# POTENTIAL STRUCTURAL MATERIALS AND DESIGN CONCEPTS FOR LIGHT AIRCRAFT

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## INTRODUCTION

This four-part paper is based on a study conducted by San Diego Aircraft Engineering Company for NASA, Mission Analysis Division, Ames Research Center. The complete report of the study was published as NASA CR-1285, March 1969; a summary report was published as NASA CR-73257.

The series of papers presented here contains material of possible interest to sailplane designers and builders. The NASA report CR-1285 is available for sale @ $3.00 through CFSTI, Springfield, Virginia 22151.

Part I was presented in Technical Soaring, Vol. I, No. 4, April 1972, and Part II in Vol.II, No.1, July 1972. Part IV will appear in a forthcoming edition of this publication.

## PART III

## FATIGUE CONSIDERATIONS

Existing requirements for the strength of light airplane structures are based largely on the concept of "one-time" loading. For many years, this appeared to be satisfactory, but recently it has been recognized that the margin of safety provided against failure under "one-time" loading may no longer be adequate with respect to the repeated loads which occur during the lifetime of the aircraft. A survey of the 1963 General Aviation Accident Reports indicates evidence that some airframe failures could be attributed to fatigue.

Whether or not the failures involved were the result of inadequate pilot proficiency, lack of respect for adverse weather, or the result of inadequate inspection and maintenance is of secondary importance. The point is that the airplane involved encountered flying conditions which resulted in loads being applied to the airframe of sufficient magnitude and frequency to cause catastrophic failure of the primary airframe structure.

### Establishing a Fatigue Load Spectrum

Up to the present time, light airplane manufacturers have designed their aircraft to FAA requirements per F.A.R. part 23. This document does not require proof by analysis or test of the "safe life" or "fail safe" characteristics of their aircraft. At the same time, little data is available with regard to what load spectra should be used by operators of the various category airplanes.

An assessment of repeated loads on general aviation and transport aircraft is being conducted with the FAA by NASA's Langley Research Center; the results to date are presented in Ref. 1 and 2. They reveal a large amount of scatter in the repeated load history, due principally to the diverse nature of general aviation.

Composite VG records (positive and negative accelerations vs airspeed) from Ref. 1 and 2 for different types of operations are presented in Fig. 1. These data are superimposed upon their respective V-n diagrams to indicate where the most severe areas might be in respect to possible exceedances of the design flight envelope. Design flight envelope exceedances in the low speed portions are probably due to landing shocks and are not considered significant.

A review of the instructional flying records (Fig. 1) reveals a case where a particular aircraft exceeded the design dive speed as well as the positive and negative limit load factors at the design dive speed.

The twin-engine executive operations (Fig. 1) show one case of exceeding the negative limit load factor at a speed slightly less than design cruise. Investigation revealed the incidence to be gust induced.

The following significant conclusions can be made after reviewing the composite VG records.

(1) Atmospheric-induced, as well as pilot-induced, loads in excess of the design flight envelope may be encountered during normal operation of the general aviation fleet.

(2) All types of operations are flown above the design cruising speed.

It is evident, therefore, that General Aviation should be classified into different roles. Needless to say, the fatigue load spectrum will be different for each role.

Estimation of Fatigue Life

The estimation of fatigue life using the "Miners" Cumulative Damage Rule involves the calculation of damage incurred on the airplane as a direct result of its operating environment.

Generally speaking, the operating environment for a light airplane, regardless of its type of utilization such as executive, personal, instructional, or commercial survey operation, can be defined as follows:

(1) Gust Environment - The airplane, while in steady flight, encounters a specified number of positive and negative gusts of varying intensities defined by the gust spectrum for the airplane.

(2) Maneuver Environment - The airplane is subject to a specified number of positive and negative maneuvering loads of varying intensity defined by the maneuvering spectrum for the airplane.

(3) Ground-Air-Ground Environment (G.A.G.) - At least once per flight, the airplane is subject to loads associated with the following conditions.

(a) Taxi condition at maximum take-off weight.

(b) Steady 1g Flight Cruise Condition at minimum landing weight.

(c) Landing impact loads at maximum landing weight.

From a structural design aspect, it is apparent that before any design fatigue load spectrum can be developed and before any safe life prediction can be made it is necessary to define not only in what roles that airplane is going to be utilized, but also for how long it is going to be utilized in one role before being used in another role. This is obvious when one is confronted by the following statements:

(1) Landing Impact Acceleration for instructional-type airplanes is more severe and more frequent, approximately 4 per 30-minute flight, than on any other category light airplane and will account for a considerable amount of damage in the fatigue life of the airplane.

(2) Commercial Survey Aircraft have the longest flight duration; therefore, less G.A.G. damage is inflicted on the airplane. They have more severe gust experience than other types of usage, since 97% of the time they air in rough air.

COMPOSITE VG RECORDS - FIVE TYPES OF OPERATIONS

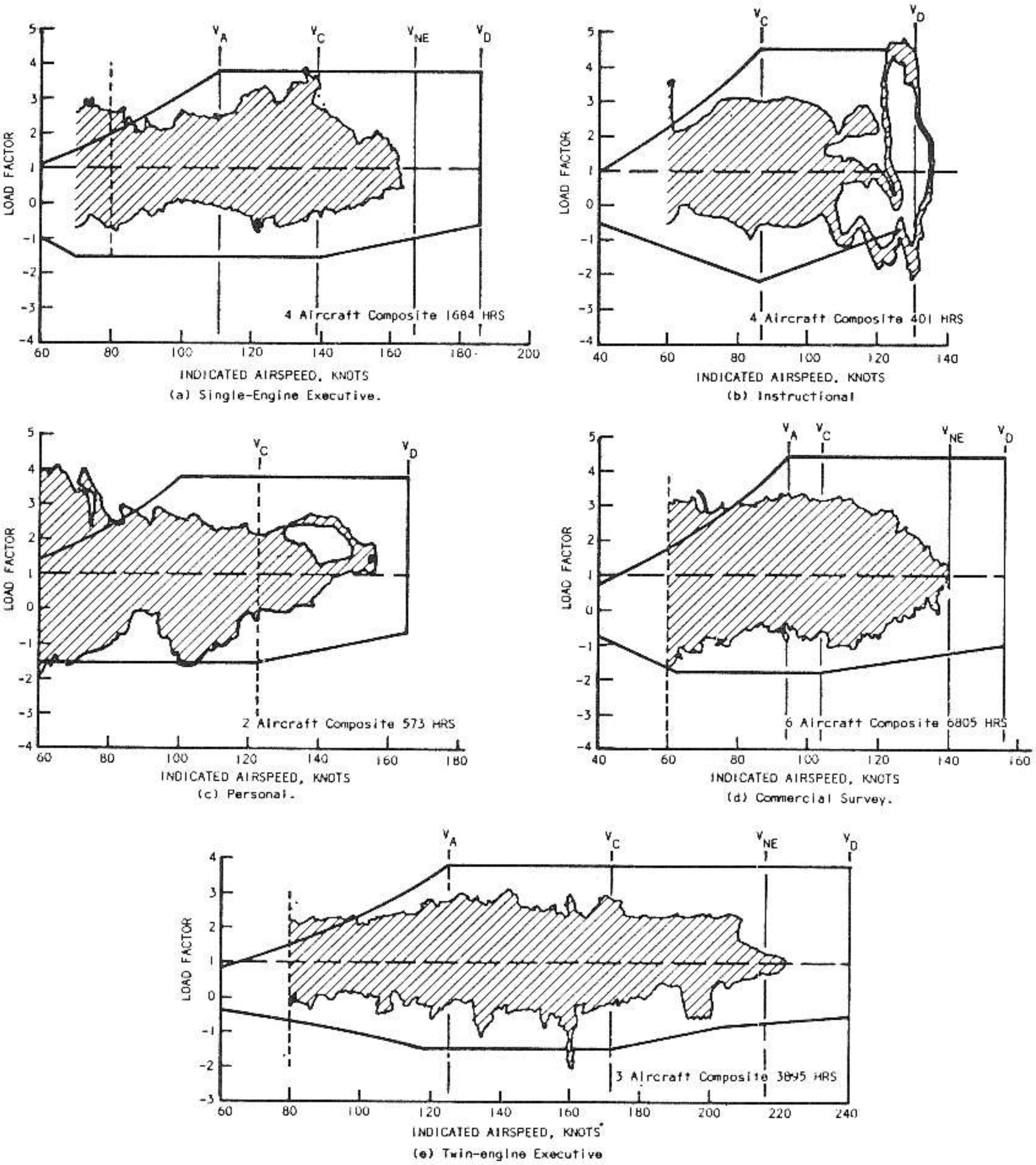



FIGURE 1

## Pressurization Considerations

The effect of pressurization produces a stress configuration consisting of hoop stress and longitudinal stress in addition to the in-flight shear, bending moment, and torque loads on the fuselage structure. It then follows that the weight of the basic pressurized fuselage will be higher than that of an unpressurized fuselage. From a minimum weight standpoint, the optimum structure is cylindrical with the elimination of flat or slab panels.

Sealing requirements demand that careful consideration be given to the number and spacing of rivets, particularly at longitudinal and transverse skin splices and at the attachment of pressure bulkheads and canopy structure. Likewise, more care must be taken in the fabrication, inspection, and quality control of the fuselage structure, particularly in the region of cut outs in the structure for windows, entry doors, and access doors, at the attachment of the floor structure to the frames of the fuselage, and at the intersection of the wing and fuselage.

Entry doors and their locking and operating mechanisms should be designed on the fail safe concept to insure that the door structure and the sealing qualities are adequate should a simple failure in one of the latches or shear pins occur.

The use of metal-to-metal adhesive bonding, particularly to reinforce areas where high stress concentrations are present, increases the fatigue life of the fuselage. It demands good quality control and considerable component testing. Materials exhibiting low crack propagation characteristics are important. As an example, it has been shown (Ref. 3) that 7075-T6 aluminum alloy is more prone to explosive fracture than 2024-T3 alloy.

From a structural standpoint, it is highly probable that any fatigue crack, once started, will tend to run longitudinally along the fuselage. This is due to the fact that, in a pressurized fuselage, the stringers are fairly closely spaced and the hoop tensile stress is twice the longitudinal stress. For this reason, circumferential reinforcing rings are placed at intervals along the fuselage to arrest the crack propagation of a fatigue crack and to reduce the hoop stress in the skin.

The spacing and cross section of the reinforcing rings are important. Williams (Ref. 4) states that rings spaced more than 30 inches apart, while locally restricting the radial expansion of the skin, allow unrestricted expansion in the area midway between the rings; with a 10-inch spacing, the radial expansion of the skin nowhere exceeds that of the rings by more than a small percentage, so that the maximum hoop stress in the skin is equally reduced by material added to the rings as by the weight added to the skin.

## Material Fatigue Properties

Many mechanical devices are subjected to forces that vary in magnitude and, often, in direction. If this variation occurs a relatively small number of times and the stresses do not exceed the yield strength of the material, design studies can be made safely on the basis of the static properties of the material. Unfortunately, this is not true in the design of airplanes, since the structure usually experiences many repeated loadings (magnitude and direction) in its service lifetime.

This section summarizes and compares the fatigue properties of some of the basic materials as previously selected for aircraft structural applications. This data has been compiled and evaluated to present a qualitative picture of the fatigue characteristics associated with the material.

For the most part, complete information was not available for the materials; therefore, various methods were utilized in extending the data to provide information which could not be obtained directly. All the fatigue data shown represents axial loading tests and is ultimately plotted as standard S-N curves, whereby the points along the curve represent the number of loading cycles a material may endure at a particular max stress before failure.

S-N curves for notched and unnotched sheet specimens representing stress ratios (R) of -1.0 and +0.25 are shown in Fig. 2, 3, 4, and 5. For the most part, these

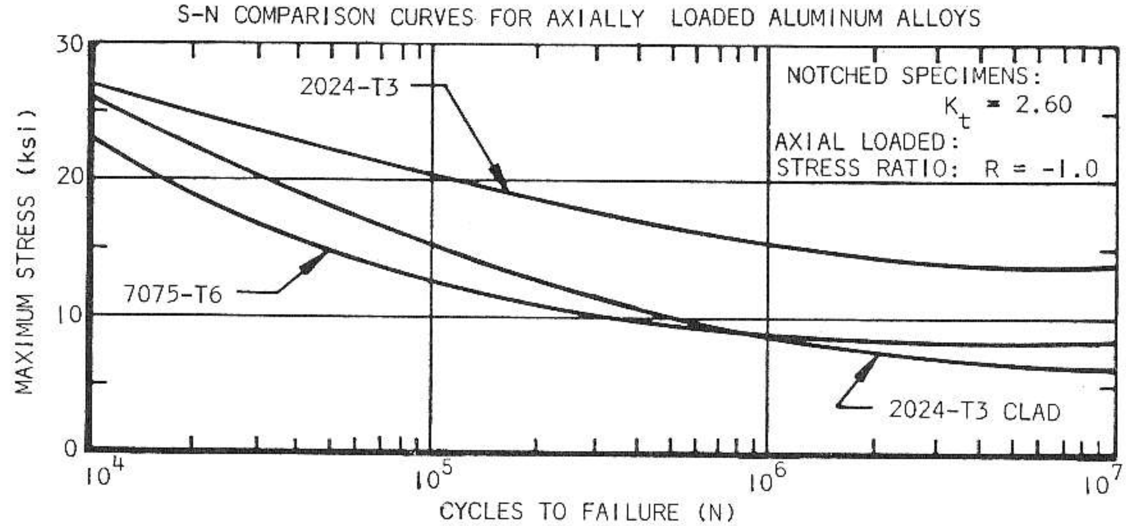



FIGURE 2

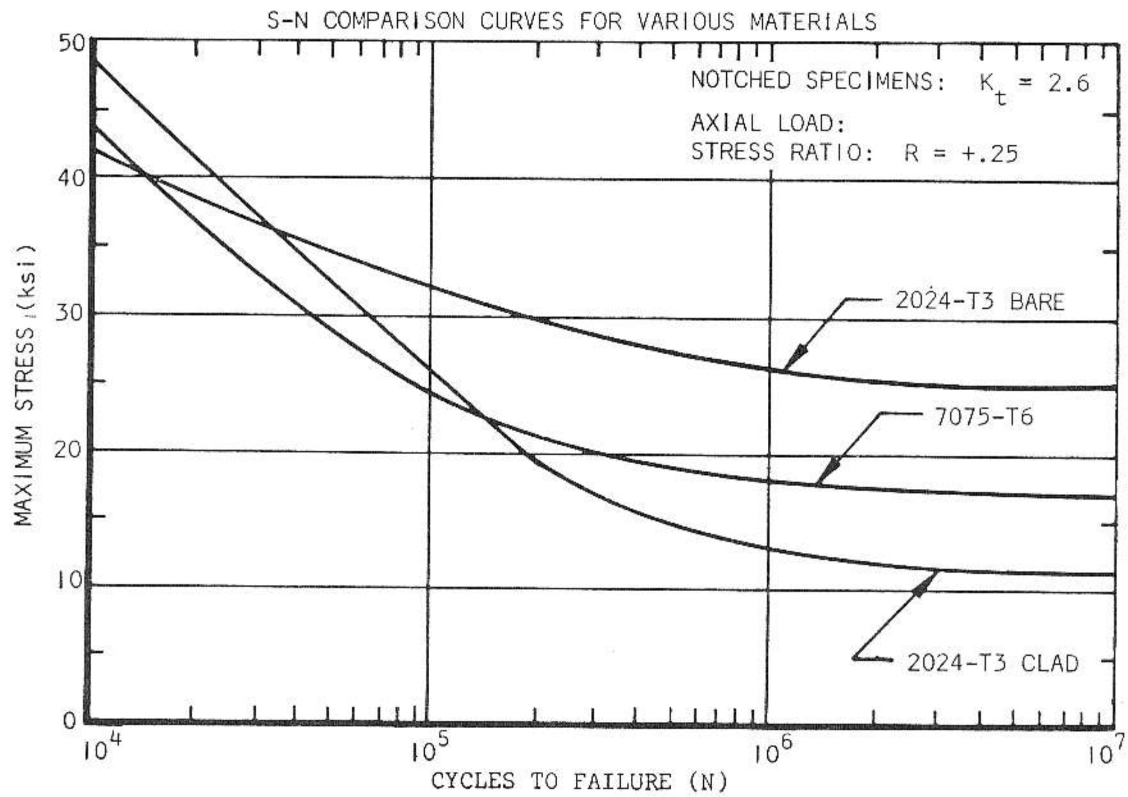
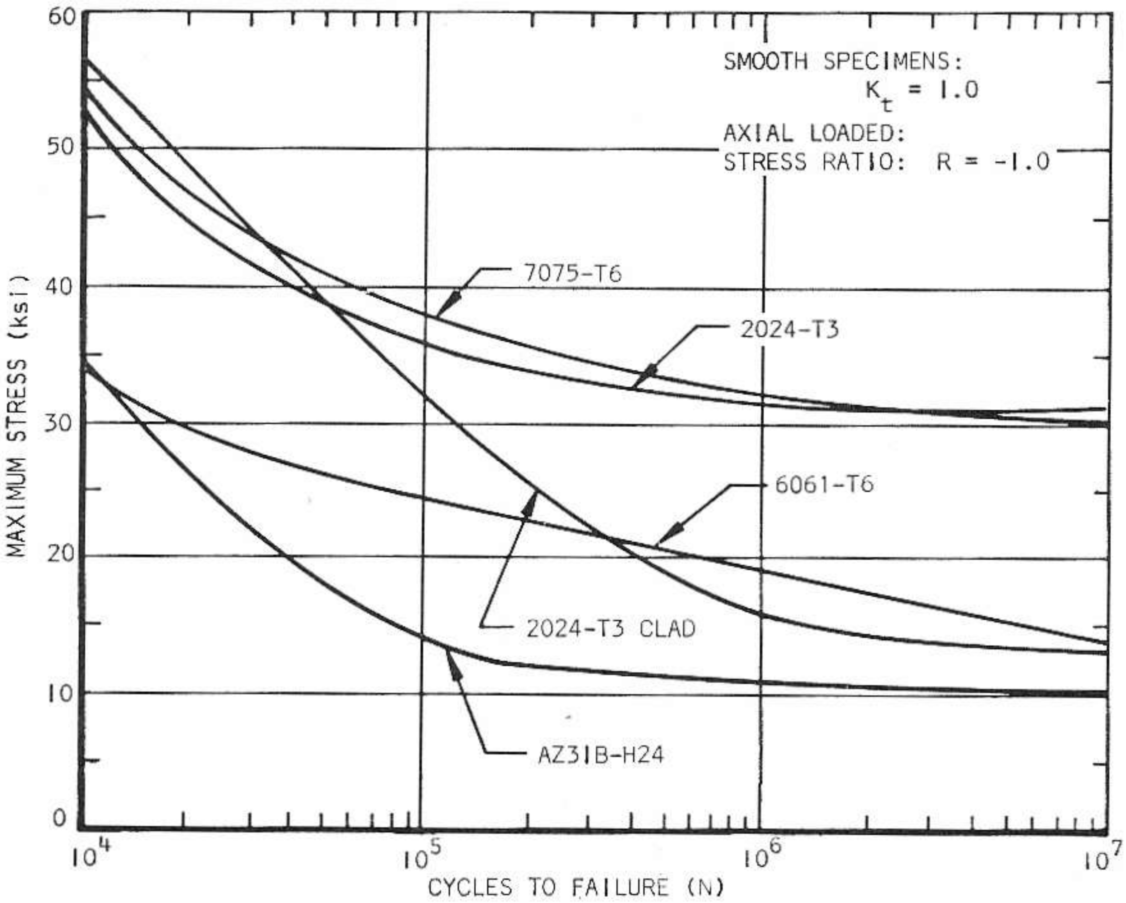



FIGURE 3

FIGURE 4

S-N COMPARISON CURVES FOR AXIALLY LOADED ALUMINUM ALLOYS




S-N COMPARISON CURVES FOR VARIOUS MATERIALS

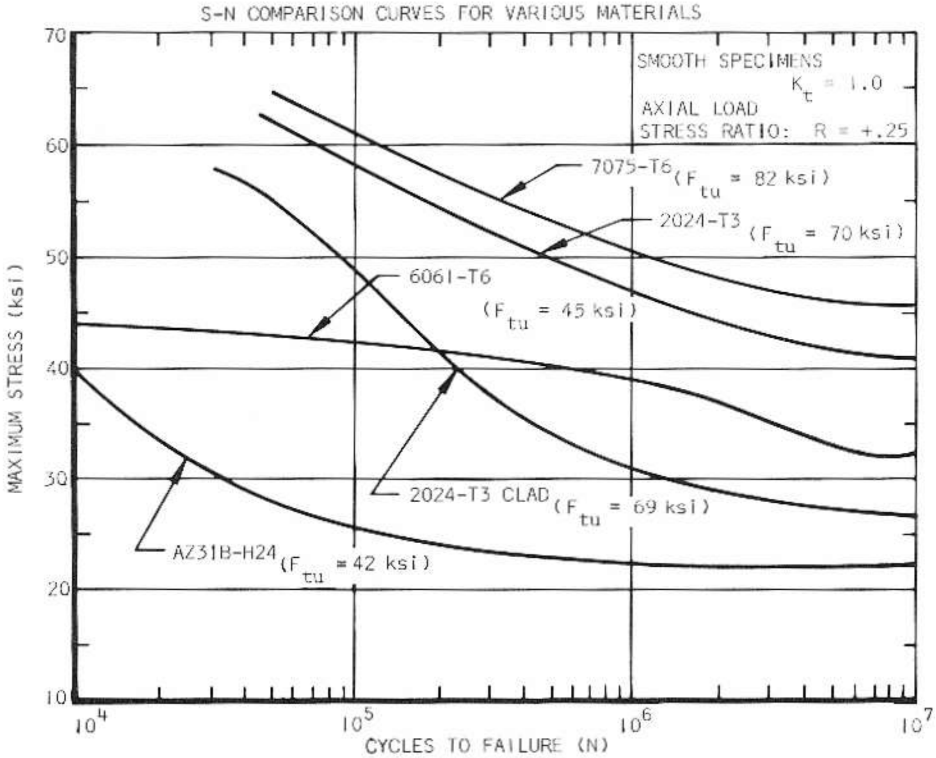



FIGURE 5

curves are derived by means of averaging the results directly from several references as shown in the respective tables.

Where basic information in the reference did not provide data representing correct stress ratios from which comparisons could be made, the basic data is expanded through use of an approximate Modified Goodman diagram. This method is described in Ref. 5.

The reference literature (Ref. 6) associated with the 4130 and 4340 materials provided fatigue data in terms of alternating and mean stress. With use of modified Goodman Diagrams, it is possible to reconstruct S-N curves (Fig. 6 and 7) as a function of maximum-stress and any stress ratio desired.

Fig. 3 illustrates that the fatigue strength of the higher strength aluminum alloy (7075-T6) actually is inferior to the lower strength alloys. This would suggest that increases in static strength have been obtained at the expense of an actual reduction in fatigue strength.

This is not true in the comparison of 4340 and 4130 steels; however, the difference in the static strength of these two materials is much greater than the difference in the fatigue strengths (ref. Fig. 6 and 7).

Comparison S-N curves for plastic laminates reinforced with unwoven glass filaments are presented in Fig. 8, 9, and 10. The curves represent three constructions: (1) all plies parallel, (2) alternate plies $\pm 5^\circ$ to the principal axis, (3) alternate plies $0^\circ$ and $90^\circ$ to the principal axis. All indicate the fatigue strength of the S-glass filaments to be superior to the E-glass type. It also appears (Fig. 9 and 10) that the fatigue characteristics of the S-glass laminates may be even further improved with the use of different resins.

In recent years, more and more consideration is being directed toward the fracture characteristics of materials. Acceptance is given to the fact that fatigue failures could occur as a result of one or a combination of several loading environments. These environments include normal working loads, noise-induced vibrations, and accidental damage. When a crack originally develops in a structure, it creates a point of high stress concentration, and subsequent application of normal service loads will cause further extension of the crack. This extension, of course, largely depends upon the load/stress level and the inherent crack-propagation characteristic of the material. It is extremely important these cracks be detected before they can extend to a length which would cause a catastrophic failure. Structural inspections take place periodically, and consist of frequent visual examinations to detect any obvious defects, together with a detailed overhaul about once a year.

Two questions which still need answering are as follows:

(1) How long must a crack be before it can be detected?

(2) How long can it become before it leads to serious failure?

The ideal condition would be such that a defect which is approaching a detectable length would not become catastrophic prior to the next scheduled inspection. A good design would therefore consider a material which would satisfy these requirements; i.e., low crack-propagation rate to allow sufficient time for crack detection and high notch resistance to insure adequate strength at any crack location. These requirements actually have led to a return to the use of lower strength aluminum alloys, particularly in fatigue critical areas.

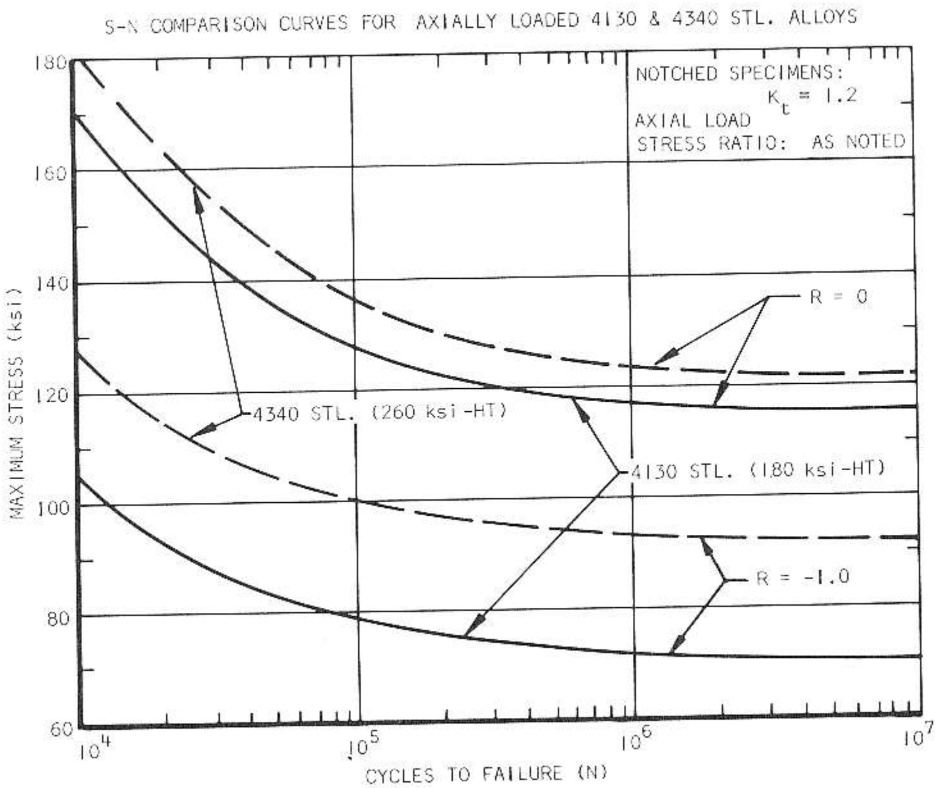



FIGURE 6

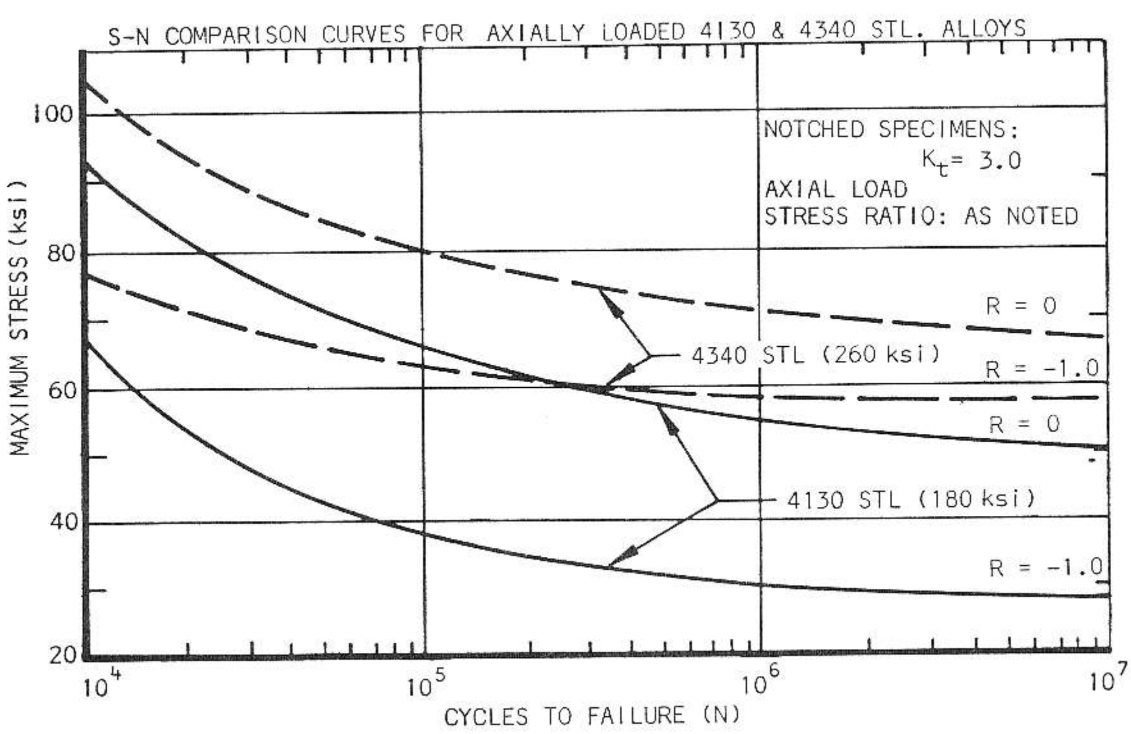



FIGURE 7

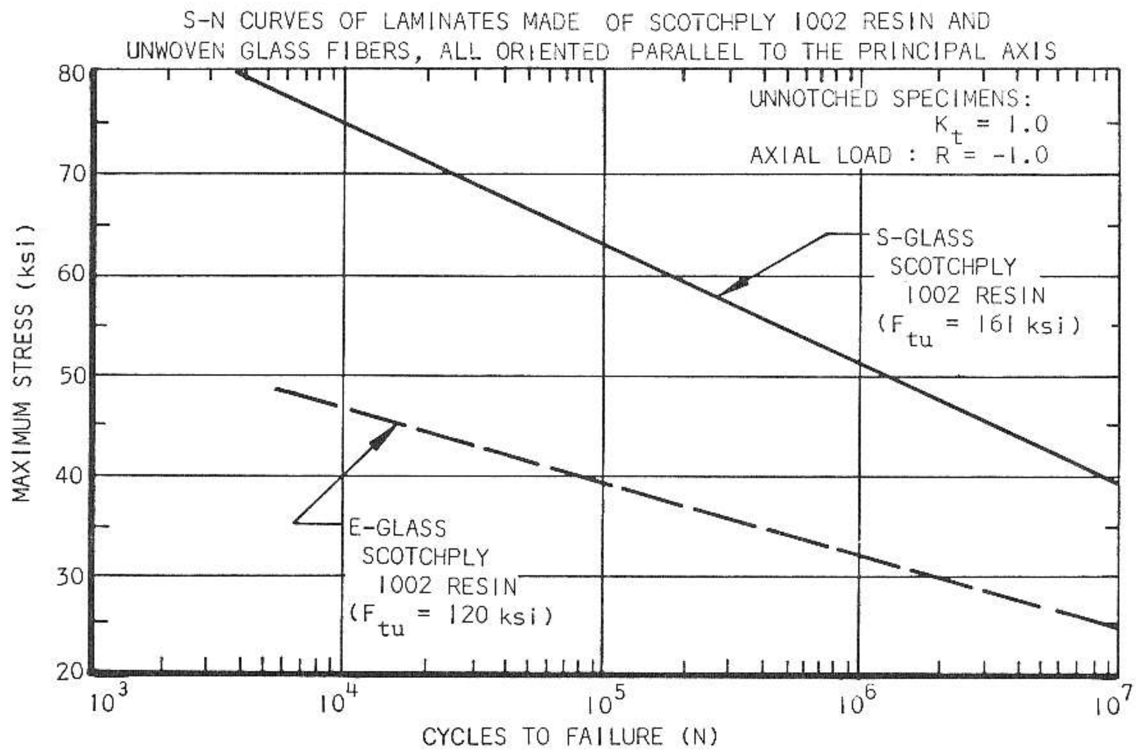



FIGURE 8

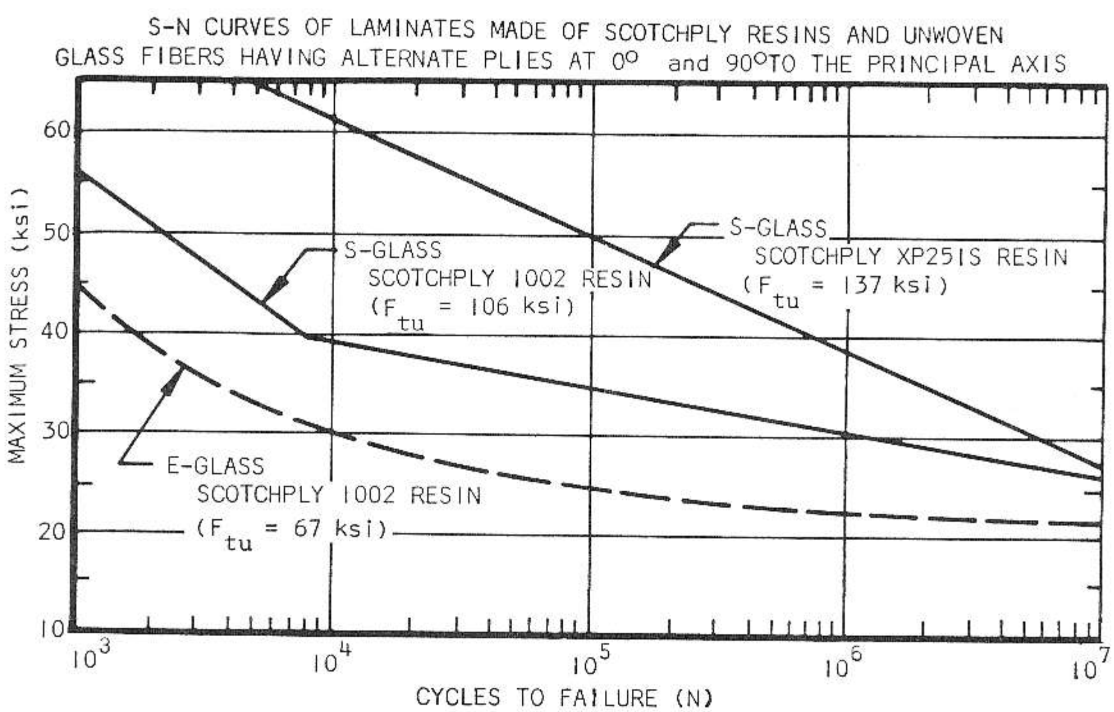



FIGURE 9

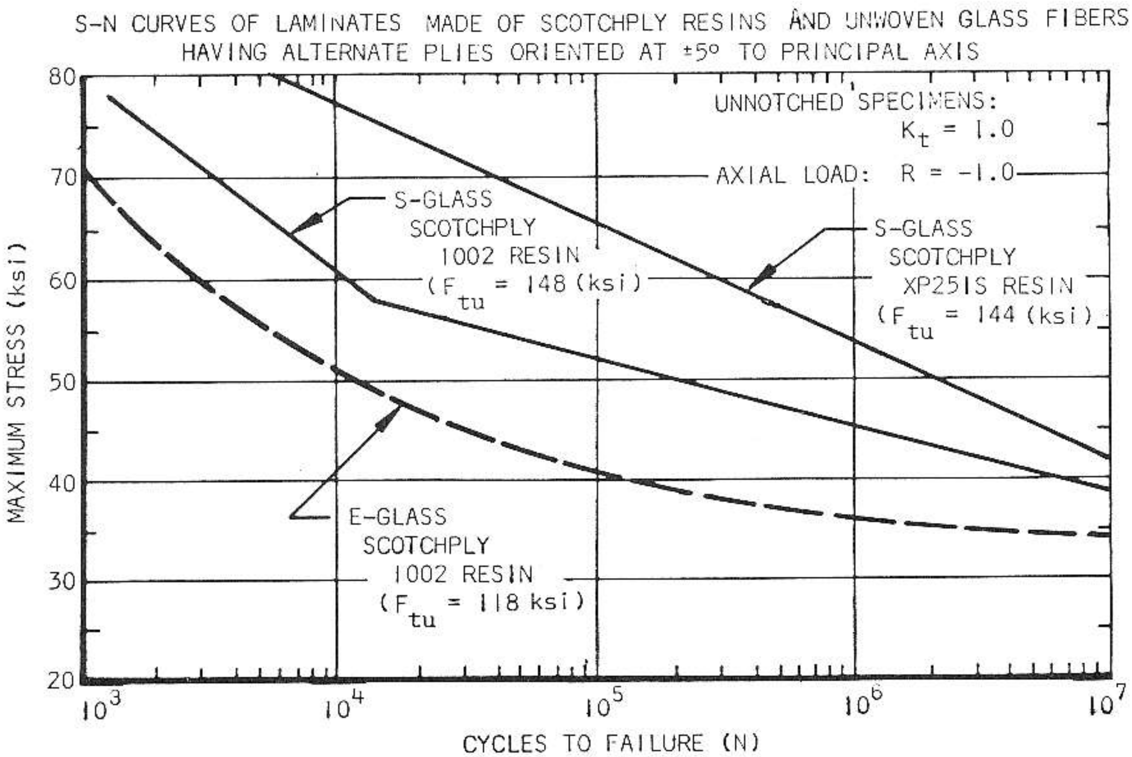



FIGURE 10

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END OF PART III