Strength and Fatigue Loads Computed with a Load-Environment Model

JAMES D. YOST* AND GUINN S. JOHNSON†
Convair Aerospace Division, Fort Worth, Texas

Documented oscillograph recorded multichannel data (primarily F-105D) was used to develop an eight variable mathematical environment model of fighter aircraft response per Mil Spec 8866A mission segments. The F-106 fighter was selected for demonstration of the model. F-106 detailed flight test results, fatigue cyclic test data and 3770 flight hours of normal service response data were available to facilitate evaluation of the model. These data made it possible to empirically compute structural component loads for comparison with F-106 design values. On the basis of these analyses, new design criteria for strength and fatigue are recommended.

Nomenclature

L = shear load, lb
n = load factor, g
p = roll velocity, deg/sec
$\dot{p}$ = roll acceleration, deg/sec^2
q = pitch velocity, deg/sec
$\dot{q}$ = pitch acceleration, deg/sec^2
Q = dynamic pressure, PSF
r = yaw velocity, deg/sec
$\dot{r}$ = yaw acceleration, deg/sec^2
s = at center of gravity minus the wing center of pressure (fuselage stations), in.
V = velocity, mph
V-G = parameter recording, velocity, and normal acceleration
VGH = parameter recording velocity, normal acceleration, and altitude
W = gross weight, lb
α = angle-of-attack, deg
δ = control surface displacement, deg

Subscripts

A = aileron
e = elevon
E = elevator
L = left
R = right
RD = rudder
t = true value (velocity)
v = vertical stabilizer root
w = wing root
Y = lateral (load factor)
Z = normal (load factor)

Introduction

It has been known for several years that V-G and VGH data are inadequate for defining structural strength and fatigue design criteria. To provide design load criteria with statistical verification it was recommended by a special NACA Subcommittee on Aircraft Loads (1954) that statistical loads programs be expanded to measure time histories of eight parameters (three linear accelerations, three angular accelerations, and airspeed and altitude). Actual implementation of the recommendations demanded: 1) developing a reliable and relatively inexpensive recorder which would meet military design specifications; 2) devising an efficient and economically feasible data reduction system; and 3) deriving methods of establishing static and fatigue design criteria by statistical inference.

Several types of recorders have been developed which are suitable for gathering the multivariable statistical loads data. Therefore, a data reduction system and data format for converting the multiparameter data into a useful form for deriving design criteria are required. Solutions to these problems together with proposals for new design criteria are the topics covered in this paper.

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*Senior Structures Engineer, F-111 Loads and Criteria Group. Associate AIAA.
†Project Structures Engineer, F-111 Loads and Criteria Group.

Method Description and Logic

A Multivariable Load-Environment Model (mathematical model) has been developed to define the statistical loads imposed upon fighter-type aircraft during each mission segment. This model has approximately the same vertical acceleration statistics as the proposed MIL-A-8866A. In addition, the model contains seven other variables correlated to vertical acceleration during normal operations. This model was derived basically from F-105D multivariable recorder data which was recorded during Tactical Air Command aircraft peace time[1] and Southeast Asia[2] operations. Additional elements were added in order to test the model and derive criteria, such as F-106 control surface position and structural component load equations, and F-106 typical mission and utilization.

With the addition of these items, the model becomes applicable only to the F-106 (Figs. 1 and 2). Hereafter, in dealing with the elements of this F-106 Multivariable Load-Environment Model it will be referred to as F-106 MVLEM or F-106 Model.

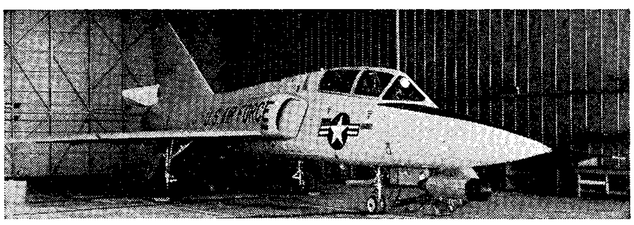

Fig. 1 F-106B.

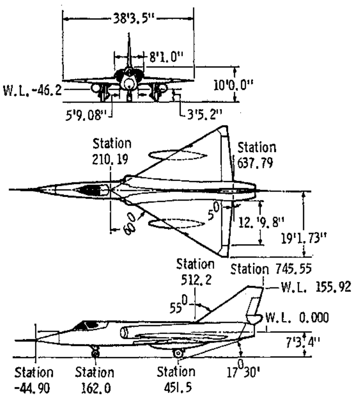

Fig. 2 F-106A three view.

Response Environment Model for Fighter Aircraft

A very limited amount of multivariable fighter aircraft response data are available in documented form. In some cases those multivariable data samples documented[1,2] have been compared with large VGH samples from the same flight operations to extend their usefulness where agreement exists.

Response probability curves exist in a number of reports[1-4] for various aircraft. The relation between response parameters is difficult to define for an eight-variable statistical data sample. Correlation is presented in the referenced reports between individual parameters from peacetime[1] (258 flight hr) and Southeast Asia combat[2] (314 flight hr) fighter aircraft operations. The eight parameters recorded in these two programs were altitude, airspeed, longitudinal acceleration, lateral acceleration, normal acceleration, angular roll rate, angular pitch rate, and angular yaw rate. Three angular accelerations were computed from the angular rates.

All of the previously noted reports were used in the derivation of the Response Environment Model. The eight variables and number of peaks compose a multivariable block or row in the Response Environment Model as shown in Table 1 with typical envelope values for 1000 flight hr as they exist in the Air-To-Air Mission segment.

The derivation of the model started with n_Z occurrences per band (band width of .5 g's) for the n_Z range of minus 3.0 g's to plus 12.0 g's, for individual mission segments. The n_Z occurrences per band were further distributed into n_Y bands (band width 0.1 g's) with an empirical n_Y probability curve (-1.5 g's to $+1.5$ g's) derived from recorded data. This two variable response spectrum was basic data for expansion to an eight variable response model (multivariable block form).

Pitch velocity and acceleration data were analyzed with respect to n_z magnitude to determine the probability of their exceeding the threshold bands ($q = -5.0$ to $+5.0$, $\dot{q} = -15.0$ to $+15.0$). These cross correlation probability curves in conjunction with the probability curves for a given q and $\dot{q}$ magnitude above threshold were used to expand the occurrences per n_Y, n_Z blocks.

The other four variables ($p, \dot{p}, r, \dot{r}$) were found to correlate with lateral acceleration (n_Y). Therefore, their probability of exceeding the threshold band was defined with test data similar to the method described for q and $\dot{q}$. Empirical cross-correlation and probability curves were derived for expanding the model to include these four variables ($p, \dot{p}, r, \dot{r}$). Further detail on band widths and cross-correlation may be extracted by the reader from the basic data.[1,2]

The Response Environment Model consists of seven mission segments (in multivariable block form, Table 1) which are listed in Table 2. The number of amplitude bands for each variable and the possible total combinations required that detail consideration and analysis be made. A maximum of 30 bands for each variable was selected as adequate with the possible total combinations of 30 to the 8th power for each mission segment. However, during analysis of recorded data[1,2] it was determined that the number of multivariable blocks shown in Table 2 would be sufficient to cover the response ranges.

The minimum number of occurrences (peaks) was limited to one per four airplane lives (16,000 flight hr) for establishing a multivariable block. This logic was very useful in limiting the total number of multivariable blocks without seriously sacrificing the number of possible combinations of variable magnitudes.

A brief outline seems in order at this point to show the expansion of the model. The first step is to add angle of attack and control surface position columns to the matrix (model) per mission segment, keeping in mind that each row of the matrix is a multivariable block. To obtain these parameters, a set of equations is required plus time spent at a given set of mission mix conditions making up the airplane usage. The condition data includes dynamic pressure, airplane gross weight, true velocity, and c.g. location. The structural load equations are used to add load columns to the matrix. Once the loads are included in the model, load spectra can be computed for each condition as well as for the total usage of an aircraft. Detailed coverage of this step by step procedure follows.

Table 1 Air-to-air mission segment

n_Z	n_Y	r	$\dot{r}$	p	$\dot{p}$	q	$\dot{q}$	No. Peaks
−1.5	0	0	0	0	0	−5	60	1.6307
+0.5	0.2	5	−10	−30	−80	0	15	3.993
1.0	0.9	−5	40	−30	−280	0	0	0.1590
1.0	0.8	5	30	−30	40	0	0	0.2479
1.0	−0.9	−5	−20	30	280	0	0	0.3534
1.0	−1.2	5	−50	30	400	0	0	0.106
4.0	0.2	−5	10	−30	−80	10	−45	1.1495
4.0	−0.4	5	−10	30	80	10	−15	0.0855
6.5	0.1	0	0	0	0	5	0	6.3223
7.0	0	0	0	0	0	10	−15	12.6126
7.5	−0.1	0	0	0	0	10	−15	1.6900

Table 2 Response environment model elements

Mission segment	Number of multivariable block
Ascent	274
Air-to-air (maneuver)	837
Special weapons delivery	781
Air-to-ground	924
Cruise	296
Descent	508
Loiter	425

Control Surface Position Equation Derivation

F-106 flight-test data[5] was the basic information input into a multivariable regression analysis-computer program for determining control surface position equations. In addition to the equations derived for control surface displacement, an equation for angle of attack was derived from flight-test data

$$\alpha = 0.479 + 961.\ n_Z/Q \tag{1}$$

The surface position equations which were computed are as follows:

$$\delta_A = 0.411 - 21.8\, p/V_t - 5.76\, \dot{p}/Q + 3800\, n_Z/Q \tag{2}$$

$$\delta_E = 3.22 + 0.0000013\, s\, n_Z W - 1280.\ n_Z/Q - 37.8\, \dot{q}/Q \tag{3}$$

$$\delta_{RD} = 2.2 + 227.\ r/V_t + 7.95\, \dot{r}/Q + 5.62\, \dot{p}/Q \tag{4}$$

A set of grid points was selected to provide coverage of the complete range of response parameters for symmetric and asymmetric flight-test demonstration maneuvers. Thus, some bias exists at least for Eq. (4) where a constant (2.2) was computed for rudder displacement from points selected on the time histories. No attempt was made to further refine these equations after final output from the computer.

Structural Load-Equation Derivation

The load equations were determined by the same methods as the control surface position equations. In some cases, the plotted response time histories had low-parameter sensitivity, thus resulting in greater efforts required to determine a good equation [Eq. (4)]. The equations for wing (5), elevon (6), rudder (7), and vertical stabilizer (8) loads are listed below

$$L_{WL} = -1563. + 11725.\, n_Z - 0.27\, \delta_E Q - 2.7\, \delta_A Q \quad (5)$$

$$L_{eL} = 330. + 1.17\, \delta_E Q - 1.35\, \delta_A Q \quad (6)$$

$$L_{RD} = -165. - 0.0667\, \delta_A Q + 1130.\, n_Y + 0.131\, \delta_{RD} Q - 7.4\, \alpha n_Y \quad (7)$$

$$L_V = 205. + 14100.\, n_Y - 0.651\, \delta_A Q + 913.\, \alpha n_Y - 0.0768\, \delta_{RD} Q \quad (8)$$

These empirical control surface position and load equations are derived directly from documented flight-test results. It is suggested that this approach is sound if the degree of goodness of fit for these equations is acceptable.

Equation Validity

The multivariable regression analysis computer program selects the variables and cross products of variables which provide the best equation for defining the dependent variable (load, control position, etc.). This program uses a least-squares fit, therefore, variation in both independent and dependent variables is desirable to obtain a good equation. Standard error and percent of residual squared explained were two indicators used in defining goodness of fit for equations. Table 3 shows the quality of the equations derived.

The residual is the difference between the actual values (original flight-test data input to computer program) and predicted values from Eqs. (1–8). The standard error of Table 3 is computed for the predicted values with the standard equation (which is a function of residuals and number of input conditions).

It was noticed during the derivation of the equations that considerable improvements in fit could be obtained with a greater number of conditions, high sensitivity, additional variables in each equation, and elimination of data errors. Wing, elevon, and vertical stabilizer load-equation validity are presented in Figs. 3, 4, and 5, respectively. The data points fall close to the perfect fit line (solid line on figures with a 45° slope).

Table 3 Equation goodness of fit

Equation	Dependent variable	Standard error	Percent of residual squared explained
(1)	α	0.914°	94.7
(2)	δ_A	0.929°	96.0
(3)	δ_E	1.934°	88.9
(4)	δ_{RD}	2.74°	76.3
(5)	L_{WL}	4143.0 lb	98.9
(6)	L_{eL}	1982.0 lb	93.4
(7)	L_{RD}	782.0 lb	73.3
(8)	L_V	2399.0 lb	95.2

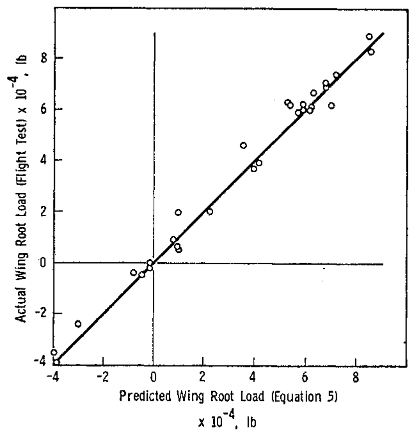

Fig. 3 F-106 wing root load, actual vs predicted.

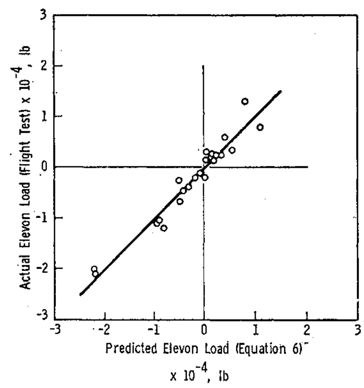

Fig. 4 F-106 elevon load, actual vs predicted.

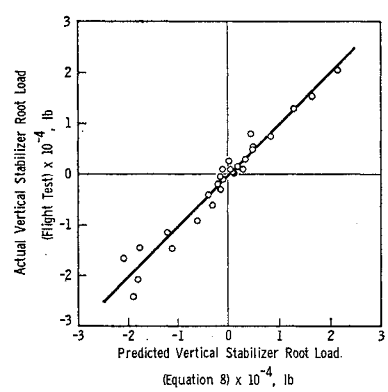

Fig. 5 F-106 vertical stabilizer root load actual vs predicted.

F-106A Typical Mission

A computer program was written to accept mission profile data, control surface-position equations, load equations, and response environment model for fighter aircraft. A review of papers[6,7,8] which presented ideas on connecting response parameter statistics and airplane characteristics to produce load statistics was conducted before the methods were selected and the computer program was written.

The F-106 was selected as an example due to available information on mission utilization,[9] flight-test data,[5] VGH data,[9] etc. In addition, combat[10] air-to-air type missions were studied in defining the typical F-106 air-to-air intercept mission shown in Fig. 6. The adding of the F-106 typical mission then converted the computer program to F-106 MVLEM (F-106 MODEL).

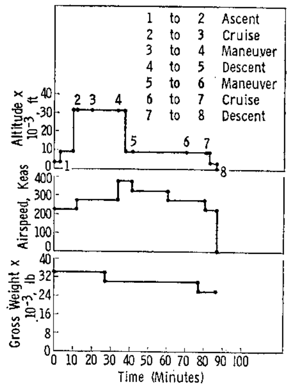

Fig. 6 Typical F-106 mission intercept air to air.

Load-Environment Model Validity

Normal load factor data[9] were available in sufficient quantity (3770 flight hours, VGH) to provide a good statistical representation of F-106 experience. The F-106 model, F-106 utilization, F-106 typical mission (Fig. 6), and F-106 equations were used to compute a normal load-factor spectra for one airplane life (4000 flight hours). A comparison of the recorded data and model results are shown in Fig. 7. The model results are more severe, which indicates that the proposed MIL-A-8866A spectra are more severe than the n_Z environment the F-106A experienced.

A detailed look at the elements included in computations of the composite n_Z spectra of Fig. 7 is in order. Let us emphasize that the F-106 data was not broken down by mission segment. Time utilization by band was available for altitude, airspeed, and gross weight. The sequence and time for mission segments were derived from other aircraft on similar missions which were broken out by mission segment time. However, the authors believe that a rerun of these analyses with Intercept and Normal Descent mission segment instead of Air-to-Air (Maneuver Fig. 6) and Descent Mission Segments (proposed MIL-A-8866A) will likely give much better agreement than presented in Fig. 7.

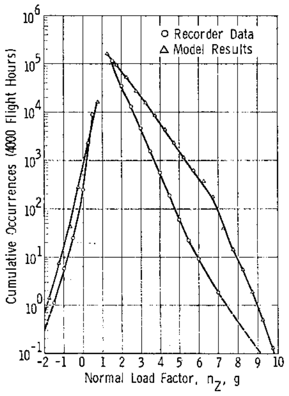

Fig. 7 F-106 cumulative occurrences of normal load factor.

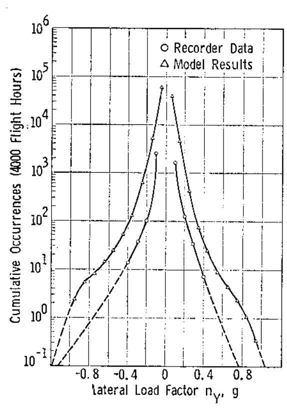

Fig. 8 F-106 cumulative occurrences of lateral load factor.

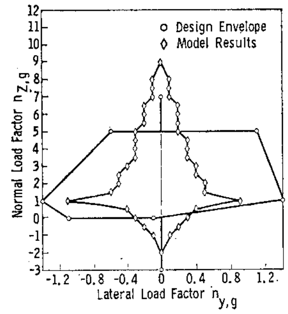

Fig. 9 Design envelope vs model results for one life (normal load factor—lateral load factor).

The lateral load-factor data is presented in a similar manner in Fig. 8 with like conclusions. The basic reason for this result is that in the model the lateral load factor is a probability distribution of each level (amplitude band) of normal load factor.

Normal and Lateral Load-Factor Envelope

The envelope of model results for normal and lateral load factor are shown in Fig. 9 along with the F-106A design envelope[5] for the same parameters. The envelope of model results are for one occurrence per airplane life.

The design envelope (Fig. 9) circles enclose the tested or validated envelope, with normal load factors of plus $7.0g$ and $-3.0g$ tested conditions. The area where the model results seriously exceed the design envelope is above $7.0g$.

The model results (Fig. 9, maximum $n_Y = -1.1$) for lateral load factor fall between the recorder data (Fig. 8, maximum $n_Y = -0.75$ at one occurrence per life) and the design (Fig. 9, $n_Y = \pm 1.4$ at $n_Z = 1.0$).

The normal load-factor model results exceed both the design (Fig. 9, $n_Z = 7$ at $n_Y = 0$) and recorded data (Fig. 7, $n_Z = 7.5$ at one occurrence per life) for positive values. However, for negative values of n_Z, the design ($n_Z = -3.0$) is greater than model results (Fig. 9, $n_Z = -2$ at $n_Y = 0$) and recorded data (Fig. 7, $n_Z = -1.6$ for one life) is less than model results.

Analysis of Design Loads for Strength and Fatigue Purposes

Normal load factor is the main variable in the equation for computing wing root loads. Figure 10 shows that the model approach produces a maximum F-106 wing-root load for one life which is much greater than the design load. Again, we note that the actual F-106 normal load-factor data were less severe than the model (Fig. 7) representing MIL-A-8866A mission segments.

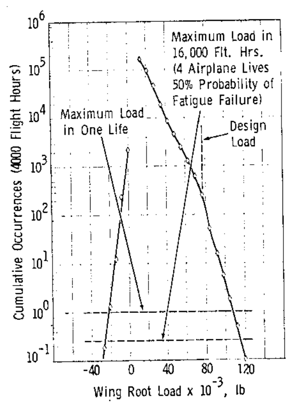

Fig. 10 Cumulative occurrences of wing root load per one airplane life.

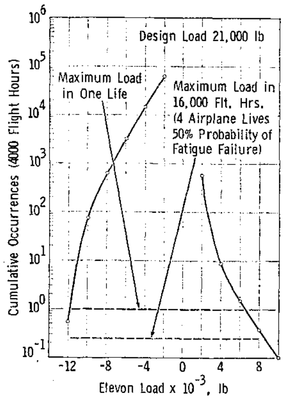

Fig. 11 Cumulative occurrences of elevon loads per one airplane life.

In Fig. 11, the F-106 cumulative positive and negative load occurrences are presented for the elevon per one life or about 2700 flights of the typical mission of Fig. 6. These load spectra were computed from the multivariable blocks with the fifteen conditions of the typical mission. Note that the maximum load for one life is a negative 12,000 lb and the design load is $\pm$21,000 lb.

Vertical stabilizer loads may be computed with the model directly without any of the currently used assumptions. The vertical stabilizer cumulative load occurrences presented in Fig. 12 are more severe than actual F-106 fleet aircraft experience. Since in the model the vertical stabilizer load equation has n_Y as the main variable and the model n_Y in Fig. 8 is greater than actual fleet F-106 n_Y experience, this means that under actual F-106 operating conditions the maximum load for one life would be less than the design load instead of giving close agreement as shown in Fig. 12.

It is quite common to use a factor of four in current fatigue analyses. Miner's equation at a damage value of 1.0 corresponds to four airplane lives. Fatigue failures are distributed about the damage of 1.0 such that no failures occur before a damage value of 0.25 (1.0 divided by the scatter factor of 4.0).

In this report, we are not attempting to bring in fleet size for rational probability of strength or fatigue failure. The methods presented are applicable to rational probability for maneuver loads. However, the ideas promoted are centered around a maximum load for four lives, consideration of scatter in load spectra (variation coefficients to account for scatter), and a safety factor.

These plots of cumulative occurrences of loads make possible consideration of both one life and four lives for one occurrence of a maximum load (structural strength considerations). Table 4 shows the limit design load values and these design data in tabular form.

Equation (9) provides a new method of defining strength safety factors with the use of an additional 10% for load distribution variation. A standard variation coefficient may

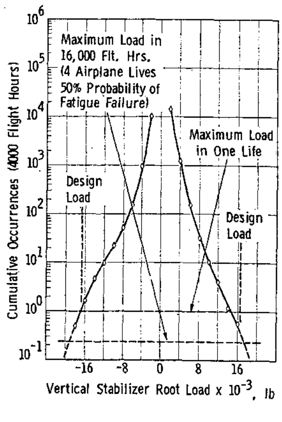

Fig. 12 Cumulative occurrences of vertical stabilizer root loads per one airplane life.

Table 4 Design loads for strength

Load	Limit design load	Maximum load One life	Maximum load Four lives	1.5 limit design load	1.1 maximum load four lives	Safety factor Eq. (9)
L_{WL}	77500	108000	115000	116000	127000	1.18
L_{eL}	21000	11800	12100	31500	13300	1.13
L_V	16600	16800	19000	24900	20900	1.24
					Avg. S. F. =	1.18

be applied instead of the 10% (empirical factor, 1.1) used in Eq. (9). A refined model is suggested for computing ultimate design strength loads with a standard variation coefficient, safety factor of 1.25, and maximum load for four airplane lives for all fighter aircraft structural components (Eq. 10).

Safety factor

$$= 1.1 \text{ maximum load four lives/maximum load one life} \tag{9}$$

$$\text{Ultimate design load} = (\text{variation coefficient})\,(1.25)\,(\text{maximum load in four lives}) \tag{10}$$

One objective in proposing new methods[11] for computing ultimate design loads is to stimulate thought by engineers for needed improvements. It was recognized that the model computed spectra were for the average fighter aircraft on a given mission segment. Therefore, individual aircraft spectra would be scattered about this average fighter-aircraft spectra and a variation coefficient would be needed to account for this scatter.

We are using four airplane lives as a link between structural strength life and structural fatigue life where a 50% probability of fatigue failure exists as noted in Figs. 10, 11, and 12. Our logic here is that the probability of structural strength failure should be less than the probability of fatigue failure.

Use of the model concept will assure close agreement between cyclic test fatigue load spectra and load spectra experienced by operational aircraft. The F-106 was exposed to cyclic testing over four lifetimes[12] of the selected load spectrum without major modification.

Table 5 shows the positive symmetrical maneuver spectrum (200 equivalent hours) used for the first two lives on the F-106 in percent of design limit load.

Cracks were found at 500 and 2100 equivalent flight hours on inspections before the 16,000 equivalent flight hr (four lives). The wing was cycled to a total of 20,000 equivalent flight hours and inspected with a number of rivets failed. The fuselage had several significant fatigue cracks by the final inspection at 18,000 equivalent flight hr. The authors consider the F-106 (from summary of crack data) superior to other aircraft for compatible load environment and fatigue design in the period before 1958. The fatigue design criteria proposed in this paper takes into consideration the advancements in fatigue design, F-106 data relative to the average aircraft, growth capability for operations in a more severe load environment than design, safe-life structure with no similar joint cyclic test data, fail-safe structure, safe-life

Table 5 Positive symmetrical maneuver spectrum

Maximum	Minimum	Number of cycles
32.3	14.5	1296
45.4	14.5	304 or 305
57.8	14.5	33
70.1	14.5	6 or 7
88.7	14.5	1

structure with adequate cyclic test data, and utilization of the load-environment model for realistic cyclic loading history for the complete aircraft structure.

Also, these criteria make allowance for the numerous occurrences (Fig. 10) of wing loads just below the design limit load which could propagate small cracks to critical length easily on safe life aircraft structural fatigue designs. Had the areas which had fatigue cracks on the F-106 fatigue cyclic test been safe-life design, the specimen may have had a catastrophic failure before four lives of cycling. However, we consider the advancements and current experience in structural design would eliminate any serious failures even in untested joints before 14,000 equivalent flight hr.

Therefore, a scatter factor of six is recommended for any joint that does not have similar specimen cyclic test data available, is shown to be a critical fatigue point using load-environment model type analysis, and a significant growth in severity of load-environment (fatigue life potential) capability is desired. Where specimen test data is available the scatter factor of four should be used on safe life structure design if load-environment model analysis is used with no considerations for aircraft utilization in a more severe load environment. However, fail-safe aircraft should not be penalized if they are successfully cyclic tested to two lives of spectrum produced from a representative model (suggested scatter factor 2.0).

Method Discussion

Current MIL-SPEC updating efforts (proposed MIL-A-8866A) have resulted in improved criteria for fatigue loads spectra development. However, since only normal acceleration statistics are included in the proposed MIL-A-8866A it is primarily applicable to development of wing loads spectra, and assumptions must be made for computing horizontal stabilizer, aft fuselage, and vertical stabilizer load spectra. The Multivariable Load-Environment Model provides a build-up of the load environment of the airplane in terms of the eight variables, segregated according to mission segment. Therefore, strength and fatigue loadings can be computed for operational and/or design stage aircraft by knowing usage (time per mission segment, gross weight, and dynamic pressure) and equations for converting the model data to structural loads. The equations (control surface position and load) can be computed from wind-tunnel model test data, analytical data, or from flight-loads test data.

Concluding Remarks

The model can be refined as additional multivariable data becomes available (F-111 data is currently being recorded) for fighter-type aircraft. The load equations can be computed empirically for both analytical and test data with a high degree of accuracy (or any reasonable accuracy level desired). The methods described in this paper are applicable to all aircraft including low-design load-factor aircraft such as bomber and transports as well as high-design, load-factor aircraft such as fighters. New criteria are suggested for establishing design loads for strength and fatigue considerations.

Another feature of the methods presented in this paper is that they enable the designer to establish control surface sizes and displacements required to perform the design missions and mission-mix utilization. This feature is possible because the model contains the maximum response values per mission segment from which the loads are computed. This means that lower-design load factors, lower structural weight and more efficient structural designs with higher confidence levels may be produced by using these methods.

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