1/4

NASA TECHNICAL MEMORANDUM

NASA TM X-7192

COPY NO.

NASA TM X-71925

(NASA-TM-X-71925) FRACTURE MECHANICS (NASA) 12 p HC $4.00 CSCL 20K

N74-21545

Unclas
G3/32 35330

FRACTURE MECHANICS

by Herbert F. Hardrath

Langley Research Center
Hampton, Virginia

Dryden Lecture presented at AIAA 12th Aerospace Sciences Meeting, Washington, DC, January 31, 1974

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LANGLEY RESEARCH CENTER, HAMPTON, VIRGINIA 23665

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA TMX -71925		

4. Title and Subtitle

Fracture Mechanics (U)

5. Report Date

January 31, 1974

6. Performing Organization Code

7. Author(s)

Herbert F. Hardrath

8. Performing Organization Report No.

9. Performing Organization Name and Address

Materials Division
Langley Research Center
Hampton, VA 23665

10. Work Unit No.

11. Contract or Grant No.

12. Sponsoring Agency Name and Address

National Aeronautics and Space Administration

13. Type of Report and Period Covered

Dryden Lecture for AIAA

14. Sponsoring Agency Code

15. Supplementary Notes Dryden Lecture presented at AIAA 12th Aerospace Sciences Meeting, January 31, 1974, Washington, DC. Will be published in AIAA Journal of Aircraft in summer, 1974.

16. Abstract

Fracture mechanics is a rapidly emerging discipline for assessing the residual strength of structures containing flaws due to fatigue, corrosion or accidental damage and for anticipating the rate at which such flaws will propagate if not repaired. The discipline is also applicable in the design of structures with improved resistance to such flaws. This paper reviews the present state of the design art using this technology to choose materials, to configure safe and efficient structures, to specify inspection procedures, to predict lives of flawed structures and to develop reliability of current and future airframe.

17. Key Words (Suggested by Author(s)) (STAR category underlined)	18. Distribution Statement
<u>32 Structural Mechanics</u> Fracture Fatigue Crack Propagation Damage Tolerance Stress Corrosion Cracking	Unclassified - unlimited

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	12 ~~10~~	NTIS

*Available from The National Technical Information Service, Springfield, Virginia 22151
STIF/NASA Scientific and Technical Information Facility, P.O. Box 33, College Park, MD 20740

FRACTURE MECHANICS

Herbert F. Hardrath
NASA Langley Research Center
Hampton, Virginia

I. Introduction

Fracture mechanics is a rapidly developing engineering discipline dedicated to the analysis, prediction, and prevention of structural failures due to flaws or cracks. Usually the cause or source of the flaw is not specifically addressed, but its growth in the expected service environment and its effect on residual strength are dealt with quantitatively. Most of the organized effort on fracture mechanics is coordinated through Committee E-24 on Fracture in the American Society for Testing and Materials. However, many other groups have sponsored symposia on the subject and several journals are devoted to disseminating current results.

This paper provides only a brief historical sketch of this discipline and a cursory summary of the current analytical procedures. These matters are treated in more detail in many other publications.

An attempt is made to assess the current status of the discipline, to suggest some engineering applications, and to recommend directions for future study. The suggested applications are illustrated by sample calculations based on materials data consistent with those compiled in the Damage Tolerant Design Handbook.[1]

II. Brief Historical Review

Fractures emanating from minute flaws have caused many spectacular disasters during the past century.[2] Consequently, much research has been conducted to quantify the circumstances that lead to failures and to design against such failures in the future.

Griffith[3] is generally credited with conducting the earliest quantitative research on the phenomenon in the early 1920's. His experiments (Fig. 1) were carried out on glass. With simple glass rods heated to a semiplastic state while under load, he determined the surface tension of glass. In other experiments he determined the pressure-induced stress at which simple glass tubes and spheres containing cracks would rupture. He then established the criterion that failure occurred when the elastic strain energy released by the growth of the crack equals the energy required to create new surface at the crack tip. His analysis resulted in the equation

$$S\sqrt{c} = \sqrt{\frac{2ET}{\pi}}$$

for plane stress in which

- S is the stress perpendicular to the plane of the crack
- c is the half-length of the crack
- E is Young's modulus
- T is surface tension

The correlation shown in Fig. 1 with $S\sqrt{c}$ equal to a constant remains as the fundamental basis for modern fracture analyses.

The sudden failure of many welded ships during and after World War II led to a nationwide cooperative effort to correct whatever deficiencies were responsible. Failures were consistently traced to small flaws, at corners of major cutouts; they occurred on very cold days; and they appeared to be "brittle" in nature. Dr. George Irwin was a member of the team that investigated these failures and his adaptation of the original Griffith relation led to the now familiar Griffith-Irwin analysis methodology However, the solution to the ship failure problem was chiefly one of choosing materials that exhibited adequate impact strength at the lowest service temperatures experienced by naval ships. Elimination of sharp corners, control of residual stress, and improved welding procedures also helped to eliminate manufacturing flaws and stress conditions conducive to creating cracks in service. Curiously, these solutions had little relation to the fracture mechanics discipline as it is known today.

Aircraft designers were brought into the circle of those concerned as an aftermath of the Comet accidents in 1954. In this case, the failures occurred in aluminum alloy and resulted from small fatigue cracks initiated at the corners of window cutouts. For civil aircraft, the solution was to deliberately introduce redundancy and thus to render the structure "fail-safe." The Federal Aviation Administration adopted[4] a fail-safe standard (FAR 25.571c) which requires "that catastrophic failure or excessive structural deformation, that could adversely affect the flight characteristics of the airplane, are not probable after fatigue failure or obvious partial failure of a single principle structural element." This standard has become the primary method of certifying airworthiness of civil aircraft for many years. Again, designing to satisfy this standard has rarely required a formal application of fracture mechanics methods. In first approximation, service loadings are assumed to be shared by the remaining members of a redundant set after one member of the set fails. In practice, the flaw size that must be tolerated without jeopardizing safety is usually 500 mm (20 in.) or so in length.

Much of the impetus for developing fracture mechanics technology came from troublesome problems with welded pressure tanks for missiles and space vehicles. Here, redundancy cannot be usefully employed and welding has a tendency to introduce significant flaws during manufacture. Extreme premiums on weight force the use of very high strength materials and high stresses - a combination that places crucial dependence upon fracture properties. Further, cryogenic temperatures and highly aggressive environments tend to aggravate the problem. A proof-test procedure based upon fracture mechanics principles[5] has been developed and is used widely for pressure vessels. Generally, the structures involved are relatively simple and only a few stress cycles are encountered during the service lifetime. Tolerable flaw sizes are usually not over a few millimeters (<0.1 in.) in depth.

Military requirements have not dealt with fail-safety or damage tolerance until very recently. However, the failure of an F-111 aircraft[6] as a result of a flaw that escaped detection during manufacture led to a reevaluation of policy. Consequently, the U.S. Air Force[7] has required the manufacturer of the B-1 bomber to design all structural parts to new, rather stringent damage tolerance criteria and is considering the adoption of similar requirements for all future systems. These criteria require that all structures be assumed to have flaws of specified size located at all critical stations and that such flaws shall not grow to critical size during a design lifetime of the structure in parts not susceptible to routine inspections. For inspectable structure, the specified size is adjustable to account for the sensitivity and frequency of scheduled inspections. These new requirements, if adopted, will probably stimulate rapid growth in the development of fracture mechanics techniques and their application, because they must be applied systematically to very complex structures and account for extremely complex load histories encountered during many years of service. The residual static strength of the structure must be maintained above a level statistically dependent upon the load sprectrum and inspection interval.

The foregoing historical review of the development of fracture mechanics may be summarized in the following table.

Decade	Problem	Flaw size (approximate)	Cause	Solutions
1920	Flaws in glass	1 mm	Inherent	Physical properties
1940	Welded steel ships	Several meters (including hatch) size)	Welds	Metallurgical properties Eliminate source of cracks
1950	Civil aircraft	1/2 meter	Fatigue	Redundancy
1960	Spacecraft	1-3 mm	Welds and stress corrosion	Proof-test
1970	Military aircraft	5-10 mm (including fastener)	Manufacturing, fatigue, corrosion	Life prediction Inspection

In only the last two instances are formal fracture mechanics employed to assist the designer in a solution to the problem confronting him.

III. Status of Modern Fracture Mechanics

The vast majority of fracture mechanics analyses are based upon linear elastic considerations. The stress distribution is recognized to contain a singularity at the crack tip and usually is of the form shown in Fig. 2. The strength of the singularity in stress is measured by the numerator of first term of the equation. This numerator is usually designated K, has the units $MN\text{-}m^{-3/2}$ (or $Ksi\text{-}in^{1/2}$), and is called the stress intensity. For most phenomena that can be correlated by linear elastic analyses, the nominal stress term (second term in the equation) is small compared to the term measuring the singularity. Thus, it is neglected along with succeeding terms of higher order. The assumption is made that behavior will be the same in a given material, every time the stress intensity reaches the same value. For example, fracture will occur whenever K reaches the same critical value K_c, or a certain increment of crack growth will occur every time an excursion ΔK is experienced.

Fracture

Attempts to correlate fracture through energy balances in the physical sense employed by Griffith in his early experiments have been futile. Orowan[8] estimated that the plastic deformations preceding fracture may require amounts of energy that are 1000 times greater than those required to produce new surfaces. In practice, fracture toughness values reported for particular materials are essentially the empirical values of stress intensity for a given load and configuration that causes fracture in prescribed experiments.

The stress intensity is critically dependent upon the configuration of the specimen or part under consideration, the size of the part relative to crack dimensions, and the type of loading. Recent compilations[9,10] of stress intensity factors for various practical and mathematically tractable configurations contain several hundred solutions. Finite element analyses are also being employed for complex two-dimensional configurations and some current studies are developing analyses for three-dimensional cases. These solutions are needed to predict fracture behavior in practical configurations from the materials data developed in standardized test specimens.

Standard specimens (Fig. 3), test technique, and analysis have been adopted[11] to assure that tests conducted in various laboratories are likely to have produced a "brittle failure." These tests are then considered "valid" for establishing plane strain fracture toughness, K_{Ic}. This value is intuitively expected to be a constant for a given material. The proportions of the specimen are restricted to force plane strain behavior at the crack tip and thus to limit local plastic deformation. For many test results, particularly those from tests of ductile materials, fracture toughness values calculated by elastic analyses vary significantly from this constant value. For example, the data in Fig. 4 were obtained from 2219-T851 aluminum alloy compact specimens[12] and demonstrate consistent trends toward higher critical elastic stress intensity values K_{Ie} for wider specimens and for thinner specimens.

The three-dimensional diagram in Fig. 5 displays the same data shown previously, but is extended to show the expected data for specimens outside the "valid" range. The horizontal plane has thickness and applied nominal stress coordinates and the vertical coordinate is the fracture toughness calculated by elastic stress intensity analyses using the maximum stress and initial crack length. The stress at fracture tends to be greater for narrow specimens, thus, the nominal stress coordinate can also be related to the reciprocal of specimen width. The crack length is, of course, included in the calculation of stress intensity plotted on the vertical axis. The surface in the figure is the locus of fracture toughness for all combinations of specimen dimensions. The ASTM standard test procedure is expected to produce a single value of plane strain fracture toughness. Such behavior would produce a plateau near the left extremity of the surface shown. However, for many materials of practical interest the specimens required to produce "valid" results are so large that tests are virtually impossible and these conditions were not achieved for the data sample shown. Most practical applications involve thicknesses that are much smaller than that producing plane strain fracture. Thus, extensions of the linear-elastic analyses and relaxation of these test restrictions are required to treat practical situations.

Many investigators have recognized this deficiency. Accordingly, several other analyses have been developed to "correct" for the plasticity that is generally accepted as being responsible for nonbrittle behavior. The intent is usually to modify the analysis so that the fracture toughness calculated from test results has a common value for a given material. Among these procedures are the Irwin plasticity correction,[13] the Dugdale plastic zone,[14] the R-curve method,[15] and the J-integral.[16] Each of these procedures has had only limited success in treating somewhat broader ranges of specimen dimensions, but fail to describe behavior over commonly used and practical ranges.

Newman[17] has proposed a two-parameter fracture criterion which, for tensile fracture test specimens, employs the equation

$$K_{Ie} = K_F\left(1 - m\,\frac{S_N}{\sigma_u}\right)$$

where

K_{Ie} is the critical elastic stress intensity

K_F and m are new material parameters

S_N is the nominal stress on the net section at failure

σ_u is the ultimate tensile strength of the material

The parameter m is a nondimensional constant that measures the influence of net section stresses on fracture behavior. The parameter K_F has the same form as fracture toughness, and represents the limiting value of K_{Ie} for very wide specimens (fractures at low stresses). The parameter K_F is a function of specimen thickness. Although, at its present stage of development, the values K_F and m can only be determined empirically to fit data for a particular material, good correlations have been achieved for large sets of data that had exhibited rather large variations in K_{Ie}. This equation was used to plot the surface shown in Fig. 5. Apparently the Newman relation introduces an appropriate and useful new concept of net section stress sensitivity that will broaden the applicability of fracture mechanics concepts. Unfortunately, the consistent strong efforts to force plane strain conditions in fracture tests has severely limited the range of data available for a given material. Inasmuch as many practical fracture problems lie outside the rather limited "valid" range of data, test data and analytical capability should be expanded beyond this limited range.

Fatigue Crack Propagation. Paris[18] was among the first to suggest stress intensity range as a suitable parameter for correlating rates of fatigue crack propagation. Today, this is the most commonly used parameter for that purpose. A typical curve is shown in Fig. 6. The rate of growth per cycle is usually plotted as a logarithmic scale because it usually varies over several orders of magnitude for a complete description of a given material.

Data such as those shown can be affected significantly by frequency of loading, the chemical environment, and temperature. The data are characterized by a threshold, below which crack growth is presumed not to occur or progresses at a negligibly slow rate. At intermediate values of ΔK the data lie along a straight line, frequently approximated by a power law.

At high ΔK levels where the stress intensity approaches the critical value, rapid crack growth occurs. A different curve is also obtained for various mean stresses or stress ratios. Several analytical expressions have been proposed to correlate these data,[19-23] each having its own limitations. Again the fit is empirical.

Elber's[24] "crack closure" concept is based upon physical observations of behavior at the crack tip. Through measurements of motion of the material surrounding the crack tip, he showed that the crack surfaces came into contact while the specimen supported significant tensile load (Fig. 7). The crack closure is apparently due to significant plastic deformations remaining in the material (shown shaded in the figure) that has been severed by crack growth. Thus, the effective range of ΔK is reduced to the portion corresponding to that part of a loading cycle during which the crack surfaces are not in contact. Further development is required to quantify the crack closure method for prediction purposes. However, it has been used to correlate results of tests at various ratios of minimum to maximum stress.

Fatigue Crack Propagation Under Variable Amplitude Loading. Practically all loading histories encountered in service are complex sequences of cycles with varying amplitude and mean values. Fatigue crack propagation tests conducted under such load histories demonstrate significant delays in crack growth following high load cycles.

This topic has recently received much attention and several empirical methods to account for the behavior have been proposed. Wheeler[25] and Willenborg, Engle and Wood[26] consider the size of the plastic zone developed at the crack tip for each stress cycle. This size is used to adjust the rate of crack propagation during successive stress cycles according to the relative sizes of plastic zones produced by each cycle. Elber's crack closure concept,[24] described earlier, would eliminate all stress excursions below the crack closure level (Fig. 8). In each of the three models, the crack growth rate is calculated for each individual cycle in the time history.

As of this writing, the current methods are empirically adjusted to fit test data. The constants developed have not been demonstrated to be transferrable from one material, configuration, and loading sequence to another.

Application of these principles to life prediction in service requires extremely tedious calculations on a cycle-by-cycle basis. The complete time history of stress expected in service is also needed. Such information cannot be anticipated before the aircraft is flown and would not be consistent for any two aircraft in a given fleet! Because the interactions always work in the direction of extending life (in spite of some short periods of accelerated growth), one practical simplification would be to ignore the effect. However, the interactions frequently cause life extensions of several times the "no-interaction" life, an advantage that seems worthy of consideration in many life-critical applications.

To deal with this state of affairs in a practical manner, research should be conducted in which the interactions are calculated for some statistical measure of the service load history. Very little such work has been conducted to date.

Stress Corrosion Cracking. For many decades stress corrosion cracking has been observed to occur in some materials, notably brass, that contained significant tensile residual stresses left during manufacture. The phenomenon has become a matter of primary concern in structural materials, especially when the tensile stress is applied parallel the short transverse grain direction. Welded pressure vessels subject to high stresses and long exposure to aggressive environments are particularly vulnerable. However, many aircraft parts have failed due to built-in stresses and moderate environments. Consequently, considerable effort has been devoted to quantifying this behavior. Typical results[27] for tests of a given material under a variety of stress levels are shown in Fig. 9. The time to cause failure is plotted against the value K_{Ii}, the stress intensity computed for the intial crack length. The stress intensity below which failure is not expected at very long times is termed K_{th}, or the threshold value. The behavior is, of course, critically dependent upon the material, its thickness, grain direction, the corrodent, and the temperature. Flaws are generally expected in weld zones, if present, where nontypical metallurgical influences may be present.

The detailed physical-chemical reactions responsible for the behavior are under intensive study and are expected eventually to lead to the development of materials with enhanced resistance to stress corrosion cracking. In an engineering sense the mechanism is one of progressive crack growth until the critical size is reached and fracture occurs.

In principle, stress corrosion tests can yield information regarding the time rate of crack propagation for particular combinations of parameters.

IV. Design Applications of Fracture Mechanics

The design of a modern airframe structure involves many compromises among strongly conflicting requirements. Certainly, one of the strongest driving forces is the very high premium placed on weight in order to achieve the desired performance. Usually, the minimum weight is achieved through the use of high strength materials, high design stresses, extensively sculptured parts, and as few joints as possible. Unfortunately, these practices lead to a structure that is highly sensitive to the fracture mechanisms discussed in this paper. If one acknowledges that flaws and cracks of significant size are inevitable in structures, some safeguard must be incorporated in design to assure adequate safety in spite of such damage. Manufacturing quality control and the frequency and sensitivity of in-service inspections must be among the factors considered. The analytical model for making the optimum choice among all these factors has yet to be developed. However, some guidance can be gained from simplified considerations based on the fracture mechanics principles outlined herein.

Material Selection

Because most research on fracture has been conducted to characterize materials behavior, the selection of materials for a particular design situation is one of the most direct applications of fracture mechanics principles. Three bases for selection are discussed in the following sections.

Fracture. The simplest and most direct selection scheme is on the basis of fracture strength. Figure 10 is one type of plot that might be used to compare several materials. The critical load intensity per unit width for four candidate materials is shown plotted against crack length. The structural configuration is assumed to be a simple wide plate with a central crack and subjected to axial tension loads. For this example, the plates were designed to have weights equal to that of a titanium plate 25 mm thick (1 inch). The curves were calculated by the Newman equation discussed earlier, but adjusted to data for the candidate materials as listed in the Damage Tolerant Handbook.[1] From this simple comparison, the modest strength 2024-T351 aluminum alloy retains a higher residual strength than the other three materials for all cracks more than 10 mm (0.4 inch) long. Further, at a given strength level, the tolerable crack is much longer in this material than in the others. The contrast is even more striking if the panels were designed to have equal unflawed strengths (not shown), but, of course, the aluminum alloy panels would have been considerably heavier than the others. For the monolithic structure considered here, residual strength is reduced below 50% of the original tensile strength for the steel when the crack is only 3 mm (0.1 inch) long. This fact alone should warn against the use of such highly sensitive materials unless inspections are frequent and sensitive enough to detect very small damage.

Crack Propagation Resistance. The choice of material might also be made on the basis of crack propagation resistance. Elber and Davidson[28] have developed the three-dimensional diagram in Fig. 11 to illustrate the logic that should be used. The coordinates of the figure are stress-to-density ratio, number of load cycles per inspection interval, and initial crack length (inspection sensitivity). The surface in the figure is the locus of stresses that will lead to failure within the inspection interval for each initial crack length. For this example, the equation of the surface was derived from the relation

$$\frac{da}{dN} = \left(\frac{\Delta K}{C}\right)^n$$

where

$\frac{da}{dN}$ is the rate of fatigue crack propagation

ΔK is the range in the stress intensity for the applied loading

C and n are material constants characterizing crack growth resistance

The surface intersects the left-hand vertical plane along the residual static strength curves just discussed and has been blended to intersect the right-hand vertical plane along the familiar S-N curve characterizing fatigue failure of an unflawed part.

The surfaces for the 2024-T351 aluminum alloy and the Ti-6Al-4V titanium alloy from the previous discussion are shown only in those regions where the displayed material can carry the higher repeated stress. For the other two materials of the previous example, the surfaces lie below those shown throughout the range. In this comparison, the titanium alloy structure could be made lighter than the other three if modest sized cracks are under consideration.

Stress Corrosion Cracking. The best way to obviate stress corrosion failures is to prevent exposure to the aggressive environment through special coatings, paints, or bladders. However, results such as those shown can have several other applications. First, and most importantly, the relative compatibility of a given material and an environment can be assessed and appropriately resistant materials selected. One needs only to compare the K_{th} values for this purpose. Second, the design stress level may be adjusted to preclude failure during the expected life of the structure. Shotpeening has been successfully used to introduce residual compressive stresses and thus to preclude failure.[29] Third, and least reliably, the life to failure may be predicted by integrating the rate of propagation over the time of exposure at appropriate stress levels and environments. Usually this last application is least likely to be useful for aircraft design because the total exposure is not known or defined to the precision desired. Further, the accumulation of the requisite data would be prohibitively expensive. Under the best of circumstances an integration over many years of service should be regarded with suspicion.

For pressure vessels on spacecraft, the stress and corrosive environments are more likely to be well defined. However, the required lives of spacecraft can only be met if their pressure vessels are designed not to exceed the threshold stress intensity for the worst flaw possible in the vessel. No consideration of rate of growth and life is needed.

Fracture Behavior in Built-Up Structure. Many structural components are built up of many parts that share the applied loadings. The forces in these parts are redistributed as a crack progresses through the structure. This redistribution can have a profound effect on the rate of crack propagation and on the residual strength. Thus, an analysis of stresses in redundant members such as fuselage rings and stringers in wing panels can be a powerful tool in designing crack resistant structures.

The curves of Fig. 12 illustrate the results of one such analysis.[30] The stress intensity, normalized by the applied stress is shown as a function of crack length for an unstiffened or monolithic panel and for a stiffened panel. The stiffened panel had 50% of its total cross-sectional area in stringers spaced 102 mm (4 inches) apart and the stringers were fastened to the sheet with rivets spaced 25 mm (1 inch) apart. The stress intensities are equal in the two panels for very short cracks, but are much lower for long cracks in the stiffened panel. The stress intensity dips sharply each time the crack grows past a stringer.

These large effects on stress intensity cause dramatic differences in life to propagate a crack from some initial size to failure of the sheet. The solid curves in Fig. 13 show the life to be an order of magnitude longer for the stiffened panel.* In addition, the curves indicate the stiffened panel would survive thousands of flights after the crack has traversed two stringer spacings on each side of its origin. Such behavior is very reassuring to an operator and is responsible for the fact that many so-called fail-safe structures have survived with rather long cracks. On the other hand, monolithic structures have critical crack lengths very much smaller.

The dashed curve in Fig. 13 illustrates behavior when the center stringer is broken.[31] Even for this case, the panel with the broken stringer retains a life of thousands of flights, and approaches the behavior of the panel with all stringers intact for cracks exceeding one or two bays. The essentially horizontal initial portion

---

*These curves were calcualted for a commercial transport spectrum and a 1-g stress of 69 MN/m^2 (10 ksi). The interaction effects discussed in an earlier section were neglected.

of this curve represents rapid crack growth in the sheet because the load in the broken stringer is carried by the sheet.

Because these curves have very steep slopes for short crack lengths, one may be tempted to assume very short cracks in design. However, such an assumption must be supported by a nondestructive inspection capability consistent with it. Candid assessments[32,33] indicate that current inspection procedures are not sufficiently sensitive to take advantage of the early part of these curves. Unfortunately, currently contemplated criteria encourage designers to consider small initial flaw sizes. The natural outcome may well be that monolithic structures will be employed to reduce drastically the number of potential crack sites and little or no tolerance to significant flaws will be provided. Certainly, such structures offer little protection against battle or accidental damage. Further, an impossibly sensitive in-service inspection procedure will be required to maintain continued confidence in fleet structures. An assumption of extremely short crack lengths also leads to large errors in life prediction for modest errors in local stress analysis.

Similar analyses may be employed to calculate the residual strength of wide panels containing cracks. In Fig. 14 the curves show the nominal stress that must be applied to the panel to cause failure of the sheet (calculated from the fracture toughness of the material and the stress intensity developed in Fig. 12) as a function of the number of flights flown. The maximum flight stress was considered to be 138 MN/m^2 (20 ksi), a level reached approximately once every 100 flights. The undulations in the curve reflect the reduction in stress intensity each time the crack passes a stiffener. The residual strength of the sheet is given by the maximum peak of the curve. For practical purposes, the minima in the curve maintain a constant level until the total cross-sectional area has been reduced significantly. Although the undulations occur at shorter intervals of life once a stringer fails (dashed curve), the minimum strength is maintained at essentially the same level as for the panel with intact stringers. A significant margin of strength is retained in the sheet for many flights. In marked contrast, the strength of the monolithic panel was reduced to its critical level at an early stage. Designers and procuring agencies will do well to consider the significant enhancements of safety possible by applying these principles without major weight penalties.

Proof-Testing

Because current nondestructive inspection techniques are incapable of finding minute flaws with high reliability, proof-tests have been employed to demonstrate flight worthiness of pressurized vessels for spacecraft.[5] A vessel passing such a test is presumed to contain a flaw just smaller than the size that should have caused fracture at the proof-test stress level. The vessel is then operated at a somewhat lower stress chosen to be safe on the basis of other tests of the same material, crack size, stress level, and operating environment.

A similar scheme was employed to qualify F-111 aircraft after this aircraft suffered a catastrophic fracture.[6] The scheme was adopted because no other economically feasible scheme was capable of detecting the small flaws that had to be found.

However, the proof-test of large complex structures can be justified only in extreme cases. Granted that large economic benefits can be achieved, proof-tests of complete airframes can rarely be conducted to challenge all parts of a structure to the same degree. Further, many parts (particularly in a wing or other structure loaded in bending) are loaded in compression at levels that may well leave residual tension stresses. Such stresses can aggravate subsequent fatigue and stress corrosion damage accumulation. The requisite prediction of life subsequent to the proof-test is also subject to many more uncertainties and potential errors than are likely for pressure vessels that must survive only a few cycles.

V. Concluding Remarks

Some applications of fracture mechanics in the design of efficient and safe aerospace structures have been reviewed. The underlying rationale appears capable of dealing with material selection for residual strength, resistance to fatigue and stress corrosion cracking, the optimum deployment of structural parts for best residual strength and crack life and the prediction of life under variable amplitude loadings. The discipline is developing rapidly toward providing quantitative predictions for each of these behaviors. However, in many cases the existing knowledge is applicable over rather limited ranges of practical variables.

Extensions are needed to account for fracture under other than plane-strain conditions and to provide a practical means for predicting life under complex time histories of loadings. Even without these extensions, designers should be able to profit from comparisons of basic behaviors of candidate materials and structural configurations. Very large improvements in structural behavior are possible by employing redundant structure and controlling the deleterious effects of agressive environments with modest weight increases. For cases where a structure must be protected against failure due to significant flaws, the limiting stress level can be established with reasonable accuracy.

Because the U.S. Air Force is considering a strong new damage tolerance criterion, research activity in this structural discipline is likely to be accelerated dramatically during the next decade.

VI. Acknowledgments

The author thanks the Honors Committee of AIAA for inviting him to prepare this lecture, the management of the NASA Langley Research Center for supporting him in the preparation of this paper, and the following personnel of the Fracture Mechanics Section, Materials Division, for invaluable assistance in writing and criticizing the manuscript - Drs. H. A. Leybold and W. Elber and Messrs. C. C. Poe, Jr., and J. C. Newman, Jr.

References

1. Damage Tolerant Design Handbook, A Computation of Fracture and Crack-Growth Data for High-Strength Alloys, Metals and Ceramics Information Center, MCIC-HB-01, December 1972.

2. Parker, Earl R., "Brittle Behavior of Engineering Structures," John Wiley and Sons, 1957.

3. Griffith, A. A., "The Theory of Rupture," Proceedings of the First International Congress of Applied Mechanics, 1924.

4. Anon, "Airworthiness Standards: Transport Category Airplanes," Federal Aviation Regulations, Vol. I, Part 25, FAA, February 1965.

5. "Fracture Control of Metallic Pressure Vessels, NASA Space Vehicle Design Criteria," SP-8040, May 1970, NASA.

6. Hinders, Urban A., "F-111 Design Experience - Use of High Strength Steel," Presented at Second Aircraft Design and Operations Meeting, AIAA, July 1970.

7. Wood, H. A., "The Use of Fracture Mechanics Principles in the Design and Analysis of Damage Tolerant Aircraft Structures, Fatigue Life Prediction for Aircraft Structures and Materials," AGARD-LS-62, May 1973.

8. Orowan, E., "Fundamentals of Brittle Behavior in Metals," Fatigue and Fracture of Metals. The Technology Press of MIT and John Wiley and Sons, 1952.

9. Tada, H., Paris, P., and Irwin, G., "The Stress Analysis of Cracks Handbook," Del Research Corporation, Hellertown, Pennsylvania, 1973.

10. Sih, G., "Handbook of Stress-Intensity Factors," Lehigh University, 1973.

11. Standard Method of Test for Plane Strain Fracture Toughness of Metallic Materials, ASTM Designation: E399-72, Annual Book of ASTM Standards, Part 31, 1972.

12. Kaufman, J. G., and Nelson, F. G., "More on Specimen Size Effect in 2219-T851 Aluminum Alloy," Seventh National Symposium on Fracture Mechanics, University of Maryland, August 1973.

13. Irwin, G., "Fracture Testing of High-Strength Sheet Material Under Conditions Appropriate for Stress Analysis," U.S. Naval Research Laboratory, Report 5486, July 1960.

14. Dugdale, D. S., "Yielding of Steel Sheets Containing Slits," Journal of Mech. Phys. Solids, 8, 1960.

15. Fracture Toughness Evaluation by R-Curve Methods, ASTM STP-527, 1971.

16. Rice, J. R., "A Path Independent Integral and the Approximate Analysis of Strain Concentration by Notches and Cracks," Journal of Applied Mechanics, June 1968.

17. Newman, J. C., Jr., "Fracture Analysis of Surface- and Through-Cracked Sheets and Plates," Symposium on Fracture and Fatigue, George Washington University, May 1972 (also Engineering Fracture Mechanics Journal, Vol. 5, 1973).

18. Paris, P., "The Fracture Mechanics Approach to Fatigue," Fatigue - An Interdisciplinary Approach, Syracuse University Press, 1964.

19. Forman, R. G., Kearney, V. E., and Engle, R. M., Journal of Basic Engineering, ASME Transactions, Series D, Vol. 89, No. 3, September 1967.

20. Walker, E. K., "The Effect of Stress Ratio During Crack Propagation and Fatigue for 2024-T3 and 7075-T6 Aluminum," ASTM STP 462, 1970.

21. Elber, Wolf, "The Significance of Fatigue Crack Closure," Damage Tolerance in Aircraft Structures, ASTM STP-486, 1970.

22. Erdogan, Fazil, "Crack Propagation Theories," CR-901, 1967, NASA.

23. McEvily, A. J., Jr., and Illg, W., "The Rate of Fatigue Crack Propagation in Two Aluminum Alloys," TN 4394, 1958, NACA.

24. Elber, Wolf, "Fatigue Crack Closure Under Cyclic Tension," Engineering Fracture Mechanics Journal, Vol. 2, 1970.

25. Wheeler, O. E., "Spectrum Loading and Crack Growth," Transactions of the ASME, Journal of Basic Engineering, 1971.

26. Willenborg, J., Engle, R. M., and Wood, H. A., "A Crack Growth Retardation Model Using an Effective Stress Concept," AFFDL, TM 71-1-FBR, January 1971.

27. Tiffany, C. F., and Masters, J. N., "Applied Fracture Mechanics," Fracture Toughness Testing and Its Applications, ASTM STP-381, 1964.

28. Elber, Wolf, and Davidson, John R., "A Material Selection Method Based on Material Properties and Operating Parameters," TN D-7221, April 1973, NASA.

29. Lisagor, W. B., Manning, C. R., Jr., and Bales, T. T., "Stress-Corrosion Cracking in Ti-6Al-4V Titanium Alloy in Nitrogen Tetroxide," TN D-4289, January 1968, NASA.

30. Poe, C. C., Jr., "Stress Intensity Factors for a Cracked Sheet with Riveted and Uniformly Spaced Stringers," TR R-358, 1971, NASA.

31. Poe, C. C., Jr., "The Effect of Broken Stringers on the Stress Intensity Factor for a Uniformly Stiffened Sheet Containing a Crack," Presented at the 10th Anniversary Meeting of the Society of Engineering Science, November 1973.

32. Packman, P. F., Pearson, H. S., Owens, J. S., and Marchese, G. B., "The Applicability of a Fracture Mechanics-Nondestructive Testing Design Criterion," Report No. AFML-TR-68-32, May 1968.

33. Knorr, Ekkart, "Reliability of the Detection of Flaws and of the Determination of Flaw Size," Chapter IIIA of the AGARD Fracture Mechanics Survey, H. Liebowitz, Editor, to be published in 1974.

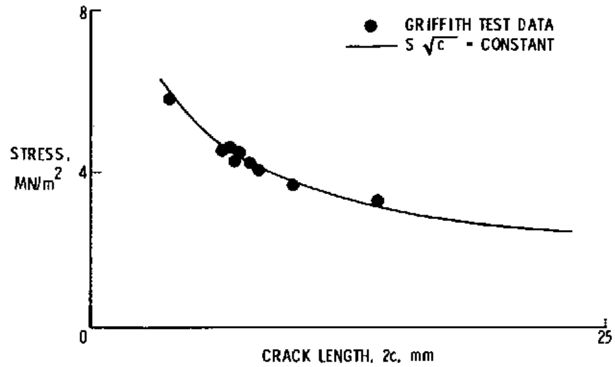


Figure 1. Griffith test results for fracture of glass.

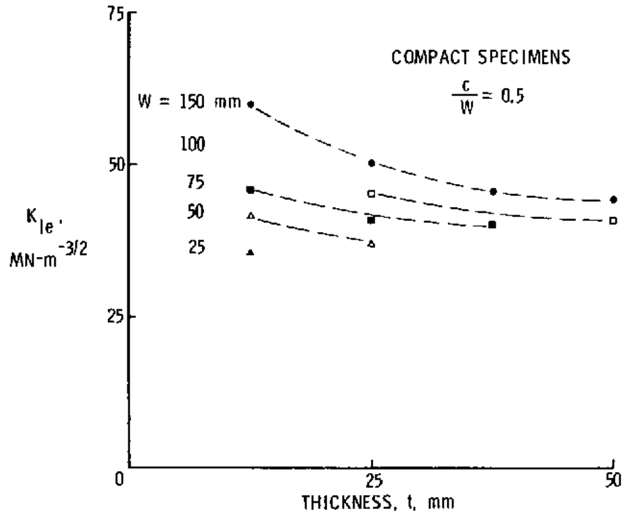
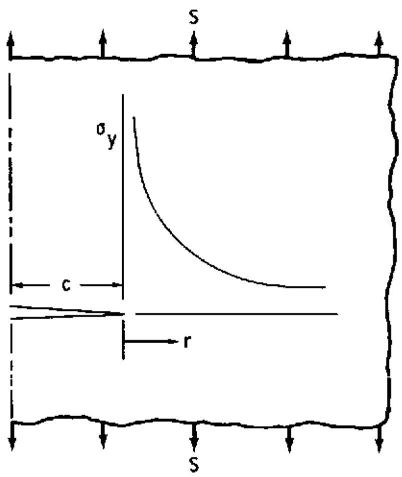


Figure 4. Observed fracture toughness for 2219-T851 aluminum alloy (Ref. 12).



$$\sigma_y = \frac{K}{\sqrt{2\pi r}} + S + \cdots$$

$$K = S\sqrt{\pi c}$$

Figure 2. Stress intensity analysis.

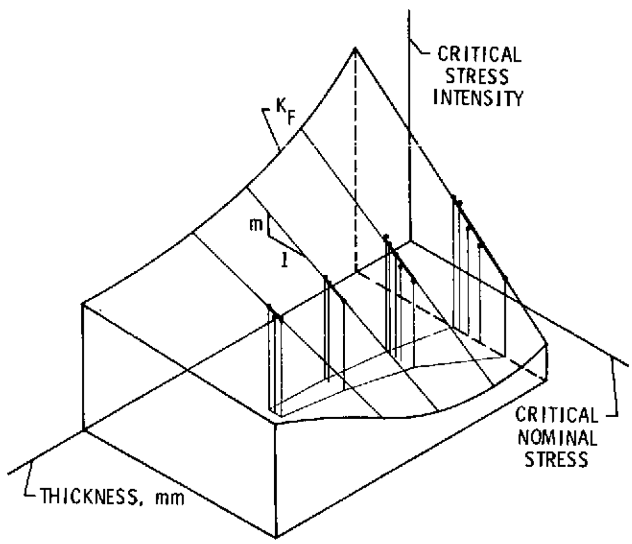


Figure 5. Expected fracture toughness for 2219-T851 aluminum alloy (data from Ref. 12, predictions from Ref. 17).

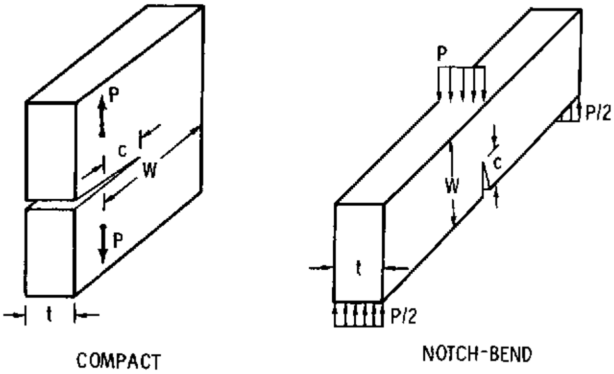


Figure 3. ASTM standard fracture toughness specimens.

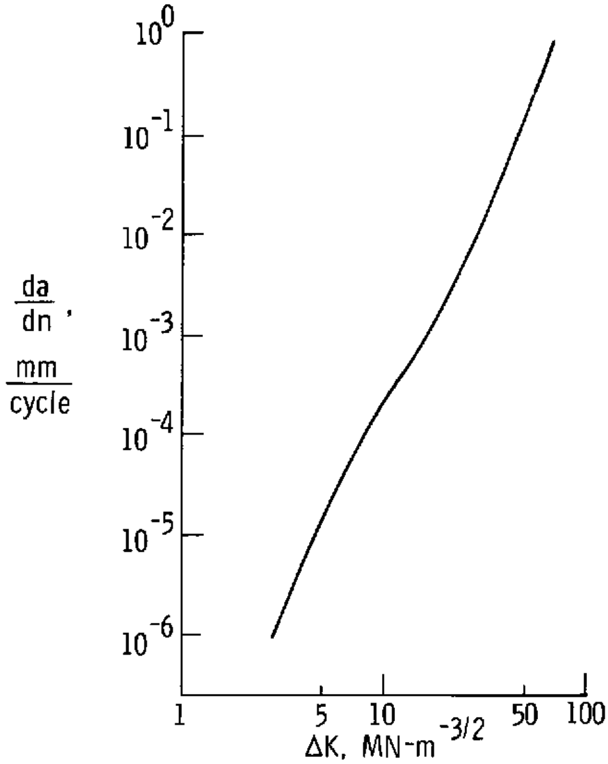


Figure 6. Typical fatigue crack propagation data.

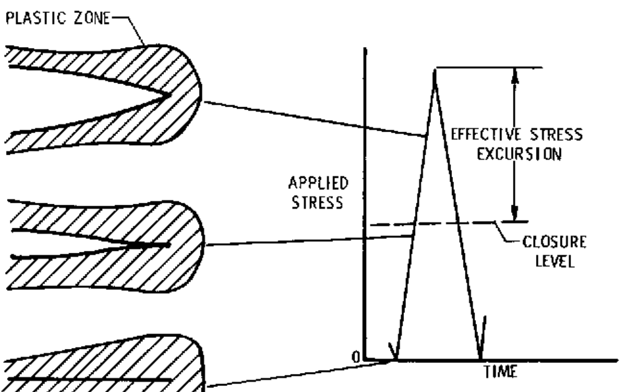


Figure 7. Crack closure concept.

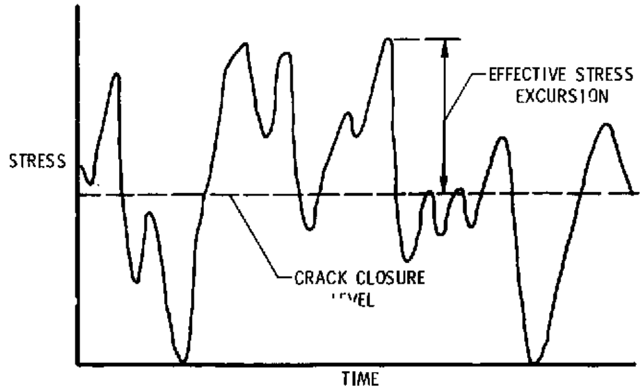


Figure 8. Effective stress excursions in complex time history.

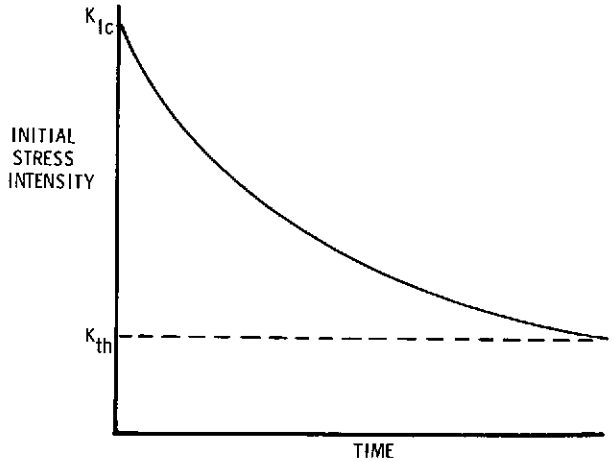
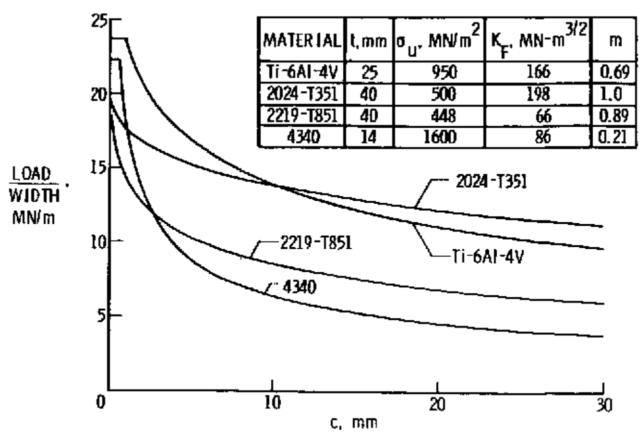


Figure 9. Typical stress corrosion cracking data.



9,

Figure 10. Fracture strength of equal weight panels made of four structural materials (W >> 1).

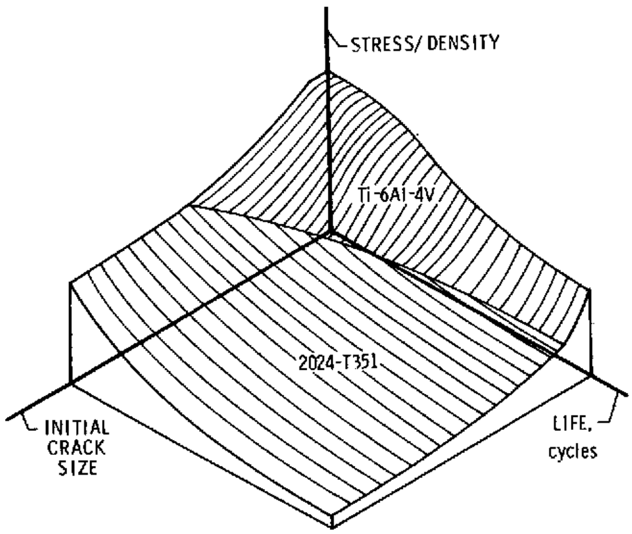


Figure 11. Better choice of two materials for specified life and initial crack.

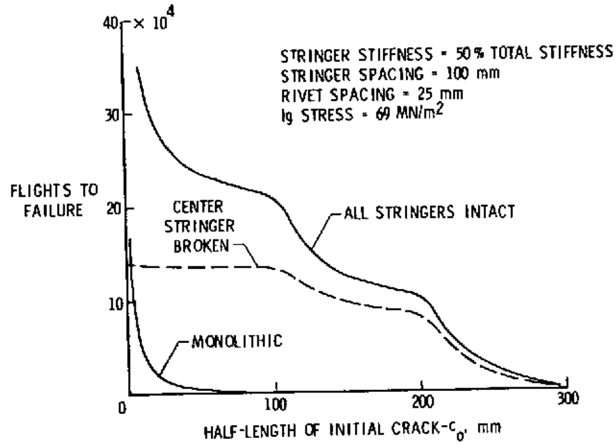


Figure 13. Crack propagation life for 7075-T6 aluminum alloy panels.

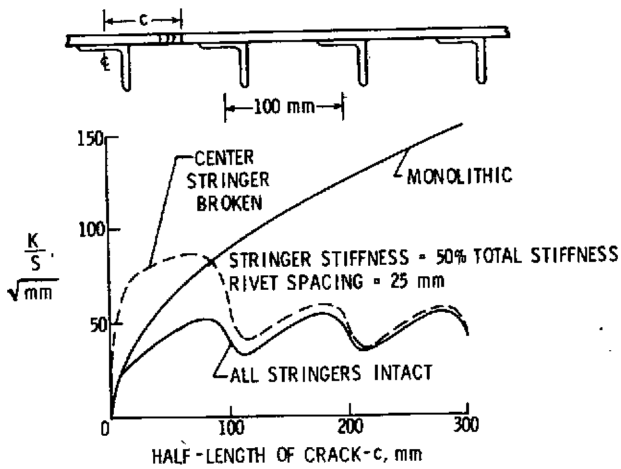


Figure 12. Stress intensity for crack in stiffened panels.

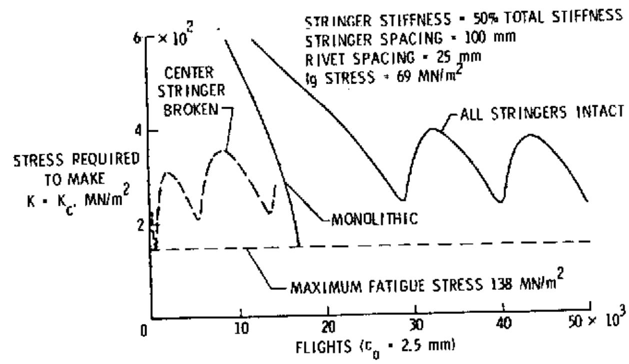


Figure 14. Fracture in stiffened 7075-T6 aluminum alloy panels.