needed for crack growth analysis predictions. The design stress concentration factors are also needed with detailed information about the analysis locations. Structural inspection of fleet aircraft is required to validate the crack growth predictions.

AIRCRAFT FATIGUE PREDICTIONS

Some of the many parameters to be considered for fatigue analysis are listed in Table 2. A scatter factor of about 40 is common for helicopter parts and 4 is used on most airplanes. The main objective is to keep the scatter to a minimum. Critical crack length is the best selection for fatigue to avoid high risk. Miner's damage equation is the best fatigue damage method to define crack initiation when corrections are included for scatter. The selection of the most severe mission for fatigue life calculations allows adequate life on any mission.

Table 2. Aircraft Fatigue Predictions

Scatter range:		
S/N curves	–	1 to 40
Loads	–	1 to 3
Design *Kt*	–	1 to 10
Define Fatigue Life:		
Select crack length		
Critical crack length		
Crack initiation		
Method of fatigue prediction:		
Miner's Damage Equation SUM(*n/N*)		
Crack growth analysis		
Select load environment:		
Most severe mission		
Mission mix		

A method is desired that provides the positive and negative load peaks for any selected location on the aircraft for design strength or fatigue analysis. The Load Environment Model (2) concept is capable of doing these things and has been computer programmed for the F-106, F-111, and A-7D aircraft. The initial model was eight variables and was derived with F-105D multichannel recorder data. The new method uses all the proven features of the above model minus the aircraft angular velocities and without any decrease in accuracy.

This new model has five variables and may need to go to six variables if the longitudinal load factor is important for the aircraft being analyzed. A step-by-step description of model derivation starts with a good statistical normal load factor spectra per mission segment that the model aircraft is expected to be flying during its usage life. The next requirement is multichannel recorder data from a similar type of aircraft, which includes angular acceleration about each of the three axes, plus Nz and lateral load factor (Ny). Then derive probability curves for Ny, roll acceleration (Ax), pitch acceleration (Ay), and yaw acceleration (Az) versus Nz. These probability curves are used to expand the Nz spectra per mission segment to a five-variable Load Environment Model per mission segment. Table 3 is taken from MIL-A-8866A (USAF) to show an example of Nz spectra per mission segment.

Table 3. Maneuver–Load–Factor Spectra A, F, TF Classes, Cumulative Occurrences per 1,000 Flight Hours by Mission Segment

Nz	Cruise	Air–Ground	Air–Air
			Positive
2.0	10,000	175,000	300,000
3.0	2,500	100,000	150,000
4.0	400	40,000	50,000
5.0	1	10,000	13,000
6.0		1,500	2,500
7.0		200	900
8.0		15	180
9.0		1	60
10.0			15
			Negative
0.5			44,000
−0.5			1,200
−1.5			60
−2.5			1

To make use of the model, regression equations are needed for control surface positions and load/stress regression equations for selected structural component locations. Next, use the most severe mission to compute load spectra with the load equations and the Load Environment Model. These location spectra are to be used to compute strength required and fatigue analysis for design. Each component spectrum will be different and will allow the minimum structural weight to be defined for strength and fatigue. Strength design limit loads are the maximum values for one life on the spectra. The flowchart in Figure 3 shows how the Load Environment Model fits in with other tasks.

LOAD REGRESSION EQUATIONS

Load equations are required to go from a Load Environment Model to a structural component location spectrum. Load equations for the F-111 were complex due to the wing variable sweep, but were high-quality since they were derived from recorded flight test data. The tiltrotor on the V-22 presented a different set of problems in obtaining valid load equations even before flight test. Data for the F-106 equations were very limited. Equations for the A-7D were very good, but consideration of Buffet was not included.

Detailed stress analysis and finite element models (FEM) must be validated, plus all critical control points for strength and fatigue must be correctly selected.

The independent variables must be logical for the component analysis locations selected. Goodness of fit for the computed equations is best measured with the standard error and correlation coefficient. Two independent variables which have high correlation with each other should not be used in the same equation. The maximum number of variables per equation should be about ten. As variables enter an equation, the standard error decreases and the correlation coefficient increases to approach 1. The point in equation derivation where the correlation coefficient does not increase when a new variable is added may be the stopping place for that equation. Also, another stopping point is when the standard error does not decrease when a new variable is added to the equation.

A short list of independent variables is as follows: *Nz*, *Ny*, *Ax*, *Ay*, *Az*, *Q*, *W*, *Iyy*, *Izz*, *Ixx*, *STW*, *FW*, *AC*, and *CG*. A list of control surface equations and other dependent variables with their crossproduct variables are listed in Table 4.

A minimum of thirty theoretical conditions are needed to cover the edges of the aircraft operational envelope and have a good statistical base for regression analysis.

COMPUTING WEIGHTED SCATTER FACTORS

It was shown earlier that scatter increases as N goes to the higher values on an *S/N* curve. So the scatter should be multiplied by damage at *N* values as a weighting consideration. This process is shown best by Equation 2.

Table 4. Key Regression Equation Variables

Dependent Variable	Independent Variable 1	2	3	4
Aileron position (Da)	NyW/Q	$IxxAx/Q$	$IzzAz/Q$	
Rudder position (Dr)	$IxxAx/Q$	B	$IzzAz/Q$	NyW/Q
Elevator position (De)	$IyyAy/Q$	$NzW(AC-CG)/Q$	NzW/Q	
Sideslip angle (B)	NyW/Q	$IzzAz/Q$		
Angle of attack (Aa)	NzW/Q	$NzW(AC-CG)$		
Wing stress	NzW	$NzSTW$	$NzFW$	
Vertical stab. stress	BQ	DrQ	Ax	
Horizontal stab. stress	AaQ	DeQ	Ay	
Wing/fuselage lug stress	DaQ	BQ	NzW	Ay

$$Vt = \frac{\sum\left[\left(\frac{n}{Nj}\right)Vj\right]}{\sum\left(\frac{n}{Nj}\right)} \qquad (2)$$

The total weighted variation coefficient (Vt) is computed for the analysis locations on the S/N curve at j values.

FATIGUE LIFE EQUATION

The fatigue life damage at crack initiation (Di) is defined with Equation 3.

$$Di = \sum\left(\frac{n}{N}\right) - \frac{3Vt}{100} \qquad (3)$$

An adjustment factor for crack initiation should be obtained from the full-scale cyclic test and is identified as Fc. When cracks start showing up in fleet aircraft, Fc should be replaced with Ff, which represents the adjustment factor for crack initiation in the fleet.

For all locations or elements with no alternate load path, not damage tolerant, noninspectable joints, critical crack length is so short it cannot be seen by visible inspection and helicopter rotor system parts, then use a safety factor ($Fs = 2$) for these designs. In addition, a safety factor of 1.5 should be used for the complete aircraft when using a low-confidence load environment, to help prevent fatigue modifications.

Weight reductions during design could affect fatigue life. The most common problem is an aircraft which is operated in a more severe load environment than that for which it was designed. In computing upper surface damage for flight, change the mean stress from negative to positive to agree with C-130 fleet damage data and to allow use of current *S/N* curves. Fatigue prediction correction factors are presented in References (1) and (4).

CRACK GROWTH ANALYSIS

Any fleet aircraft with cracks should be inspected periodically to validate crack growth predictions and define its location on the damage versus crack growth plot. The damage versus crack growth plot is the best way to define where individual aircraft are in the scatter distribution and their Ff. Residual strength for cracked structure is very important and analysis/test at the end of the full-scale cyclic test is required to define risk of aircraft flight with cracks. Crack growth analysis should begin on a fleet airplane when cracks are found during inspection.

The Kt of a crack is about 18, so crack growth is predictable and high loads cause retardation as they do in fatigue. Critical crack length is defined as the length at which theoretical failure will occur when limit load is applied. The crack growth analysis and fatigue analysis can be used as tools to define inspection requirements. Repair of areas which have cracks approaching critical length and their total number is basic data for defining the aircraft economic life. These methods provide a direct path for updating design criteria for strength and fatigue.

FLIGHT LOADS RECORDING

A large sample of fleet Nz recorder data is needed with altitude and airspeed to allow breakdown to mission segment spectra per aircraft type and mission. Also, a smaller sample of five-variable (Nz, Ny, Ax, Ay, and Az) multichannel recorder data is needed to expand mission segment Nz spectra to a five-variable spectra for the same or similar aircraft type.

HUMAN FACTORS

The current design of aircraft to a normal load factor is not good when fleet aircraft exceed this limit value more than once per each aircraft's life. The normal load factor limit value is mainly important for wing design. The methods presented in this paper allow the design per fleet aircraft usage for crew and passenger safety, since about 80% of accidents are related to human factors. This method will greatly decrease risk by designing all structural components for the load environment experienced during the fleet usage.

AGING AIRCRAFT

The C-130 data gives a detailed view of what is occurring in aging aircraft structure and how inspections are needed to avoid unacceptable risks. These inspections allow those aircraft which do not have high risk cracking to be used much longer, even though they may have high flight hours.

CONCLUSIONS

In review of the interrogation of the C-130 fleet damage sources, ground-air-ground was almost twice as great for the lower surface as it was for the upper surface. Taxi damage was high for only the upper surface on the Shuttle mission, which included operations on unpaved runways. Gust damage was about the same for upper and lower surfaces for all nine missions. Missions with a high percentage of maneuver damage were Sky-Hook, Support, and Proficiency Training where the upper surface damage was a little higher than the lower surface damage. However, the damage rate per flight hour on the Support mission was low, thus the evasive maneuvers for missiles and ground fire would account for this maneuver damage.

The most extreme case is finding the 2–4*g* maneuvers on the Air Drop mission caused a small percentage of maneuver negative damage for the lower surface. Keep in mind the C-130 has a 2.5*g* design limit load factor. Extremely high loads such as C-130 air drops changed the shape of flaws and reduced the *Kt* only in the wing lower surface.

Compression load cycles on the wing upper surface must be considered during design. Actual damage in the C-130 fleet was linear and had low scatter.

The methods for design loads in this paper are needed to prevent pilots from taking their airplanes to nearly twice the design limit load, as occurred with the F-86's in Korea. Thus the load limiter concept may put the pilot in great danger in air-to-air combat.

Strength and fatigue design for composite structure can make use of these methods with consideration that flaws exist and in some cases they can be removed. In addition, the damaged hot wet specimens represent the worst case for structural conditions in the fleet. Risk of adhesive and resins to debond with age, temperature, and humidity effects is another problem designers must face.

Most of the NASA-airplane large statistical samples of normal load factor data include maneuvers with gust cycles superimposed on them. USAF recording programs for aircraft response also do not separate gust from maneuver, but do have a wider range of aircraft types.

There are many advantages for deriving one equation per material type and its family of *S/N* curves. First, the scatter and error due to interpolation are cut to a minimum. Also, high-cycle conditions such as buffet and helicopter rotor loads can be easily handled.

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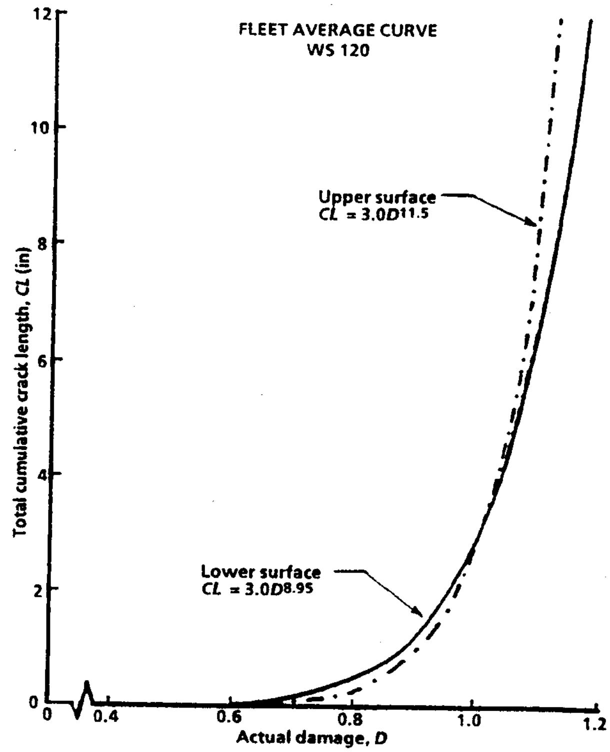

Figure 1. C-130 fatigue crack propagation.

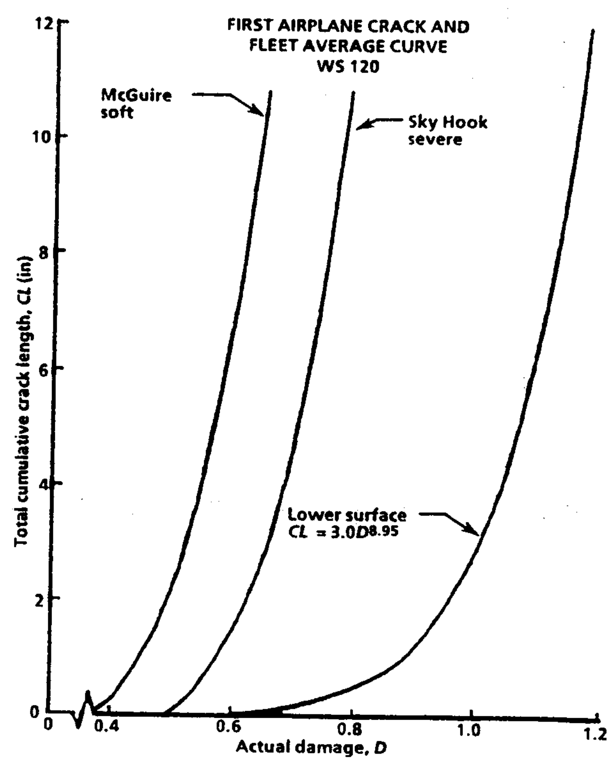

Figure 2. C-130 fatigue crack propagation: 1st aircraft on two missions.

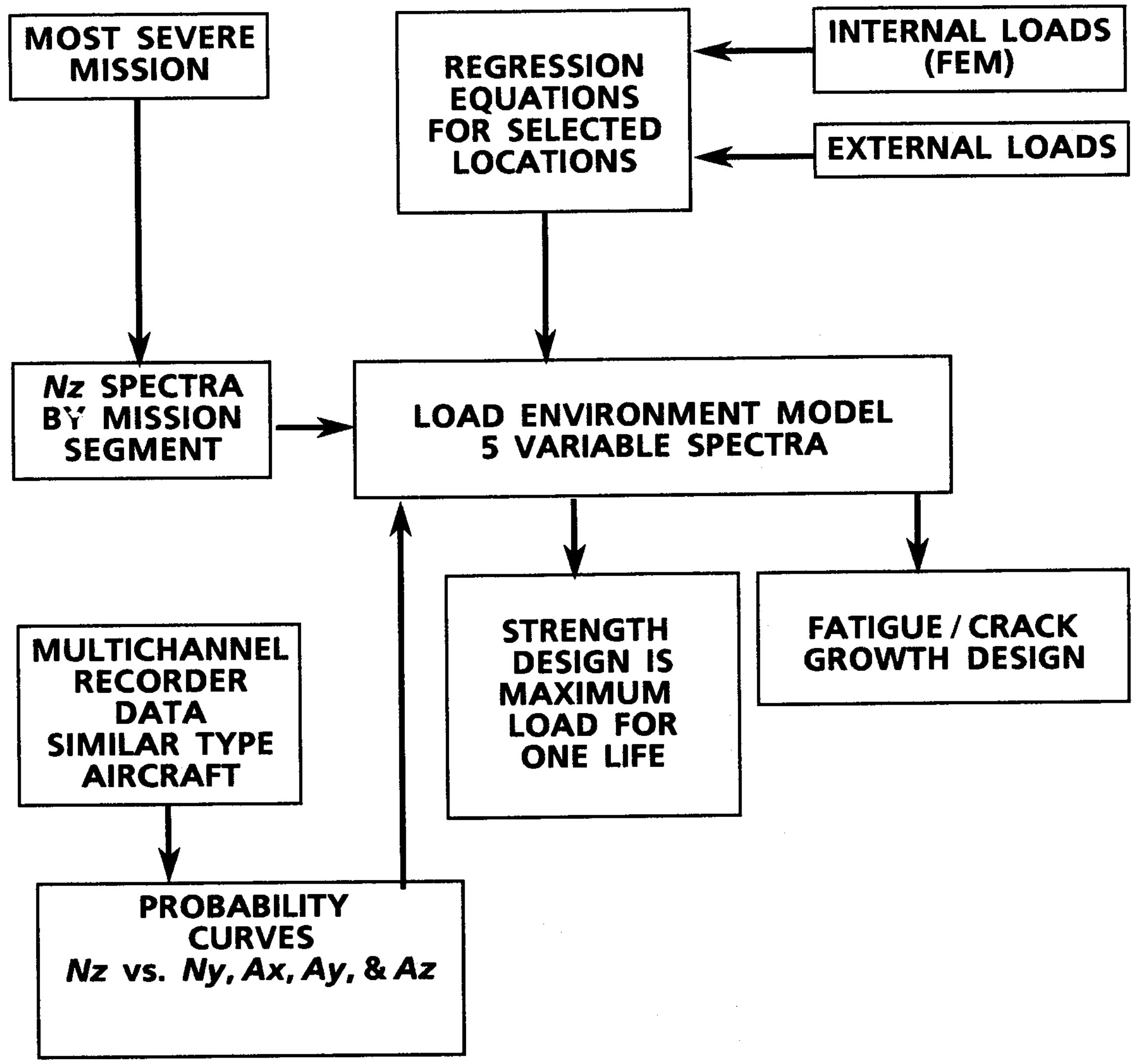

Figure 3. Design loads per structural component.