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Methodology for Implementing Fracture Mechanics in Global Structural Design of Aircraft

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The analysis and design criteria of fracture mechanics are investigated for implementation with the automated structural optimization system (ASTROS) global optimization design tool. The main focus is the optimal design of aircraft wing panels by applying fracture mechanics design criteria within the global finite element model. This effort consists of four main phases: investigation of fracture mechanics analysis methods and design criteria, formulation of a computational technique consistent with global optimization requirements, integration of the technique into the ASTROS design tool, and demonstration of the results.

Introduction

THE advent of computational optimization tools has changed the way modern aircraft are designed. Recognizing the potential benefits of optimization, the U.S. Air Force sponsored the development of the automated structural optimization system (ASTROS) (Ref. 1) for use by government, academia, and commercial aerospace companies. ASTROS integrates several mathematical models of a system (finite element, aerodynamic panel, control system, and design models) in a modern software architecture. The standard ASTROS procedure minimizes structural weight subject to constraints on system performance, including natural frequencies, stresses or strains from static loads and aerodynamic loads, and aeroelastic performance. However, the implementation of optimization in such a complex and multidisciplined problem as aircraft design remains a challenge.[2] As is often the case in computational methods, modeling techniques are one of those challenges.

Despite the amazing advances in computer capabilities in recent years, modeling every stringer, rivet, and cutout of an aircraft remains impractical in computational optimization. Therefore, coarser global models are used to optimize the overall aircraft design and more detailed design is conducted after the fact. However, if the detailed local design cannot be achieved within the global constraints, the original global optimization may be invalidated. This may result in inaccuracies requiring modification of the global model and a new optimization. Obviously, this is not desirable considering the cost and time factors ever-present in today's aircraft programs.

The more accuracy that can be injected into the global design (without a disproportionate increase in computational requirements), the more easily the local design can be achieved within the global constraints. Therefore, the interaction between local and global design must be investigated to identify design variables that may produce the best increase in accuracy for the least expense. One design issue that plays a major role in both local and global aircraft design is damage tolerance.[3]

Fracture mechanics is the field of engineering that studies the tolerance of a structure to damage. The objective of this study is to demonstrate the use of conventional fracture mechanics analysis to establish fatigue criteria in global optimization. Another goal is to evaluate the interaction between the local modeling for fatigue criteria and global design. Therefore, a review of the basic equations of fatigue analysis is presented next to establish the groundwork for the methodology to be employed.

Fracture Mechanics

A crack in a structure can be stressed in three different modes: 1) opening, 2) sliding, and 3) tearing modes. Mode 1 is the most significant, especially for the fatigue of shell-type structures used in aircraft wing panel design. Therefore, only mode 1 will be considered in this paper. The stress intensity factor K characterizes the crack tip stress and can be calculated from the following equation:

$$K_I = \beta\sigma\sqrt{\pi a} \tag{1}$$

where σ is the far field stress, a is the crack length, and β is a dimensionless factor dependent on geometry, crack size, and other factors. A subscript corresponding to the appropriate mode is used to identify the direction of the stress intensity factor. Solutions to specific crack geometries can be found in Refs. 4 and 5. Analogous to an ultimate stress, the fracture toughness K_{Ic} of a material is the limit of the stress intensity factor. Although fracture toughness is viewed as a material property, caution must be used because it is dependent upon the material stress state (plane stress vs plane strain) and, therefore, the thickness and cold working of the material.

For a constant amplitude stress and a given material, the crack propagation can be predicted by relating the stress intensity factor to the change in crack length per cycle. Although many equations have been used to represent the da/dN vs K relationship, one commonly used equation was proposed by Walker[6]:

$$\frac{\mathrm{d}a}{\mathrm{d}N} = C[K_{\max}(1 - R)^m]^n \tag{2}$$

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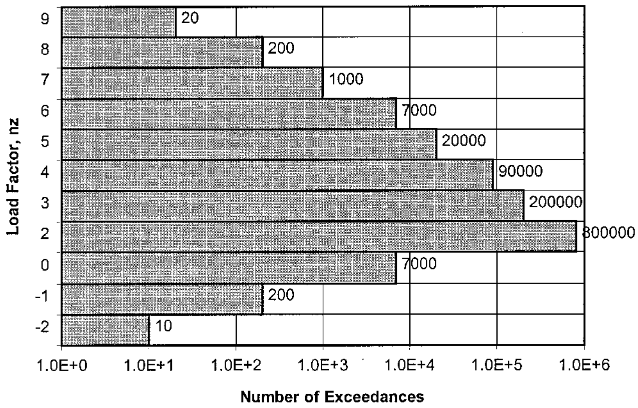

Fig. 1 Composite load-exceedance spectrum for fighter aircraft.

where a is the crack length, $K_{\max}$ is the stress intensity for the maximum stress, R is the stress ratio ($\sigma_{\min}/\sigma_{\max}$), and C, m, and n are material constants.

By integrating Eq. (2), a crack growth prediction of crack length vs cycle can be calculated. When the stress intensity factor equals the material fracture toughness, fracture occurs, therefore

$$K_{Ic} = \beta\sigma\sqrt{\pi a} \tag{3}$$

where a_c is the critical crack length (crack length where failure occurs).

In most real-life situations the stress cycling is variable and not a constant amplitude. Several attempts have been made to develop semiempirical retardation models that account for the sequential effects of residual stresses and crack closure. The method that will be used in this paper is the generalized Willenborg retardation model[7] and is dependent on a parameter called the shutoff ratio, which is the ratio of overload maximum stress to the subsequent minimum stress required to stop further crack growth.

Unlike conventional fatigue analysis, i.e., Miner's Rule-type calculations, fracture mechanics-based fatigue analysis is dependent on the order of the stress history. However, when any retardation model is used, the sensitivity of the crack prediction to stress history sequence increases significantly. Therefore, careful thought must be put into the methodology used to sequence the stresses.

As part of the U.S. Air Force Aerospace Structural Integrity Program,[8] actual in-service usage data of military aircraft have been accumulated. The monitoring program has produced load-exceedance spectra for different segments of aircraft missions. As seen in Fig. 1, the spectrum is defined as the number of times a load factor is exceeded per 1000 flight hours for a given mission segment. Giessler et al.[9] outline the methodology for using these load-exceedance spectra to develop an aircraft's design usage. Their guidelines are followed in developing the design usage applied to wing panels in this study.

Aircraft Wing Panels

For most modern aircraft, fatigue criteria play a vital role in the design of wing panels. The first step in analyzing the panel is to develop a stress history for the panel. This is typically accomplished by using data provided by the load and stress groups to convert the aircraft's design usage into a stress history, normalized by the maximum stress. The panel design and normalized stress history are then used to conduct a fracture mechanics analysis. The maximum allowable stress is the highest stress that can multiply the normalized stress history without exceeding the fatigue design criteria. It is found by iterating around the fracture analysis until the life of the panel converges to the design criteria for a given stress history. As a result, the maximum stress on a panel is not as important for fatigue analysis as the relative quantity of higher and lower stresses in the stress history.

Although the preceding analysis could be accomplished for every wing panel, allowables obtained from critical panels are usually used as allowables for all similar panels. In addition, design changes that significantly affect the stress history or panel design require that the iterative fracture analysis be repeated.

Approach

Objectives

The first objective is to demonstrate a method of linking the ASTROS program to a fracture mechanics prediction code to develop maximum allowable stresses for aircraft wing panels. The second objective is to investigate how local modeling of the panels affects the accuracy of the fracture mechanics analysis. A demonstration case was selected to obtain trends based on the effects of different panel features. The following panel features were examined: panel location, crack type, material type, panel shape, panel stiffeners, and panel thickness. The third objective is to investigate the optimal global design variables and their sensitivity to the fracture mechanics analysis. Thus, possible benefits to including this procedure within a global optimization loop can be identified. The demonstration case was used to obtain trends based on the effects of panel thickness and stress history.

Method

A computer program, Usage, was developed to define the aircraft's design usage (or life) in terms of ASTROS load case numbers. The aircraft life is described by a series of flights that typically consist of 1000 flight hours. These flight hours are broken up into blocks, missions, and mission segments (Fig. 2). A mission segment is defined by a load-exceedance spectrum, and each load level of the spectrum is identified with an ASTROS load case number. The order of flights, blocks, missions, and segments is defined by user input; however, the actual load cases within each segment are randomly sequenced. Because the design usage is completely independent of panel design, it only needs to be conducted once for all panels. In this manner, the entire aircraft load history can be defined in terms of ASTROS load case numbers.

A single ASTROS stress analysis (with multiple load cases) provides the data to convert the load history into panel stress histories. ASTROS allows a user to define a panel as a group of one or more (four-noded, isoparametric plate) finite elements. It will be assumed that the panel definitions made in the ASTROS model correspond to the physical wing panels on the aircraft. Therefore, geometric and stress data for a wing panel can be obtained directly from the ASTROS database.

A second computer program, Panel, uses as input the aircraft load history (defined by Usage as ASTROS load cases) and the results of the single ASTROS analysis. In addition, the user specifies the panel, design criteria, material properties, and

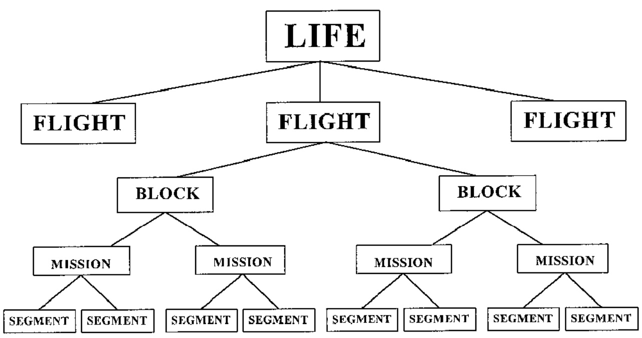

Fig. 2 Aircraft design usage.

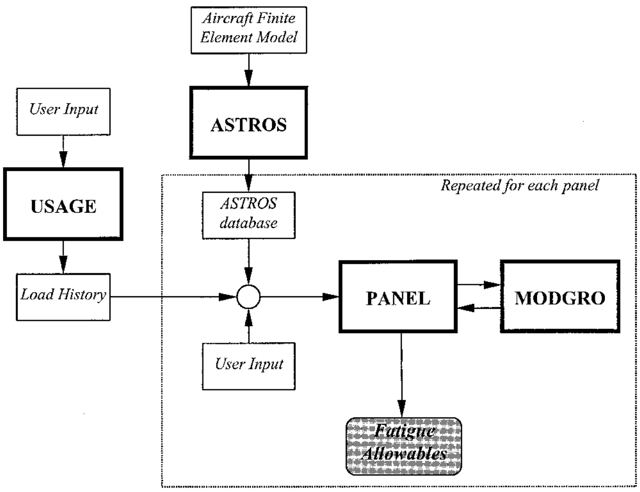

Fig. 3 Program flow.

crack definition. The ASTROS database is used to obtain the current panel geometry (length, width, and thickness) and the stresses on the panel for each load case in the load history. The resulting panel stress history is normalized by the maximum stress. These data are used to generate an input file for a government-developed crack growth analysis program called MODGRO.[10] The MODGRO program is iterated until the maximum stress allowable is obtained. The program flow is depicted in Fig. 3. Once the ASTROS model is updated, the ASTROS optimization can be performed using the new fatigue stress allowables.

The MODGRO program determines solutions to Eq. (1) for specific crack geometries. In addition, it is also possible for the user to input tabular data of β vs crack length (or a $\beta_{\text{correction}}$ multiplication factor vs crack length). The material properties can be specified by the Walker equation [Eq. (2)] or in the form of tabular test data. The $\mathrm{d}a/\mathrm{d}N$ vs K relationship [Eq. (2) or tabular data] is numerically integrated according to the generalized Willenborg retardation model (Ref. 7) using the stress intensity solutions from Eq. (1). The process is continued until a specified crack length is reached or until fracture as defined by Eq. (3).

In the current approach, the MODGRO program is iterated until a maximum allowable stress is found that satisfies the fatigue criterion. The user defines this criterion as a multiple of the aircraft life the panel must sustain without exceeding a user-specified crack length. If a crack length is not given, then the critical crack length calculated from Eq. (3) is assumed to be the criterion. The aircraft component is assumed to have some initial damage resulting from the fabrication process that is just under the detectable size of the inspection technique used. A common fatigue criterion in aircraft design is that the panel must survive two aircraft lives without fracture with an assumed initial flaw size of 0.635 cm (or a 0.127-cm crack at a fastener hole).[3] The iteration process continues until the number of stress cycles resulting from the fracture analysis converges to the number of cycles defined by the user (multiple of the aircraft life). The percent difference between the calculated and specified number of stress cycles is used as the convergence criterion and can be set by the user.

Results

A finite element model of an F-16 fighter aircraft was selected as the demonstration case for the trade studies. Wing panel features were independently varied to establish their effect on the overall aircraft design. Throughout these trade studies, the percentage change in fatigue stress allowable was used as the figure of merit. This is consistent with aircraft design because a change in the stress allowable has a direct effect on aircraft weight—the most critical design parameter. In addition, using the same figure of merit for all of the design variables permitted direct comparison of their effects on the design.

Every aircraft design is unique and, therefore, the results from the trade studies on the demonstration case cannot be directly applied to other designs. However, the relative magnitudes of these trades do provide fundamental insight into the damage-tolerant design of aircraft wing panels. These data can be used to establish research and design priorities concerning wing panel design.

Description of Demonstration Case

An existing ASTROS finite element model of an F-16 aircraft[11] was selected as the demonstration case. The model was selected primarily because the modeling details of the wing box are consistent with the level of detail typically found in a global optimization model (Fig. 4). In fact, the model was originally created for a wing skin optimization study. In addition, multiple load cases were available and well documented.

The finite element model included the six critical design load conditions for the F-16C wing. Although limited in Mach/altitude and weapons configurations, the load cases are a good mixture of maneuver types, including symmetric and unsymmetric maneuvers at both positive and negative load factors.

In the global model, the finite elements are not grouped or identified with individual wing panels. However, wing thickness design variables were defined as groups of finite elements in the current ASTROS design study model. These design variable groups were adopted as the definition of wing panels for this study.

It is important to understand that the intention is not to conduct an accurate damage tolerance analysis for the actual F-16 aircraft. Rather, the objective is to use this demonstration case as a realistic global model to investigate the behavior of damage tolerance analysis as it pertains to aircraft wing panel design. Therefore, it is more important that the wing panel features examined in this study are indicative of typical aircraft design, and less important that they accurately represent the F-16 aircraft portrayed by this particular model.

The number of maneuvers available in the demonstration model is insufficient to accurately represent the true design usage of the F-16. However, the diverse maneuver types in the model can be used to develop a design usage realistic enough for this study. Because the load cases are the critical design conditions for the F-16 wing, they represent the extremes in different loading situations. Therefore, these few load cases should sufficiently demonstrate the variation in stresses in the wing at different flight conditions.

Given the limited Mach/altitude conditions available in the model, it was futile to develop a design usage based on a

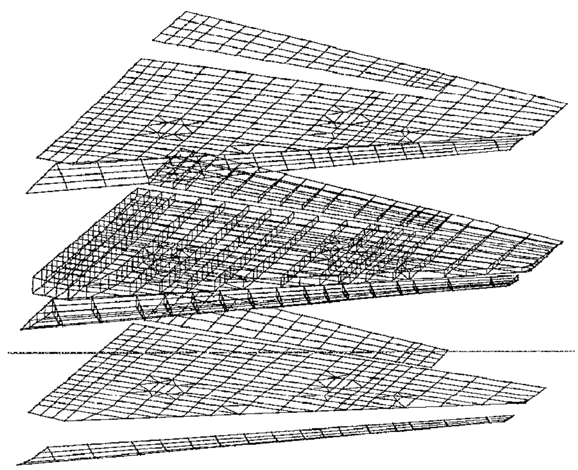

Fig. 4 Finite element model of the F-16 wing.

complex arrangement of blocks, missions, and mission segments as shown in Fig. 2. Therefore, a composite load-exceedance spectrum was used to develop the design usage (Fig. 1). The composite load-exceedance spectrum is based on an average of all load-exceedance spectra for aircraft in the fighter and attack categories.[12]

A 10,000-flight-hour load history was developed with the Usage program and the design usage information. Because the design usage is defined in terms of ASTROS load cases, it applies to all wing panels. Therefore, this load history was used as the basis for all of the trade studies to follow.

Design Criteria

The slow-crack-growth design criterion for uninspectable structures was used for the wing panel design studies. The U.S. Air Force criteria require that an uninspectable panel must sustain a stress history equivalent to twice the life of the aircraft while maintaining a residual strength greater than the maximum stress that would occur in 20 lifetimes.[3] Because the load cases in the demonstration model are the critical design load cases, the maximum panel stress from these load cases was used as the residual strength, regardless of whether it occurred in the load history.

Unless otherwise stated, the initial crack was assumed to be a 6.35-mm (0.25-in.) center through-crack in an infinite plate. The crack length is consistent with the uninspectable slow-crack-growth criterion. In addition, the crack geometry has the simplest fracture mechanics solution and requires no panel dimensions. Although this is an oversimplification of an actual wing panel, it provides an ideal baseline from which to make design study variations.

Material Properties

The Walker equation [Eq. (2)] was used to represent the material's fatigue properties. Although the Walker equation is typically applied to several segments of the da/dN vs K curve, only one segment was used to facilitate trade studies involving the fatigue properties. Typical values for the fatigue properties of aluminum were obtained for Eq. (2) from data presented in Ref. 13: $C = 3.5(10^{-10})$, $m = -0.3$, $n = 3.8$, K_{max} is in ksi-in.$^{1/2}$, da/dN is in in./cycles; and $K_{Ic} = 45.0$ ksi-in.$^{1/2}$, $K_{th} = 4.0$ ksi-in.$^{1/2}$, $\sigma y = 66.0$ ksi.

The Willenborg model[7] was used as the retardation model in this study. In this model, a parameter can be specified that defines the magnitude of the peak stress ratio required to cause crack growth shutoff. This parameter, called the shutoff ratio, generally varies between 2.0 and 3.0. A shutoff ratio value of 2.5 was selected as the baseline for the trade studies.

Wing Panel

A wing panel near the wing root was selected for the baseline case in the trade studies. A wing root panel was chosen because it is typically the focus of attention in most damage-tolerance analysis. This wing panel is modeled as 12 quadrilateral finite elements in the ASTROS model; a finite element near the panel's center was selected as the panel's master element. Therefore, this panel's stress history was determined based solely on the finite element solution for the master element.

Design Trade Studies

Material Properties

The slope of the da/dN vs K curve (n in the Walker equation) was varied from +30 to −30% of the nominal value. The change in fatigue stress allowable from the allowable for the original slope is shown in Fig. 5. As expected, the fatigue stress allowable displayed a high sensitivity to the slope of the Walker equation. Obviously, results like these justify the intense amount of research that has been conducted to accurately characterize the fatigue properties of a material.

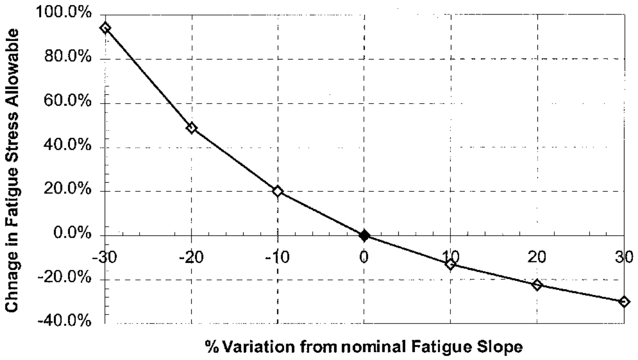

Fig. 5 Effect of fatigue slope on fatigue stress allowable. Nominal fatigue slope = 3.8.

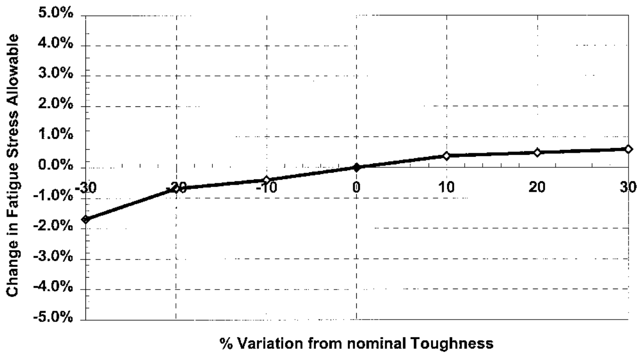

Fig. 6 Effect of toughness on fatigue stress allowable. Nominal toughness = 45 ksi-in.$^{1/2}$.

The toughness of a material (K_{Ic}) determines the level of stress intensity the material can sustain before catastrophic failure. The material toughness was varied from +30 to −30% of the nominal value. Figure 6 demonstrates that in this situation the material toughness actually has little effect on the fatigue stress allowable. The reason for the low sensitivity is that the majority of the stress cycles occur when the crack is very small. By the time the crack is approaching the point of instability, the crack propagation rate is very high. Therefore, the increase or decrease in the number of stress cycles caused by a change in material toughness is small relative to the number of stress cycles that occurred at the smaller crack sizes. Conversely, if a different crack geometry or stress condition caused higher crack rates earlier in the crack life, the material toughness would play a larger role.

Panel Geometry

Up to this point, the wing panel has been assumed to be an infinite plate. When the plate is considered to have finite width and length, the stress intensity factor increases. Theoretically, the finite panel solution must approach the infinite plate solution as the crack size decreases. Because the majority of a crack's life is spent at the smaller crack sizes, the effect of finite length and width should be somewhat diminished.

Assuming the panel length to be infinite, the width was varied from 15.24 cm (6.0 in.) to 121.92 cm (48.0 in.). In Fig. 7, the change in the fatigue stress allowable from the infinite plate solution is plotted vs plate width. Even for very small panel widths, the fatigue stress allowable did not change more than 3%. The actual panel width of 30.48 cm (12.0 in.) resulted in less than a 1% change in the fatigue stress allowable.

Similar results were obtained when the width was fixed at 30.48 cm (12.0 in.) and the length-to-width ratio was varied from 0.4 to infinity. As seen in Fig. 8, the maximum change in fatigue stress allowable was less than 2.5% for the F-16 wing panel near the wing root. The actual panel dimensions

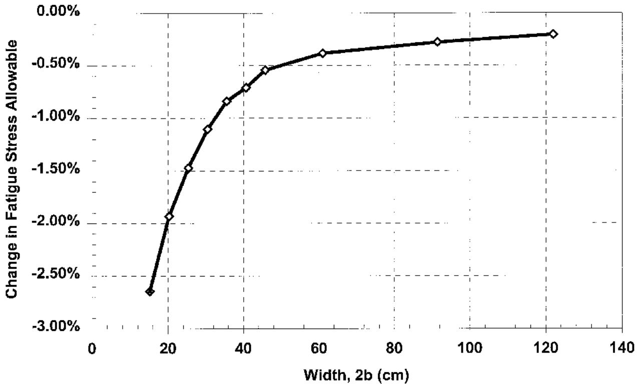

Fig. 7 Change in fatigue stress allowable vs panel width.

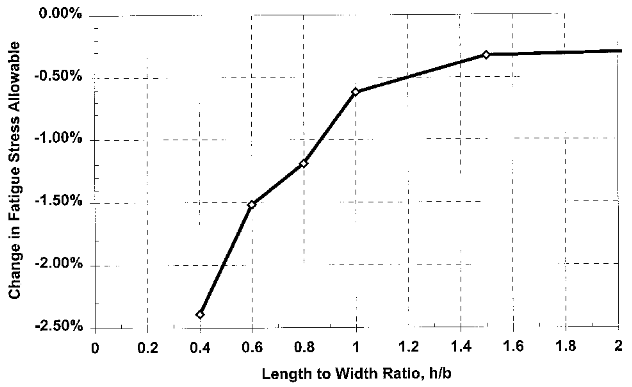

Fig. 8 Change in fatigue stress allowable vs panel length.

of 58.42 cm (23.0 in.) by 30.48 cm (12.0 in.) changed the fatigue stress allowable by about 2%.

Stiffened Panels

Aircraft wing panels are rarely simple sheets of metal as assumed in the previous analyses. Typically, the wing panel is constructed of numerous stringers riveted to a thin sheet (or skin). However, the global finite element model intentionally does not often include these structural details. Instead, the cross-sectional area of the stringers are smeared across the finite elements and the rivets are usually ignored completely. In other words, the thicknesses of the finite elements in the wing panels are adjusted to produce a global behavior equal to the skin-stringer combination. This global structural modeling technique is ideal for calculating stress intensities in stiffened panels.

In Poe's method,[14] the overall panel stress (stress applied to the skin-stringer combination) is used to establish the independent skin and rivet loadings in the stiffened panel. Therefore, stresses from smeared finite elements in a global aircraft model are exactly what are required for calculating stress intensities in a stiffened panel.

The stress intensity solutions for stiffened panels depend on three parameters: 1) stiffener spacing, 2) rivet spacing, and 3) stiffness ratio (ratio of the stiffener and skin stiffnesses). The stiffness ratio is calculated by

$$s = E_2 A / E_1 bt \tag{4}$$

where s is the stiffness ratio, E_1 is Young's modulus of the skin, E_2 is Young's modulus of the stiffener, t is the skin thickness, b is the distance between stiffeners, and A is the cross-sectional area of the stiffeners (Fig. 9). If the skin and stiffeners are made of the same material, i.e., same Young's modulus, then the stiffness ratio becomes the ratio of stiffener and skin areas.

The stress intensity as a function of crack length is graphically presented in Ref. 4 for varying values of stiffener spacing, rivet-to-stiffener-spacing ratio, and stiffness ratio. These data were used as $\beta_{\text{correction}}$ tables in the MODGRO program to calculate the crack propagation in a stiffened plate.

A baseline condition was established and each of the parameters were varied independently about the baseline. The baseline consisted of a 15.24-cm (6.0-in.) stiffener spacing, a rivet-to-stiffener-spacing ratio of 1/3, and a stiffness ratio of 0.428. In this manner, the effect of each parameter on the stiffened panel could be individually investigated.

In Fig. 10, the rivet-to-stiffener-spacing ratio was varied from 1 to 1/12, whereas the stiffener spacing and stiffness ratios were held constant at the baseline values. The fatigue stress allowable was normalized by the nonstiffened, infinite plate result. As the number of rivets between stiffeners increases, more of the stress in the crack vicinity can be transferred to the stiffeners through the rivets. Therefore, the increase in fatigue stress allowable as the rivet-to-stiffener-spacing ratio decreases was expected.

Figure 10 illustrates that the sensitivity of the fatigue stress allowable to the rivet spacing is significant. For example, a design change from a 5.08-cm (2.0-in.) rivet spacing to a 2.54-cm (1.0-in.) rivet spacing would increase the fatigue stress allowable by 7.4%. This could produce a significant weight savings in aircraft design.

The stiffened panel stress intensities used in the preceding text are based on Poe's analysis method,[14] which assumes that the rivets are rigid. Swift[15] demonstrated that treating the rivet as rigid results in overestimation of the stress intensity factor for small cracks and an underestimation as the crack approaches the stringers. Because the majority of a crack's life

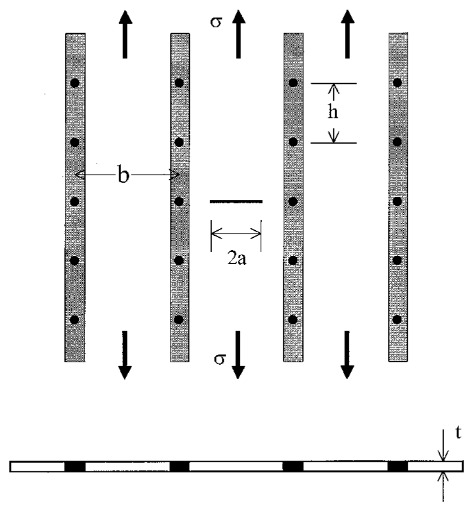

Fig. 9 Stiffened panel geometry.

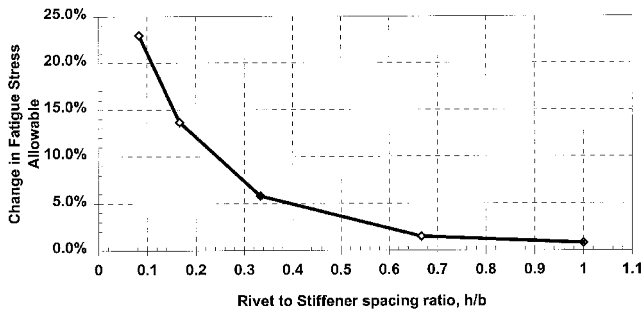

Fig. 10 Effect of rivet spacing on fatigue stress allowable. $s = 0.428$, $b = 15.24$ cm.

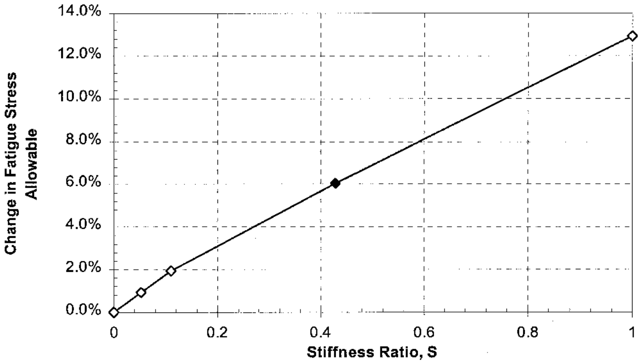

Fig. 11 Effect of stiffness ratio on fatigue stress allowable. *h*/*b* = 1/3, *b* = 15.24 cm.

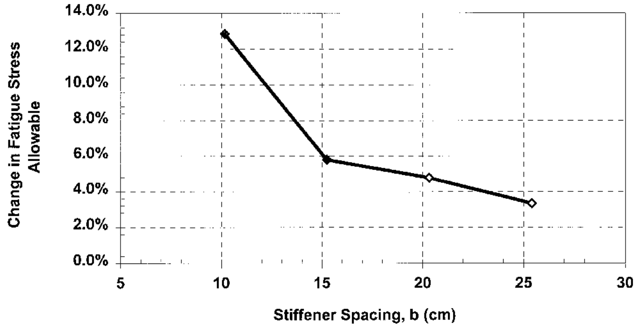

Fig. 12 Effect of stiffener spacing on fatigue stress allowable. *s* = 0.428, *h*/*b* = 1/3.

is spent at small crack sizes, Poe's method[14] should produce a conservative result.

Variation of stiffness ratio and stiffener spacing (Figs. 11 and 12) also produced significant changes in the fatigue stress allowable, although not as large as observed for the rivet spacing.

These results would change for a less severe stress environment, such as the design usage of a transport aircraft. Transport aircraft are typically designed for a higher number of flight hours, but with much less severe stress levels. Therefore, the crack would propagate at a slower rate over a longer distance than for a fighter aircraft. Because the stress intensity becomes progressively lower as the crack extends toward the stiffeners, a long and slowly progressing crack would be more affected by the stiffeners.

Variation of Stress over the Wing

If an aircraft wing were a true cantilevered beam with a uniform bending load, then the location of the panel on the wing would have little effect on the normalized stress history. However, the stresses in an aircraft wing can be much more complex because of the wing's structural design, aerodynamic loading, control surface loading, weapon carriage loads, and other factors.

A damage-tolerance analysis was conducted on every wing panel in the demonstration case except for the hardpoint areas. Each panel was modeled as an infinite plate with a 0.635-cm (0.25-in.) through-crack. Therefore, the only difference between each analysis was the difference in panel stress histories. The finite element nearest the panel's center was selected as the master element. Figure 13 illustrates the variation in fatigue stress allowable over the wing. The percentage change in fatigue stress allowable from the average allowable is indicated for each wing panel. The fatigue allowable varied approximately ±15% from the average across the wing. This amount of change in fatigue stress allowable cannot be ignored.

The stress exceedance distributions for the panels with the maximum and minimum fatigue stress allowables are plotted

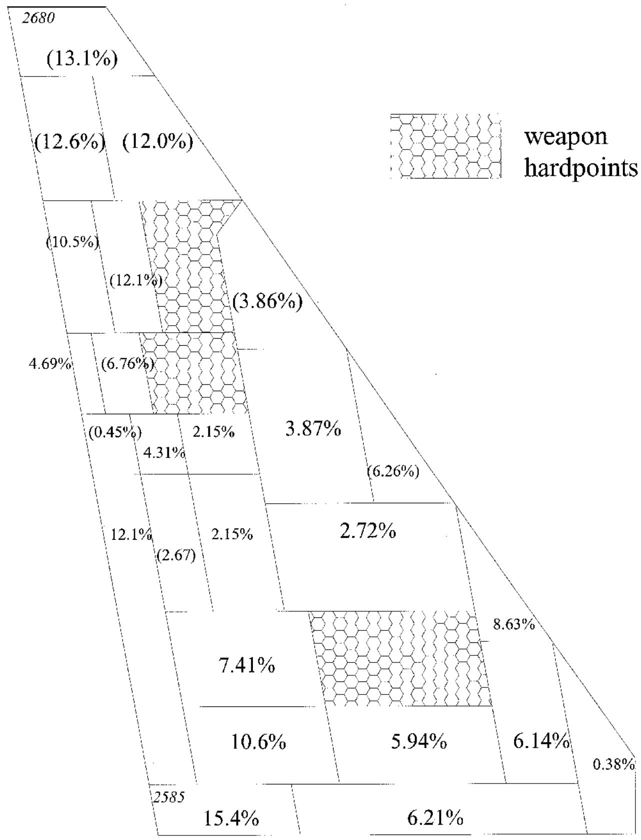

Fig. 13 Variation of fatigue stress allowable (from the average) over the wing.

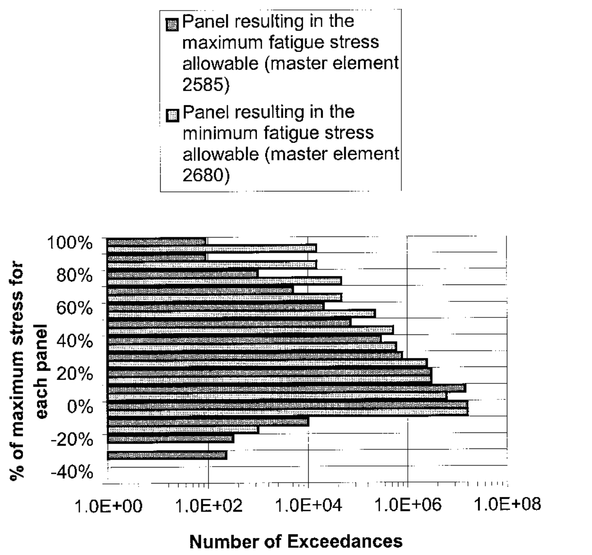

Fig. 14 Difference in exceedance distribution for panels with maximum and minimum fatigue stress allowables.

in Fig. 14. The panel with the lowest fatigue stress allowable is subjected to a more severe stress history, containing many more stress occurrences between 90 and 100% of the panel's maximum stress. This is because of high stresses at the wingtip in the 5.86 *g* unsymmetric maneuver.

Optimization

The most important feature of ASTROS is its ability to optimize a structural design. In the case of wing panels, the panel thicknesses are adjusted to optimize the wing design. The objective of the optimization is to produce the lightest possible wing without exceeding any of the design allowables. One of the design allowables used in wing design is the fatigue stress allowable.

The fatigue stress allowable is used as a wing design criterion during the ASTROS optimization to change the wing panel thicknesses. However, what effect does changing the wing panel thicknesses have on the fatigue stress allowable? The panel thickness can affect the fatigue stress allowable in two direct ways. First, the panel thickness is a critical parameter in the calculation of stress intensity for surface cracks. Second, adjusting the panel thicknesses changes the load path through the wing and, therefore, affects the panel stress histories. The objective of the last two trade studies was to determine the sensitivity of the fatigue stress allowable to the changes in wing panel thickness during the optimization process.

For all of the prior trade studies, panel thickness was not considered because a through-the-thickness crack was assumed. However, panel thickness does become an important variable in the stress intensity calculation for surface cracks. The slow-crack-growth criterion set forth by the U.S. Air Force specifies two different types of surface cracks: a semielliptical surface crack, and a corner crack at a hole.[3] Figure 15 illustrates the crack dimensions for the two surface cracks. The criterion does not specify a hole diameter for the corner crack, and so a reasonable rivet hole size of 0.508 cm (0.2 in.) was assumed.

The ratio of panel thickness to minor axis (through-thickness direction) crack length was varied for the two crack types. The thickness-to-crack-length ratio equal to one is identical to a through-crack solution. In Fig. 16, the change in fatigue stress allowable from the through-crack solution is plotted for different thickness ratios. Clearly, the fatigue stress allowable demonstrates a significant amount of sensitivity to the panel thickness when the crack is a semielliptical surface crack.

When a wing is optimized in ASTROS, all of the wing panels can be initialized at a constant thickness. During the optimization, the wing panel thicknesses are altered to produce resulting stresses close to, but not exceeding, the design allowables. Clearly, the optimization process affects the load paths throughout the wing; hence, the panel stress histories are changed as well.

The wing panel thicknesses in the demonstration model are based on the production F-16 design. Therefore, the thicknesses in the model represent a wing that has already been optimized to some extent. For comparison purposes, the model was modified so that the (smeared) thicknesses for all of the wing panel elements were 0.625 cm (0.25 in.) to represent a nominal wing before optimization. The fatigue stress allowables for a through-crack were calculated for the constant-thickness wing model and were compared to the original demonstration model.

Figure 17 illustrates the change in the fatigue stress allowables from the constant thickness model to the optimized model. The majority of the fatigue stress allowables changed only ±2%; however, the fatigue stress allowable for one panel decreased by 14% for the optimized wing. Therefore, the fatigue stress allowable calculated prior to optimization is nonconservative. Ignoring the change in fatigue stress allowables resulting from optimization could lead to a reduced service life of the production aircraft. However, it is important to note that this change in fatigue stress allowable results only from the change in stress distribution in the wing caused by the change

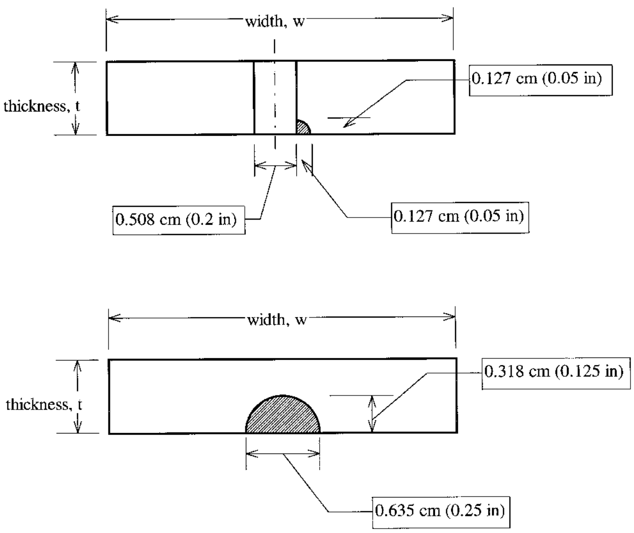

Fig. 15 Surface cracks.

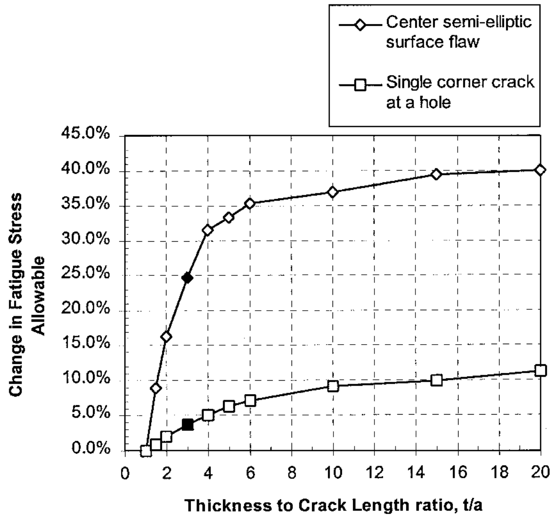

Fig. 16 Effect of panel thickness on fatigue stress allowable.

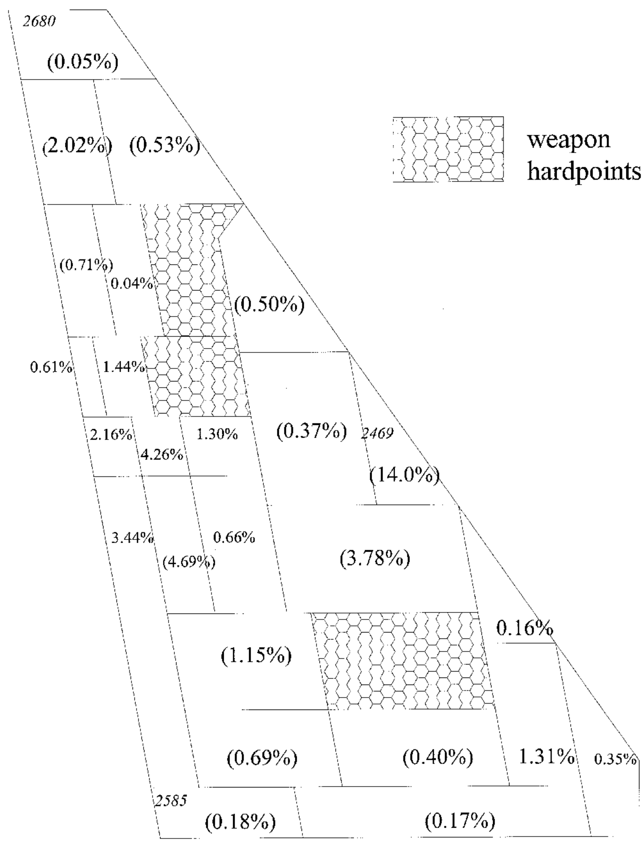

Fig. 17 Change in fatigue stress allowable as a result of optimization.

in panel thickness, and not from the panel thickness itself. The direct effect of panel thickness on the fatigue stress allowable (Fig. 16) is eliminated because a through-crack is assumed for all panels. The changes in fatigue stress allowables as a result of optimization are large enough that they should not be ignored. Therefore, it is recommended that fatigue stress allowables be updated at least once during the optimization process.

Conclusions

A methodology for implementing fatigue design issues into the global design of aircraft was developed using current damage-tolerance design guidelines and analysis techniques. The methodology represents a fusion of the ASTROS finite element capabilities with MODGRO and other independent procedures and analytical tools for fracture mechanics, fatigue analysis, and aircraft design usage. This synthesis promises not only to reduce unexpected fatigue problems, but also to increase aircraft performance through a weight savings in less critical panels.

The methodology was implemented in a software code to facilitate an investigation of the effect of different design variables on the aircraft design. The trade studies indicated that the sensitivity of the fatigue stress allowable to the panel design warrants analysis of all wing panels, instead of only the critical panels at the wing root.

The panel design features that had the most significant effect on the fatigue stress allowables were material fatigue properties, stiffened panel design, panel thickness, and panel location on the wing. As expected, the fatigue stress allowable was very sensitive to the material's fatigue characteristics ($\mathrm{d}a/\mathrm{d}N$ vs K). In contrast, the fracture toughness of the material had little effect. Panel thickness had a moderate effect by redistributing stresses in the stress history. In addition, for surface cracks the direct impact as a function of panel thickness was pronounced. Finite length and width corrections had such limited influence that modeling a nonrectangular panel as an equivalent rectangular panel with an average length and width should introduce little error. Stiffened panels also significantly affected the fatigue stress allowable. These results reinforce the importance of multidisciplinary design optimization (MDO) because stiffened panel design is largely influenced by other disciplines, such as local panel buckling.

Based on these results, other capabilities are recommended for implementation in the methodology, including the effect of the wing spars on the fatigue analysis and the capability to automatically handle complex multiphase crack scenarios.

Mixed-mode fatigue is another area that deserves future attention. Following published recommendations, the mixed-mode fatigue situation was handled by using the maximum principal stress as an effective stress in pure mode 1 fatigue. However, this procedure has not been adequately validated for components subjected to the complex stresses typical in real-life aircraft design.

Damage tolerance has been a part of aircraft design for only a little more than two decades. Although many advances have been made in the field in this time, damage-tolerant design is still in its infancy. With the recent emphasis on MDO, damage tolerance is only one of many engineering disciplines that are being re-examined for integration into aircraft optimization. The ultimate goal is to develop an optimization procedure capable of increasing the safety and performance of modern aircraft while reducing the time and cost of development. It is hoped that the methodology presented in this paper can act as a starting point for future developments and refinements in this area of research.

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