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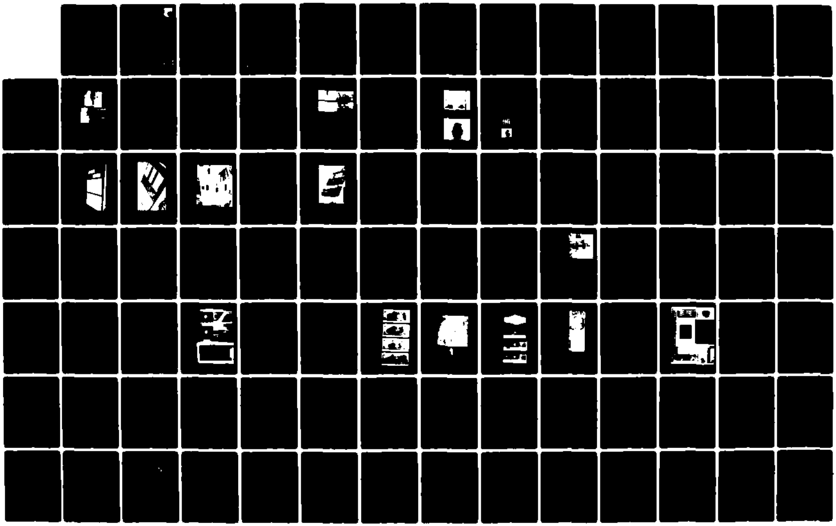

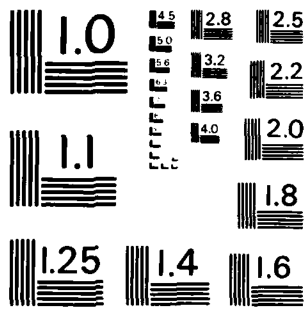

MICROCOPY RESOLUTION TEST CHART
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AFWAL-TR-83-4005

PROCEEDINGS OF THE EIGHTH ANNUAL MECHANICS OF COMPOSITES REVIEW

Lisa A. Wilson

Mechanics and Surface Interactions Branch
Nonmetallic Materials Division

April 1983

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MATERIALS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
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Lisa A. Wilson

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Conference Coordinator
Eighth Annual Mechanics of Composites Review

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FRANKLIN D. CHERRY, Chief
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20 ABSTRACT (Continue on reverse side if necessary and identify by block number)

This report contains summaries of the presentations of the Eighth Annual Mechanics of Composites Review sponsored by the Materials Laboratory. Each paper was prepared by its presenter and is published here unedited. In addition to the presenter's summaries, a listing of both the inhouse and contractual activities of each participating organization is included.

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FOREWORD

This report contains the abstracts and viewgraphs presented at the Eighth Annual Mechanics of Composites Review sponsored by the Materials Laboratory. Each was prepared by its presenter and is published here unedited. In addition, a listing of both the inhouse and contractual activities of each participating organization is included.

The Mechanics of Composites Review is designed to present programs covering activities throughout USAF and NASA. Programs not covered in the present review are candidates for presentation at future mechanics of composites reviews. The presentations cover both inhouse and contract programs under the sponsorship of the participating organizations.

Since this is a review of on-going programs, much of the information in this report has not been published as yet and is subject to change; but timely dissemination of the rapidly expanding technology of advanced composites is deemed highly desirable. Works in the area of mechanics of composites have long been typified by disciplined approaches. It is hoped that such a high standard of rigor is reflected in the majority, if not all, of the presentations in this report.

Feedback and open critique of the presentations and the review itself are most welcome as suggestions and recommendations from all participants will be considered in the planning of future reviews.

We express our appreciation to the authors for the contribution of their summaries and to the points of contact within the organizations for their effort in supplying the program listings.

AGENDA

MECHANICS OF COMPOSITES REVIEW

5-7 OCTOBER 1982

AD P001244

POSTBUCKLING BEHAVIOR OF GRAPHITE-EPOXY PANELS LOADED IN COMPRESSION

BY

JAMES H. STARNES, JR.; MARSHALL ROUSE; MANUEL STEIN
AND NORMAN F. KNIGHT, JR.

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HAMPTON, VIRGINIA 23665

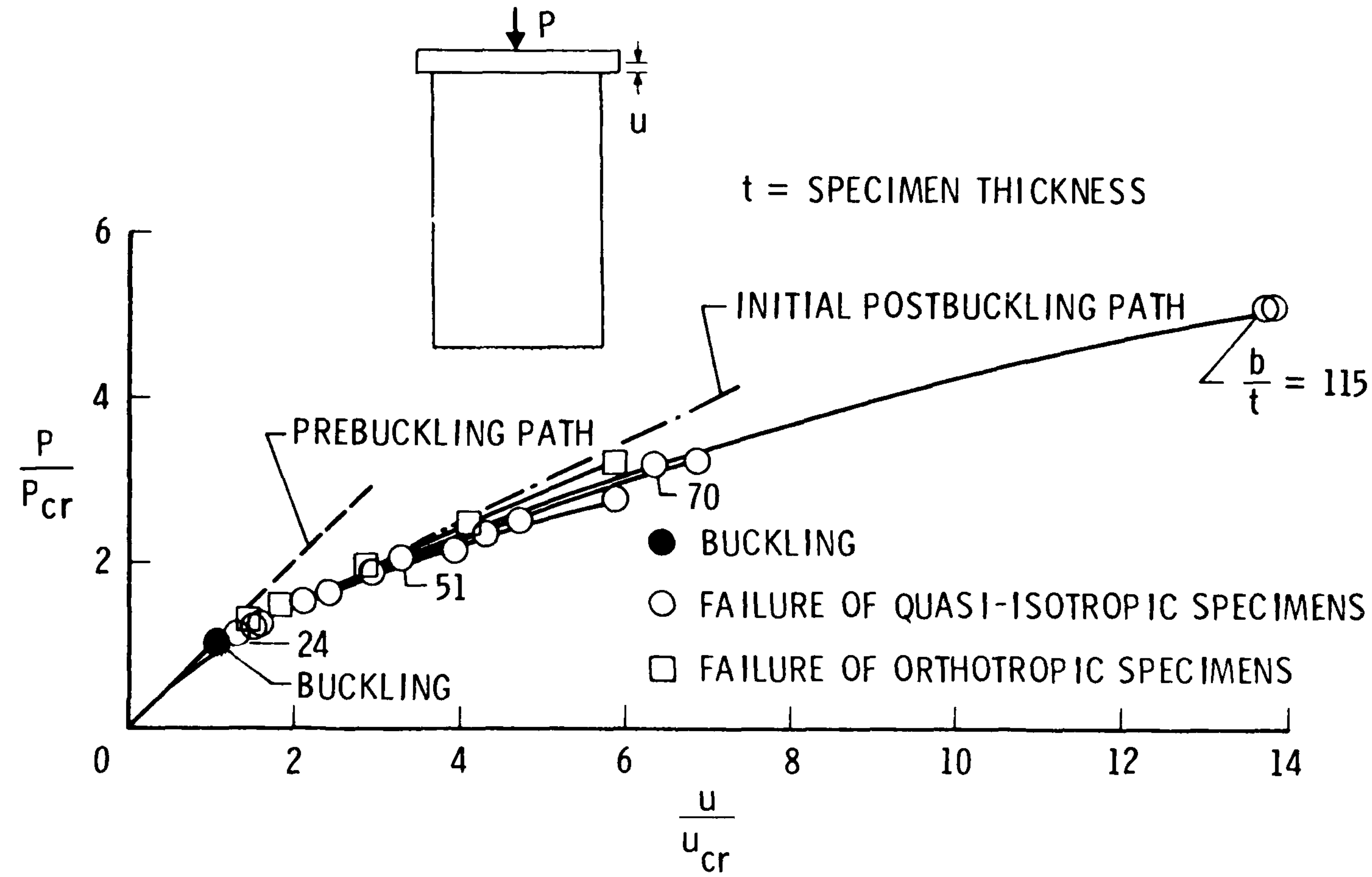
OBJECTIVES

- TO STUDY THE POSTBUCKLING BEHAVIOR OF UNSTIFFENED AND STIFFENED GRAPHITE-EPOXY PANELS LOADED IN COMPRESSION;
- TO IDENTIFY THE FAILURE MODES OF COMPRESSIVELY-LOADED GRAPHITE-EPOXY PANELS WITH POSTBUCKLING STRENGTH;
- TO PREDICT ANALYTICALLY THE POSTBUCKLING RESPONSE OF GRAPHITE-EPOXY PANELS LOADED IN COMPRESSION.
- TO DETERMINE THE EFFECTS OF LOW-SPEED IMPACT DAMAGE AND CIRCULAR HOLES ON POSTBUCKLING STRENGTH.

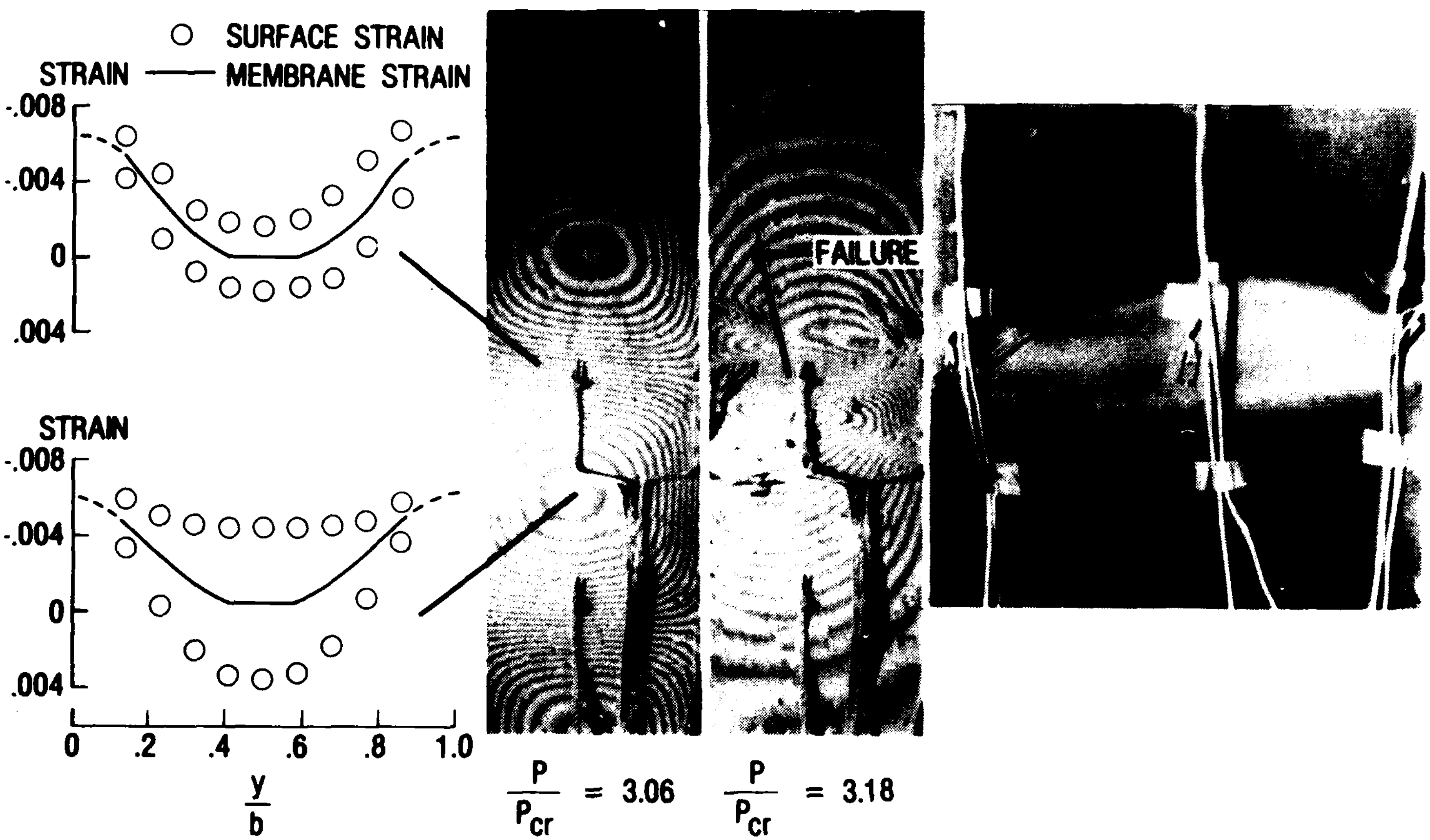
CONCLUDING REMARKS

- FARAMETERS GOVERNING THE POSTBUCKLING BEHAVIOR OF ORTHOTROPIC PLATES HAVE BEEN DETERMINED;
- POSTBUCKLING STRENGTH IS LIMITED BY HIGH TRANSVERSE SHEAR FORCES AND SKIN-STIFFENER SEPARATION;
- POSTBUCKLING RESPONSE CAN BE PREDICTED ACCURATELY;
- CIRCULAR HOLES AND LOW-SPEED IMPACT DAMAGE CAN REDUCE POSTBUCKLING STRENGTH.
- IMPACT DAMAGE LOCATION CAN ALSO INFLUENCE POSTBUCKLING STRENGTH.

POSTBUCKLING RESPONSE OF GRAPHITE-EPOXY PLATES LOADED IN COMPRESSION



STRAIN DISTRIBUTION AND FAILURE OF A GRAPHITE-EPOXY PLATE



PARAMETERS GOVERNING POSTBUCKLING OF ORTHOTROPIC PLATES

DIFFERENTIAL EQUATIONS OF EQUILIBRIUM

$$\frac{b^2}{a^2}\sqrt{\frac{A_{11}}{A_{22}}}\,F_{,xxxx} + 2\left(\frac{A_{11}A_{22} - A_{12}^2 - 2A_{12}A_{66}}{2A_{66}\sqrt{A_{11}A_{22}}}\right)F_{,xxyy} + \frac{a^2}{b^2}\sqrt{\frac{A_{22}}{A_{11}}}\,F_{,yyyy}$$

$$= w_{,xy}^2 - w_{,xx}w_{,yy}$$

$$\frac{b^2}{a^2}\sqrt{\frac{D_{11}}{D_{22}}}\,w_{,xxxx} + 2\left(\frac{D_{12} + 2D_{66}}{\sqrt{D_{11}D_{22}}}\right)w_{,xxyy} + \frac{a^2}{b^2}\sqrt{\frac{D_{22}}{D_{11}}}\,w_{,yyyy}$$

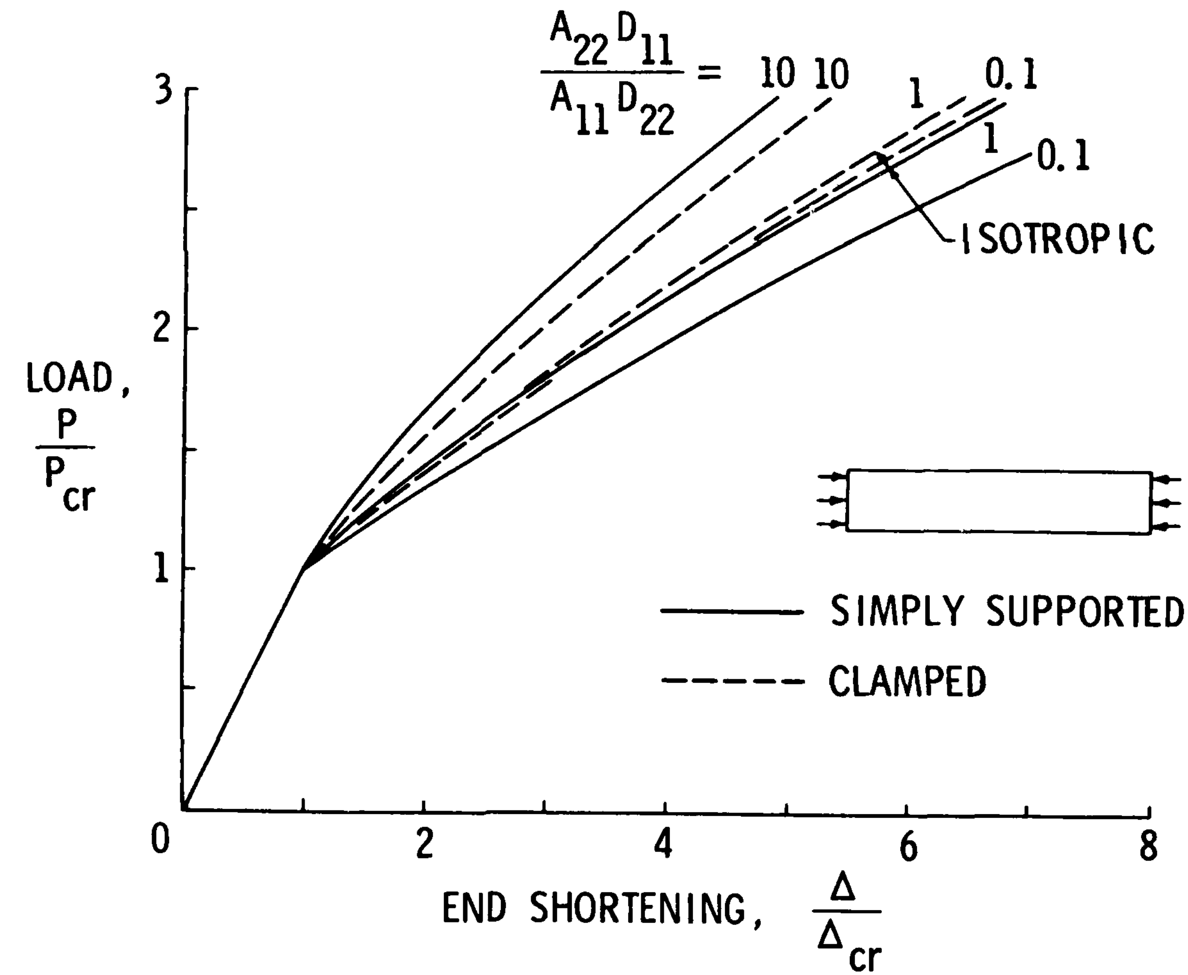
$$= F_{,yy}w_{,xx} + F_{,xx}w_{,yy} - 2F_{,xy}w_{,xy}$$

NEW PARAMETERS FOR POSTBUCKLING MAY BE TAKEN TO BE

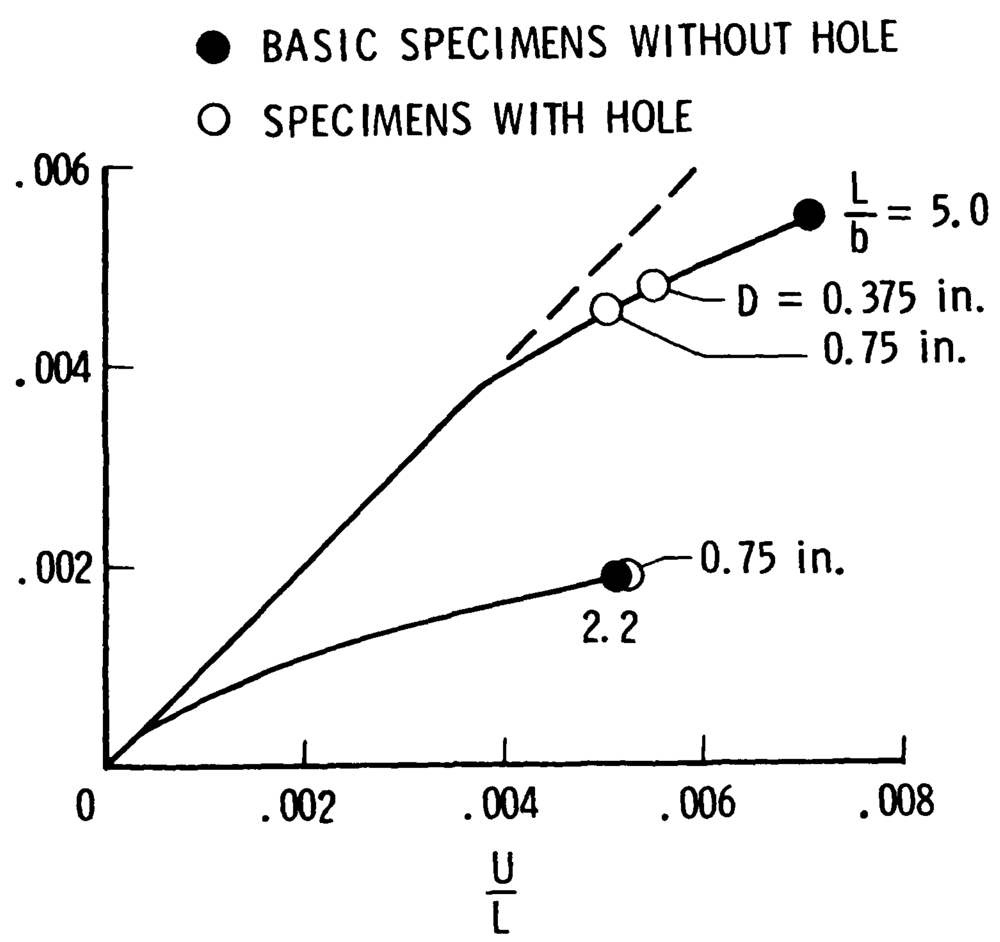
$$\frac{A_{22}D_{11}}{A_{11}D_{22}} \quad \text{AND} \quad \frac{A_{11}A_{22} - A_{12}^2 - 2A_{12}A_{66}}{2A_{66}\sqrt{A_{11}A_{22}}}$$

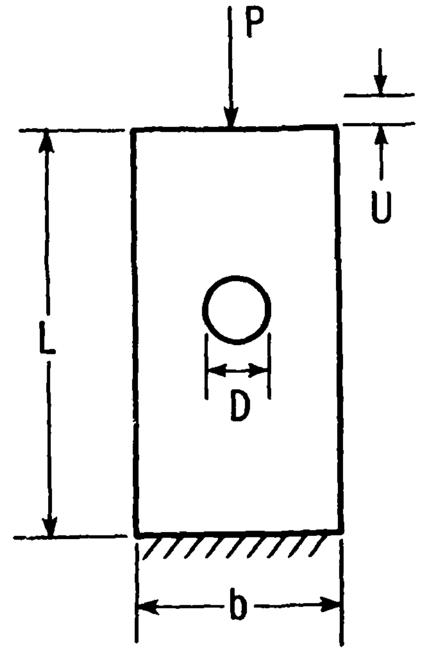
LOAD SHORTENING CURVES

$$\frac{D_{12} + 2D_{66}}{\sqrt{D_{11}D_{22}}} = 1$$

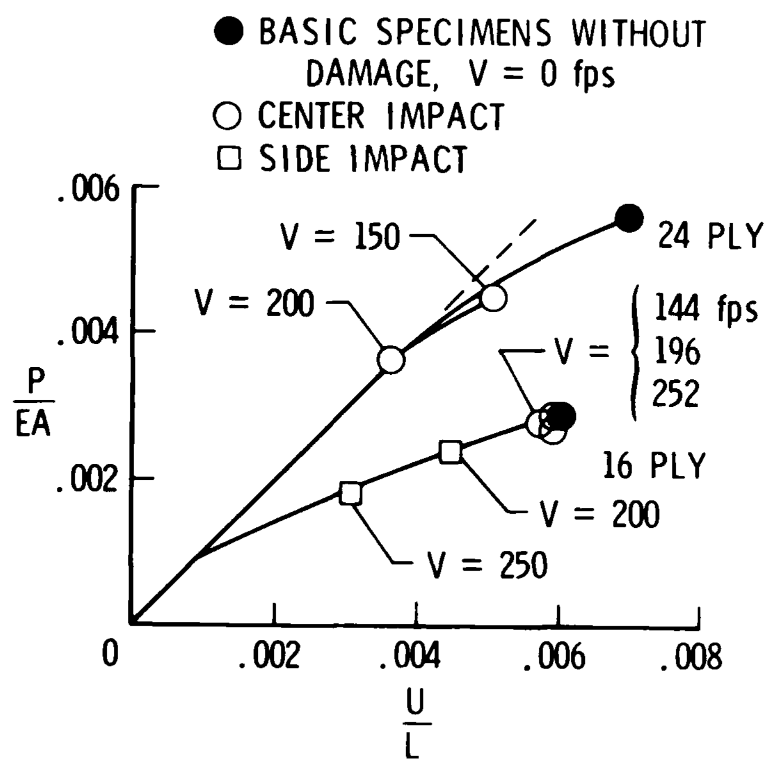


POSTBUCKLING RESPONSE OF SPECIMENS WITH HOLES

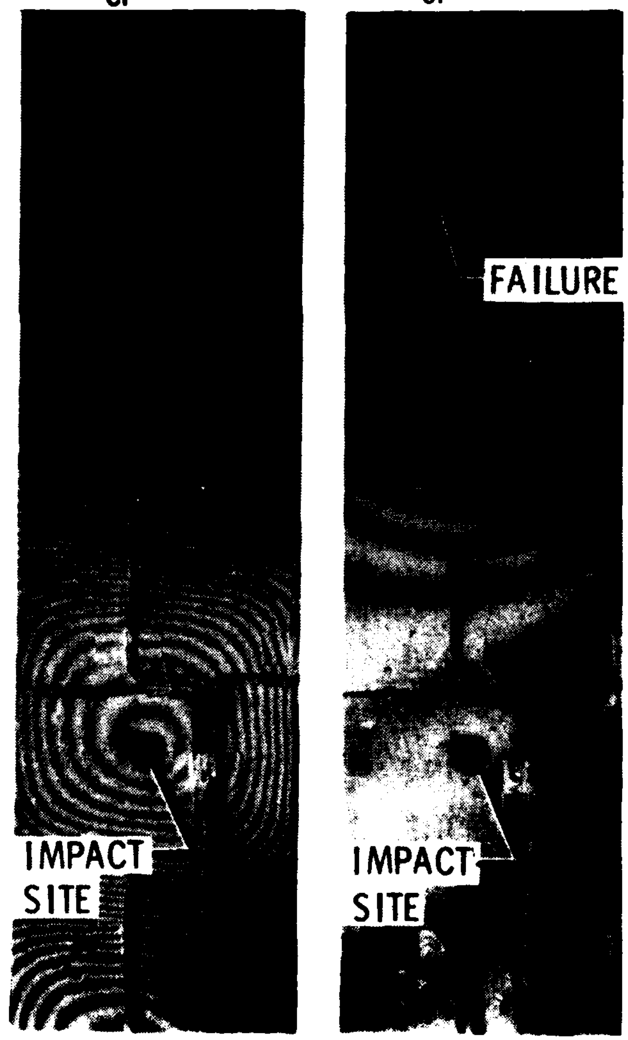




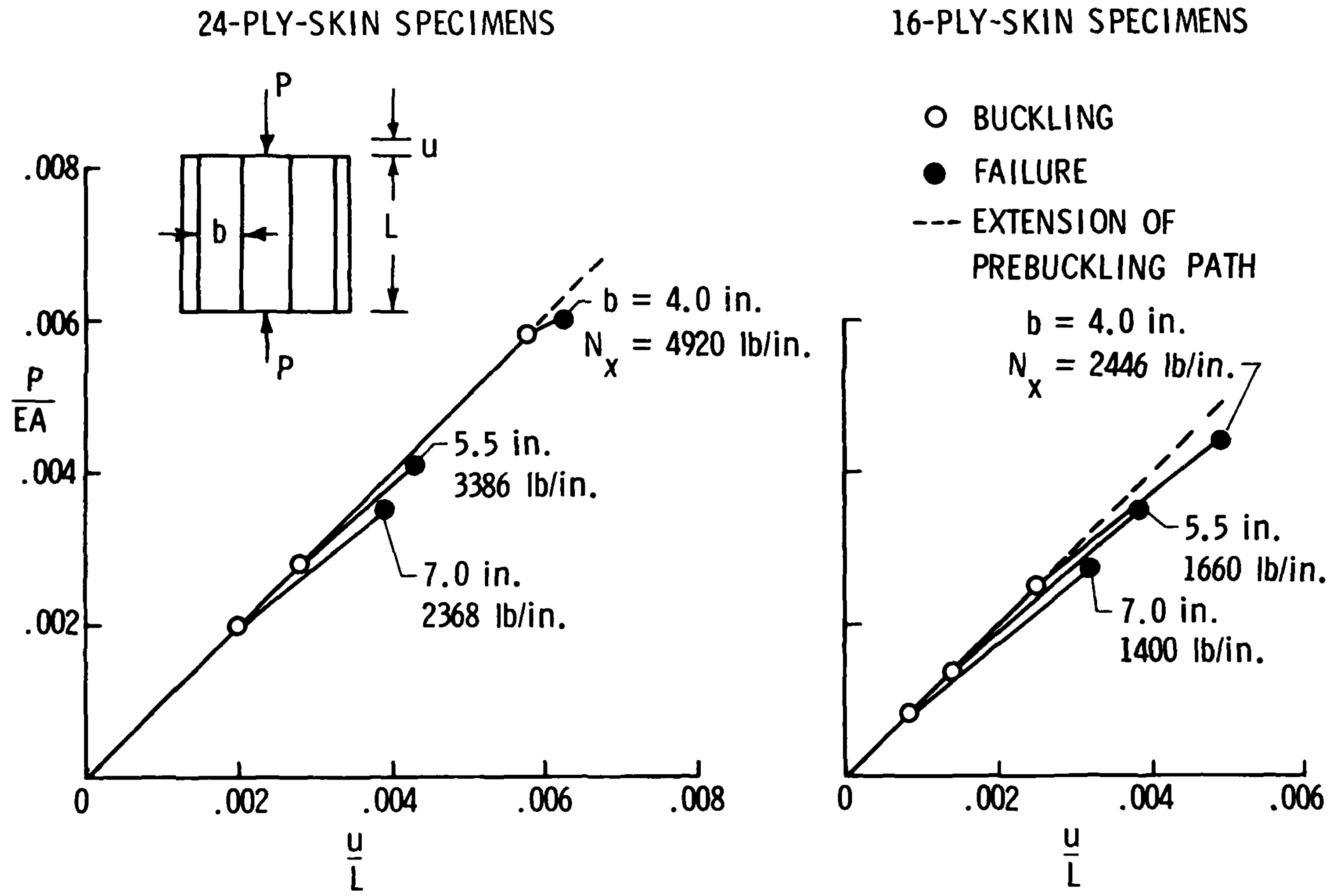
EFFECT OF LOW-SPEED IMPACT DAMAGE



$\frac{P}{P_{cr}} = 3.15$ $\frac{P}{P_{cr}} = 3.26$



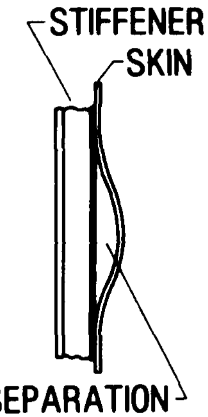
EFFECT OF STIFFENER SPACING ON POSTBUCKLING RESPONSE



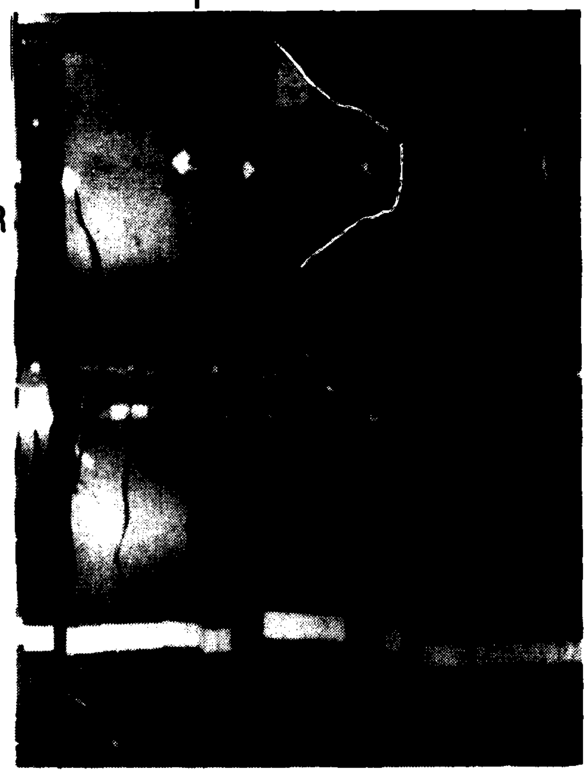
FAILURE MODE OF STIFFENED GRAPHITE-EPOXY PANELS

16-PLY SKIN AND 7.0-inch STIFFENER SPACING

FRONT VIEW
$P = 99\,500$ lb

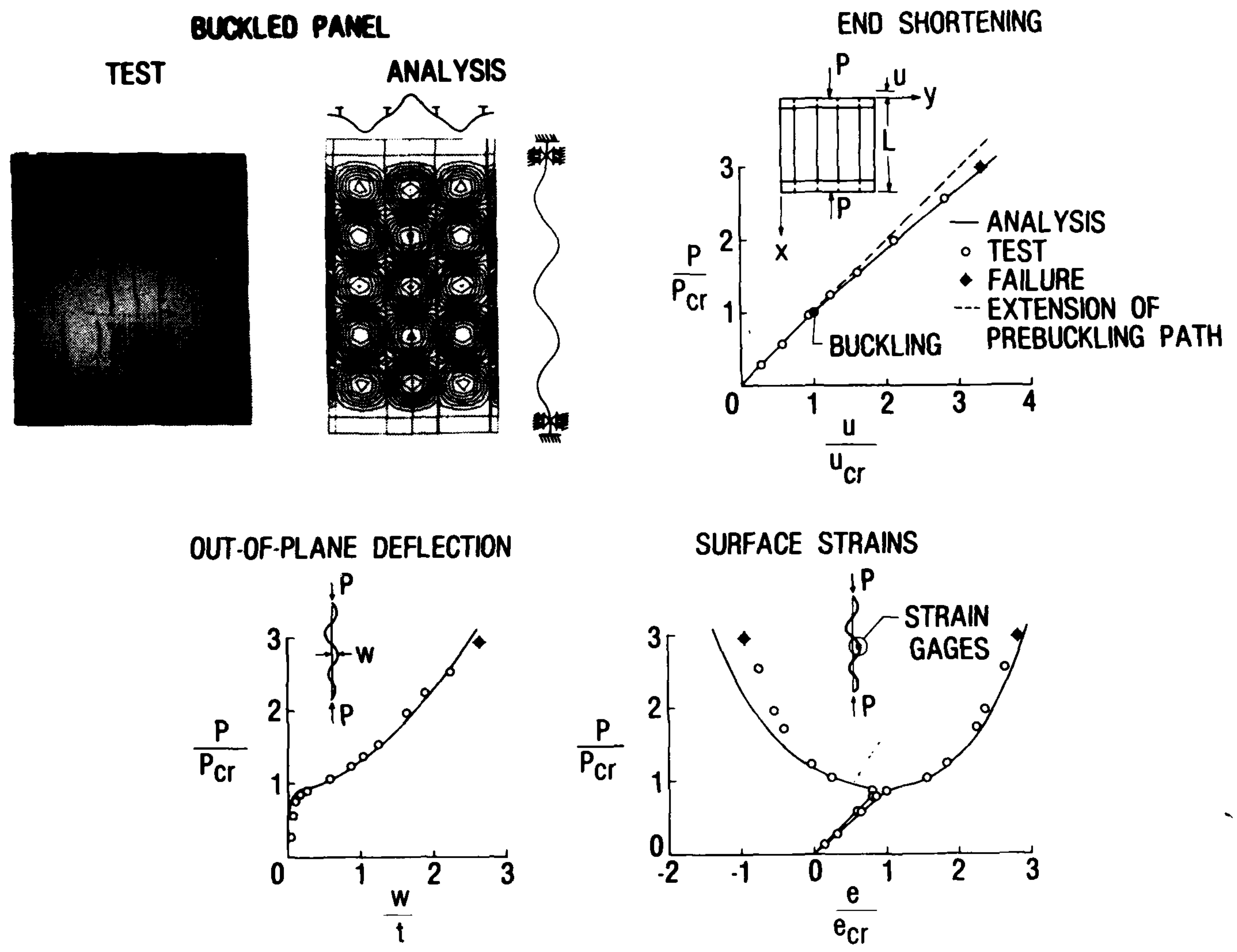


REAR VIEW OF FAILED PANEL
$P_f = 103\,600$ lb



POSTBUCKLING OF STIFFENED GRAPHITE-EPOXY PANEL

16-PLY SKIN AND 7.0-inch STIFFENER SPACING



SPECTRUM FATIGUE BEHAVIOR OF POSTBUCKLED SHEAR PANELS

BY

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HAWTHORNE, CA 90250

* WORK REPORTED HEREIN WAS SPONSORED BY THE NAVAL AIR DEVELOPMENT CENTER UNDER CONTRACT N62269-81-C-0321

OBJECTIVES:

- EVALUATE THE EFFECT OF SPECTRUM FATIGUE ON POSTBUCKLING STRENGTH
- COMPARE THE BEHAVIOR TO CONSTANT AMPLITUDE FATIGUE TESTS

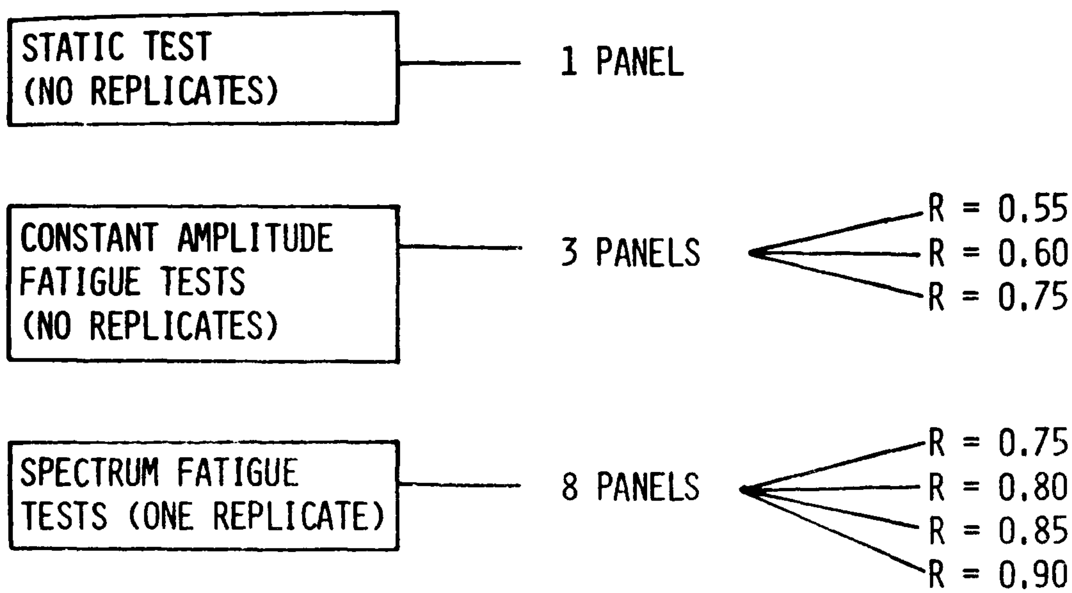
SCOPE:

- GENERATE SUFFICIENT DATA TO DEFINE FATIGUE LIFE (S-N) CURVE FOR PANELS SUBJECTED TO SPECTRUM FATIGUE

- SUPPLEMENT EXISTING CONSTANT AMPLITUDE FATIGUE DATA TO DEFINE S-N CURVE FOR PANELS SUBJECTED TO CONSTANT AMPLITUDE FATIGUE

- DEVELOP CUMULATIVE DAMAGE THEORY TO CORRELATE CONSTANT AMPLITUDE AND SPECTRUM FATIGUE DATA

TEST MATRIX

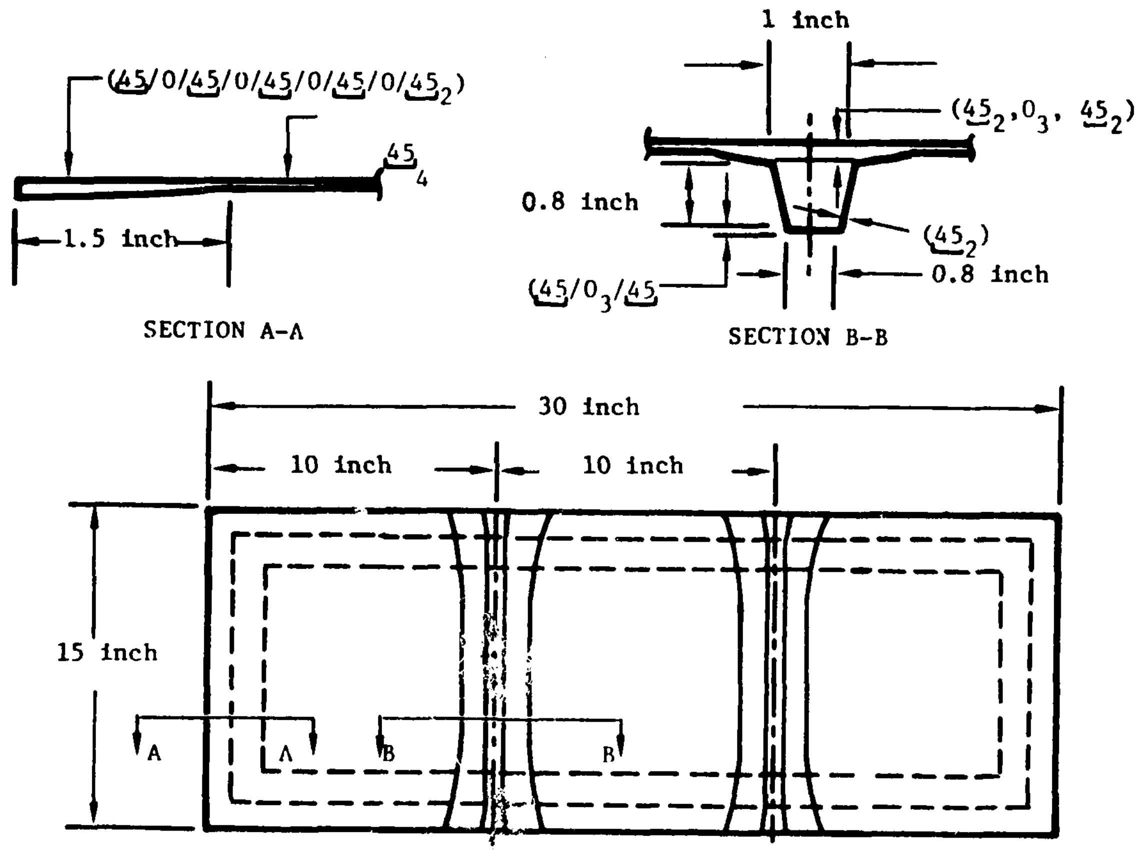


- TOTAL TESTS = 12
- ALL TESTS AT ROOM TEMPERATURE DRY CONDITIONS
- R = RATIO OF MAXIMUM LOAD DURING FATIGUE TO PANEL STATIC ULTIMATE LOAD

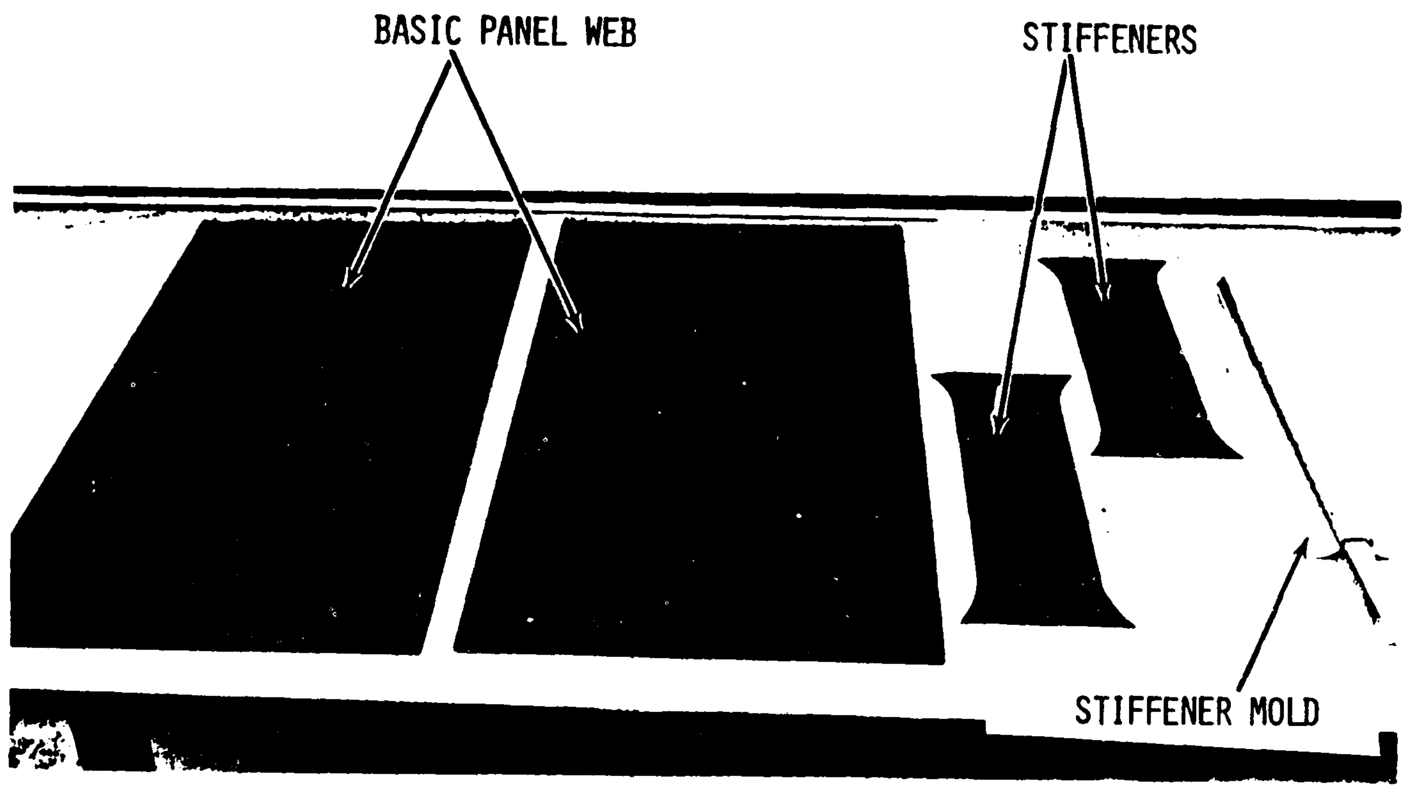
TEST SPECIMEN DESIGN

- DESIGN CRITERIA
  - DESIGN LIMIT LOAD = 400 LBS/INCH
  - PANELS TO BUCKLE AT 30% OF THE DESIGN LIMIT LOAD
  - DESIGN ULTIMATE LOAD = 600 LBS/INCH
- DESIGN CONSIDERATIONS
  - USE PRODUCTION GRAPHITE/EPOXY BROADGOODS
  - USE PRODUCTION FABRICATION TECHNIQUES

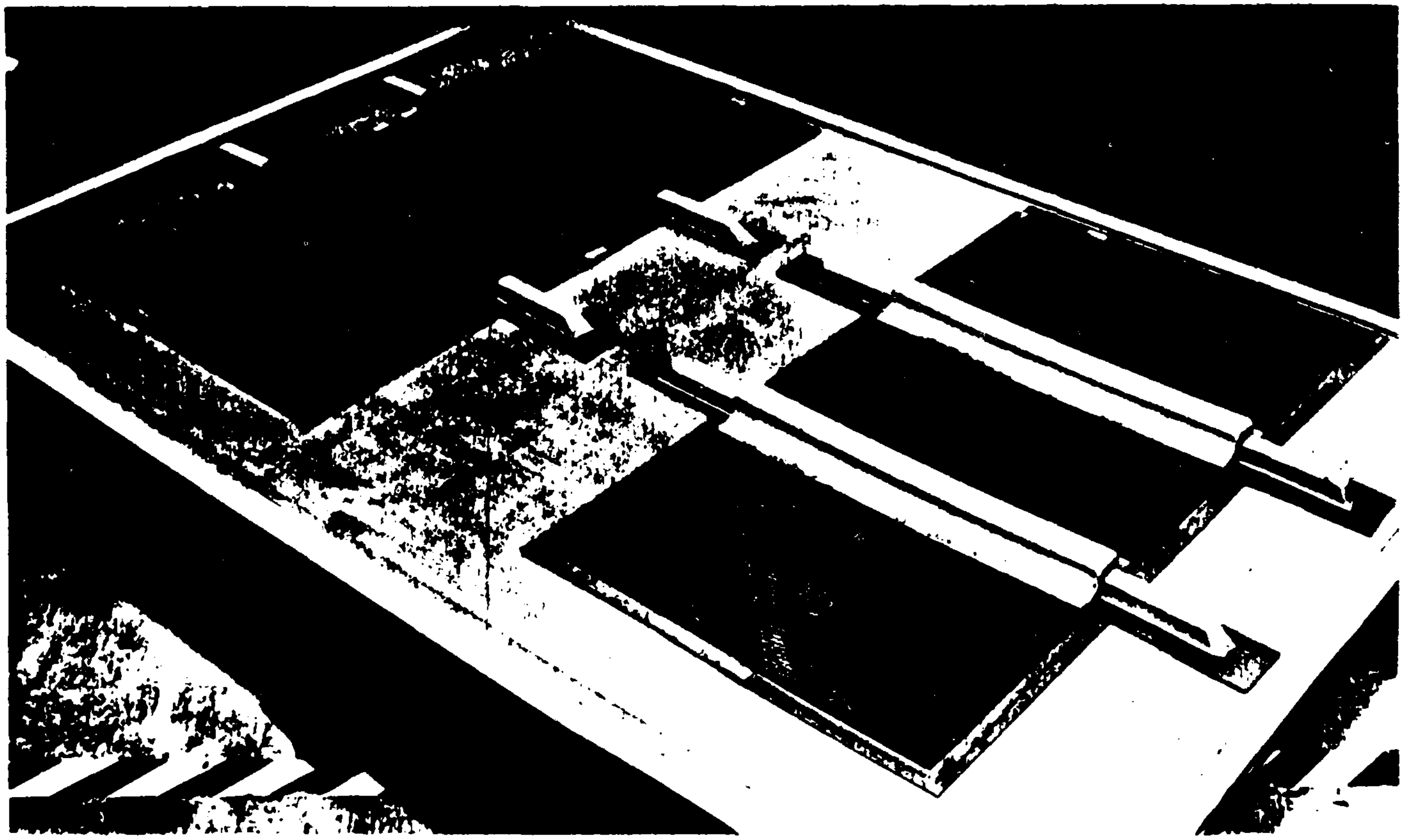
TEST SPECIMEN DETAIL



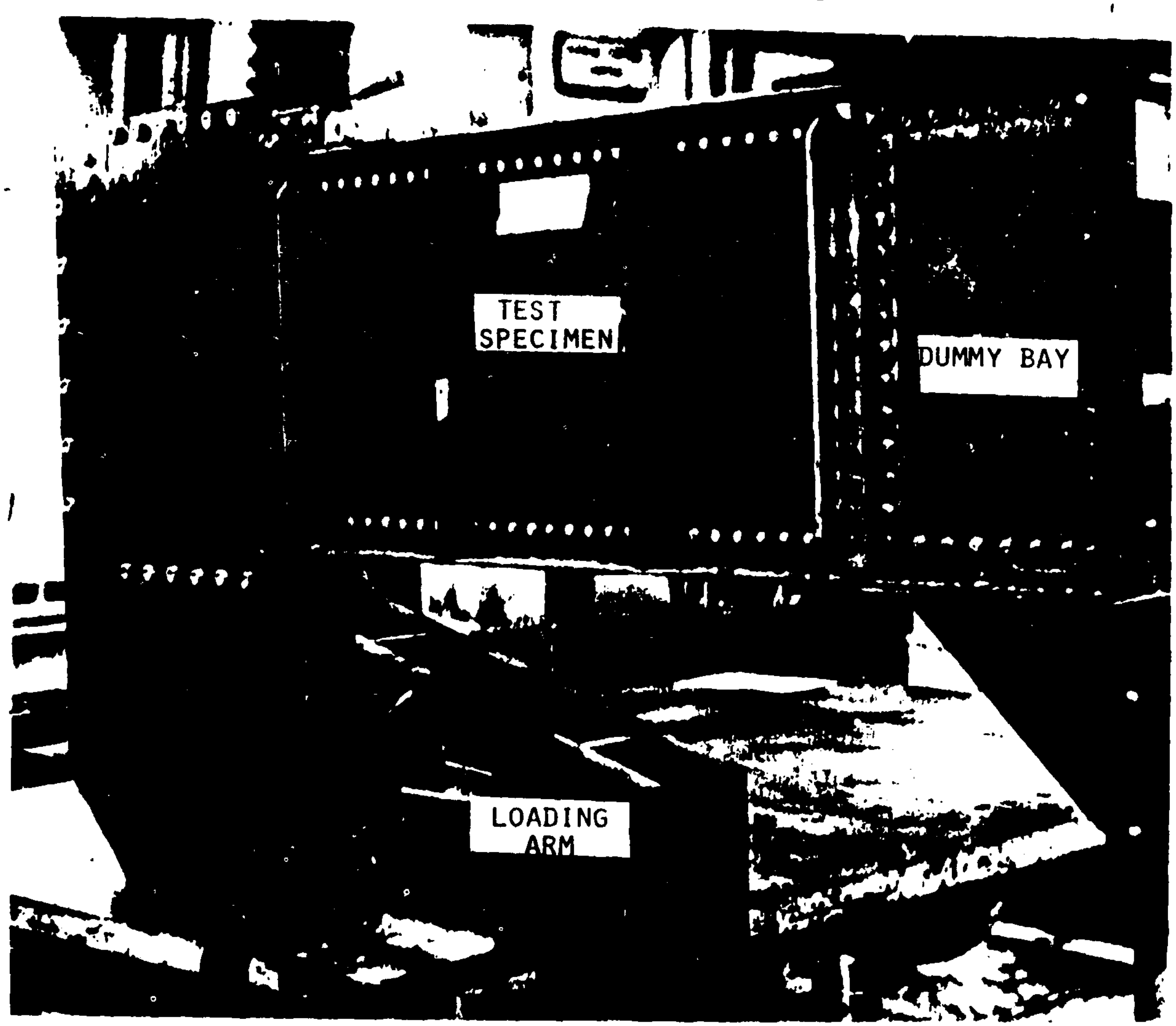
PANEL COMPONENTS



PANEL READY TO BE VACUUM BAGGED



TEST PANEL AND TEST SETUP



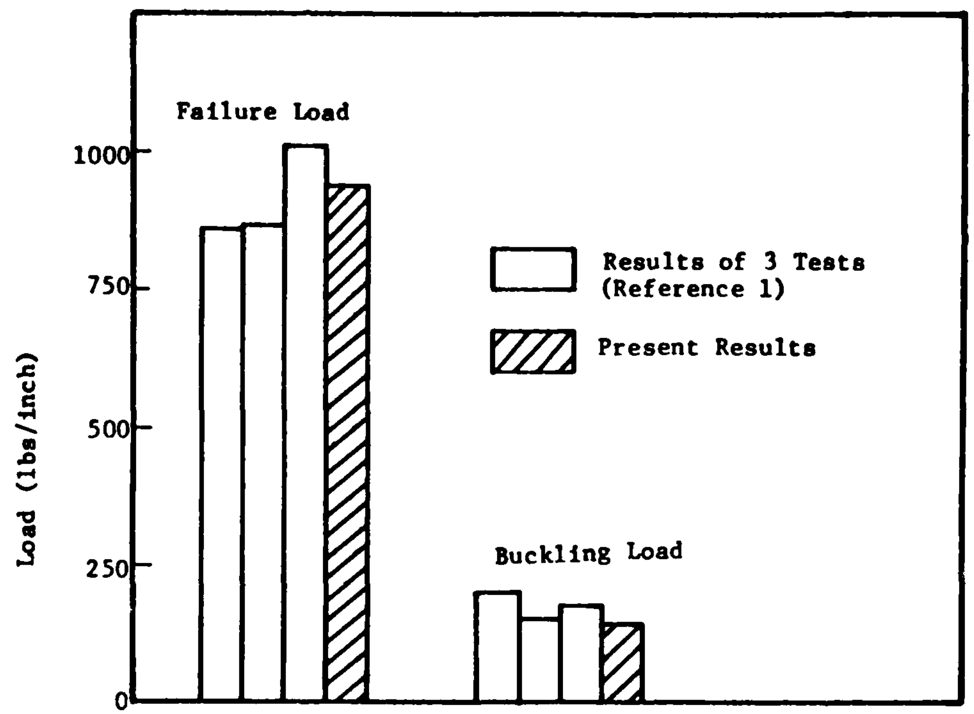
STATIC TEST RESULT

- SIGNIFICANT POSTBUCKLING STRENGTH
- FAILURE LOAD 55% HIGHER THAN DESIGN ULTIMATE LOAD
- INITIAL BUCKLING LOAD IS IN AGREEMENT WITH PREDICTED VALUE
- PANEL FAILURE DUE TO STIFFENER/WEB SEPARATION

A TYPICAL FAILURE MODE

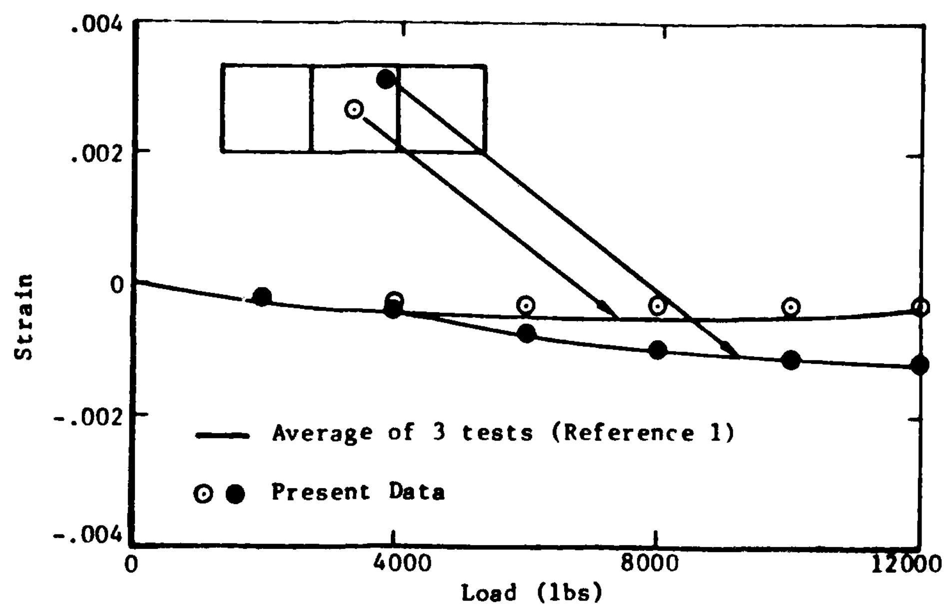
COMPARISON OF FAILURE LOADS

AND BUCKLING LOADS

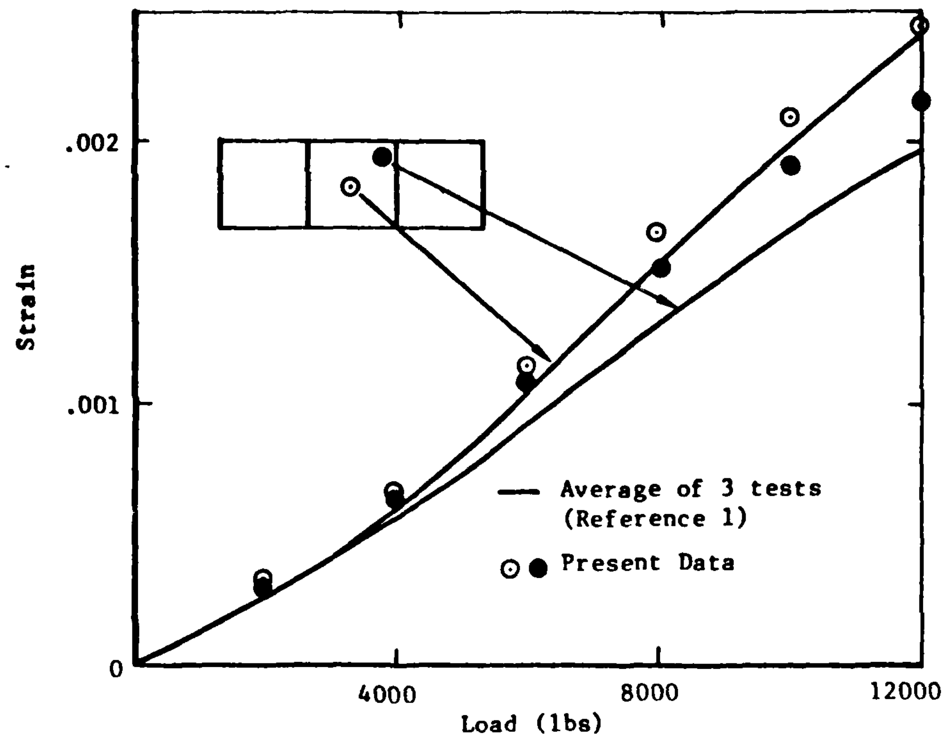


COMPARISON OF STRAINS NORMAL TO THE

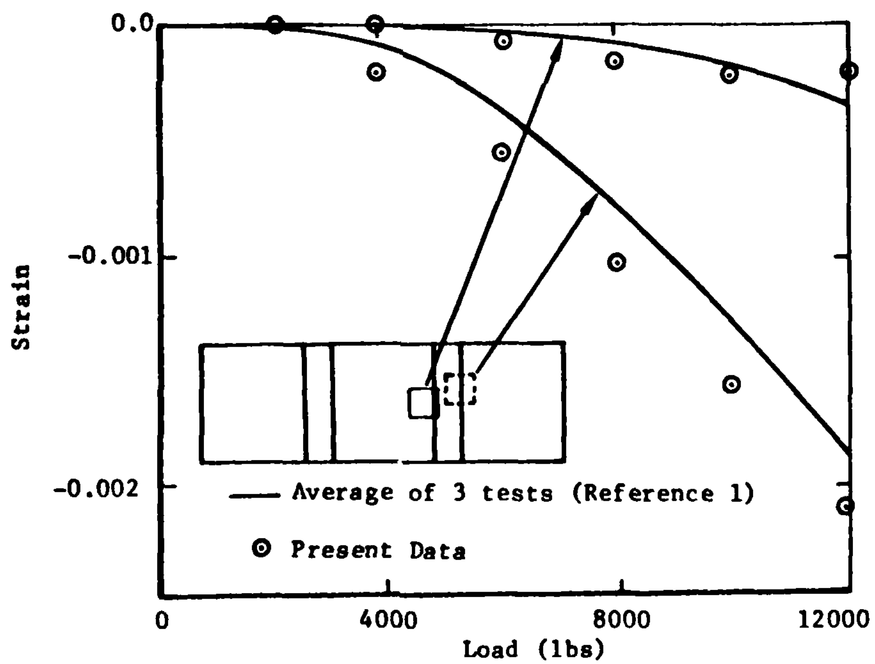
TENSION FIELD DIRECTION



COMPARISON OF STRAINS IN THE DIRECTION OF TENSION FIELDS



COMPARISON OF STRAINS IN THE STIFFENERS AT MID HEIGHT OF THE BEAM

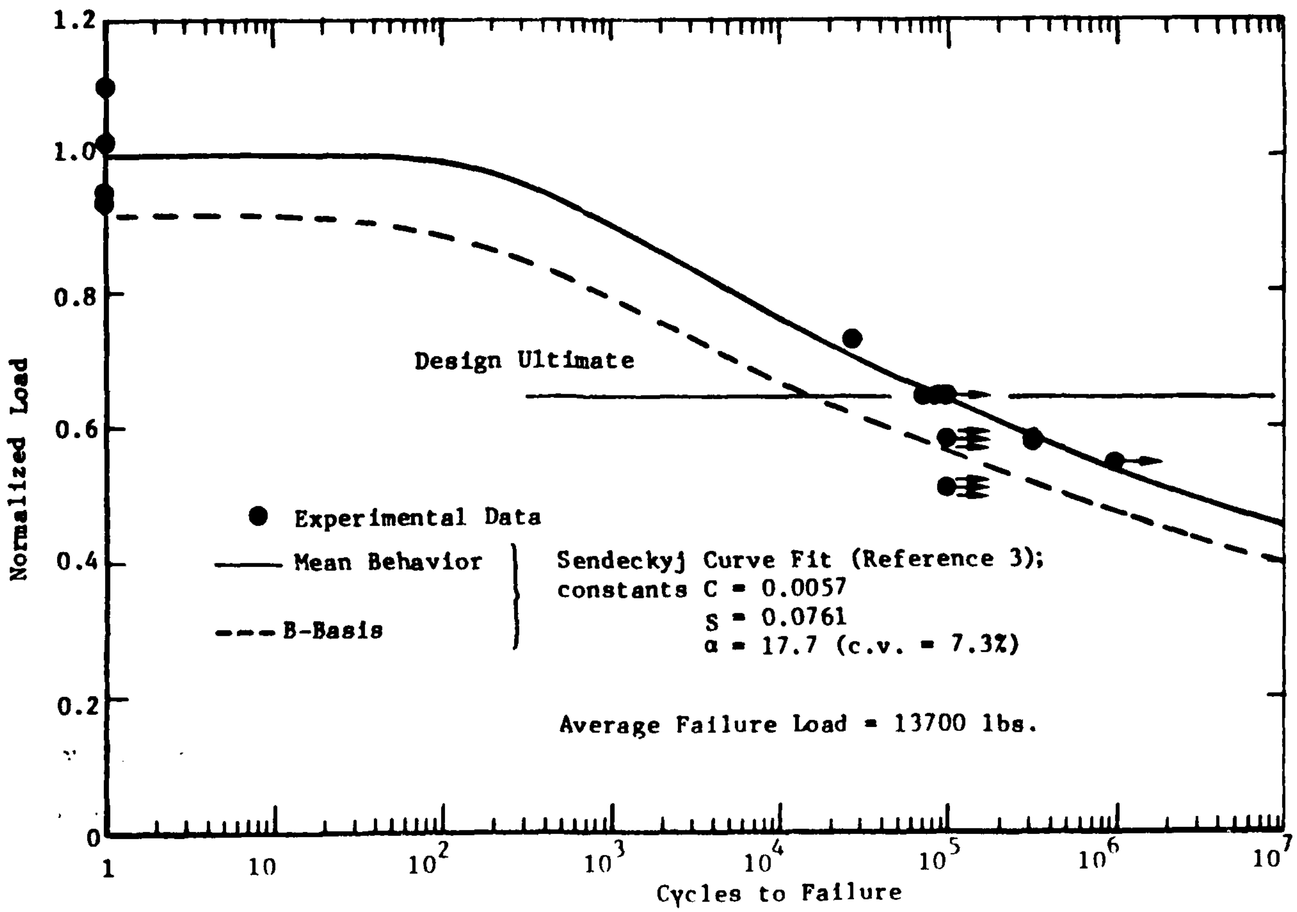


CONSTANT AMPLITUDE FATIGUE RESULTS

- PANELS SHOW NO REDUCTION IN RESIDUAL STRENGTH DUE TO FATIGUE
- FATIGUE FAILURES OCCUR DUE TO STIFFENER WEB SEPARATION AT RELATIVELY HIGH LOAD AMPLITUDES
- TEST DATA GENERATED IN THIS PROGRAM IN CONJUNCTION WITH PREVIOUS DATA ADEQUATELY DEFINE FATIGUE LIFE (S-N) CURVE

FATIGUE LIFE CURVE FOR PANELS

SUBJECTED TO FULLY REVERSED CONSTANT AMPLITUDE LOADING

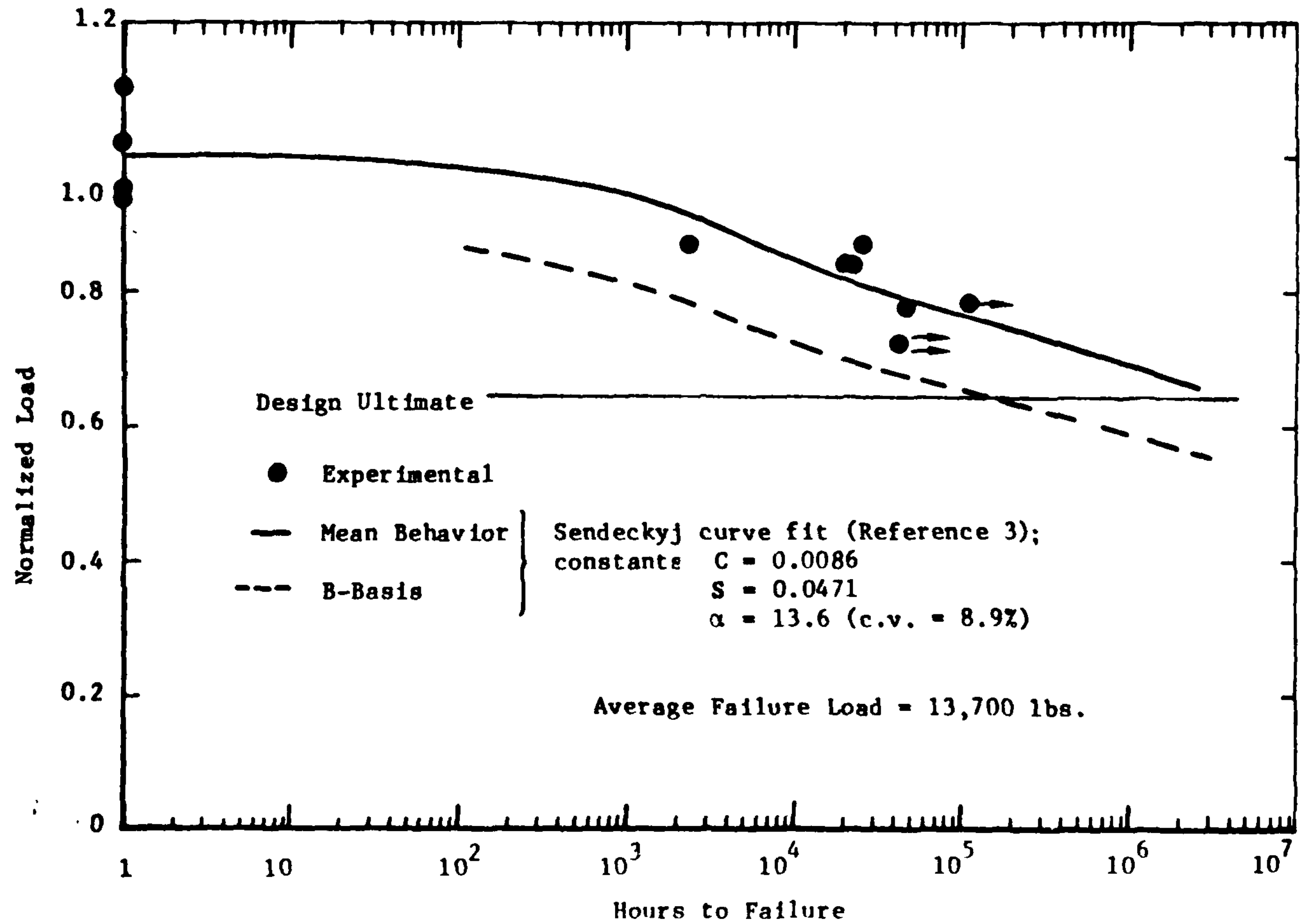


SPECTRUM FATIGUE TEST RESULTS

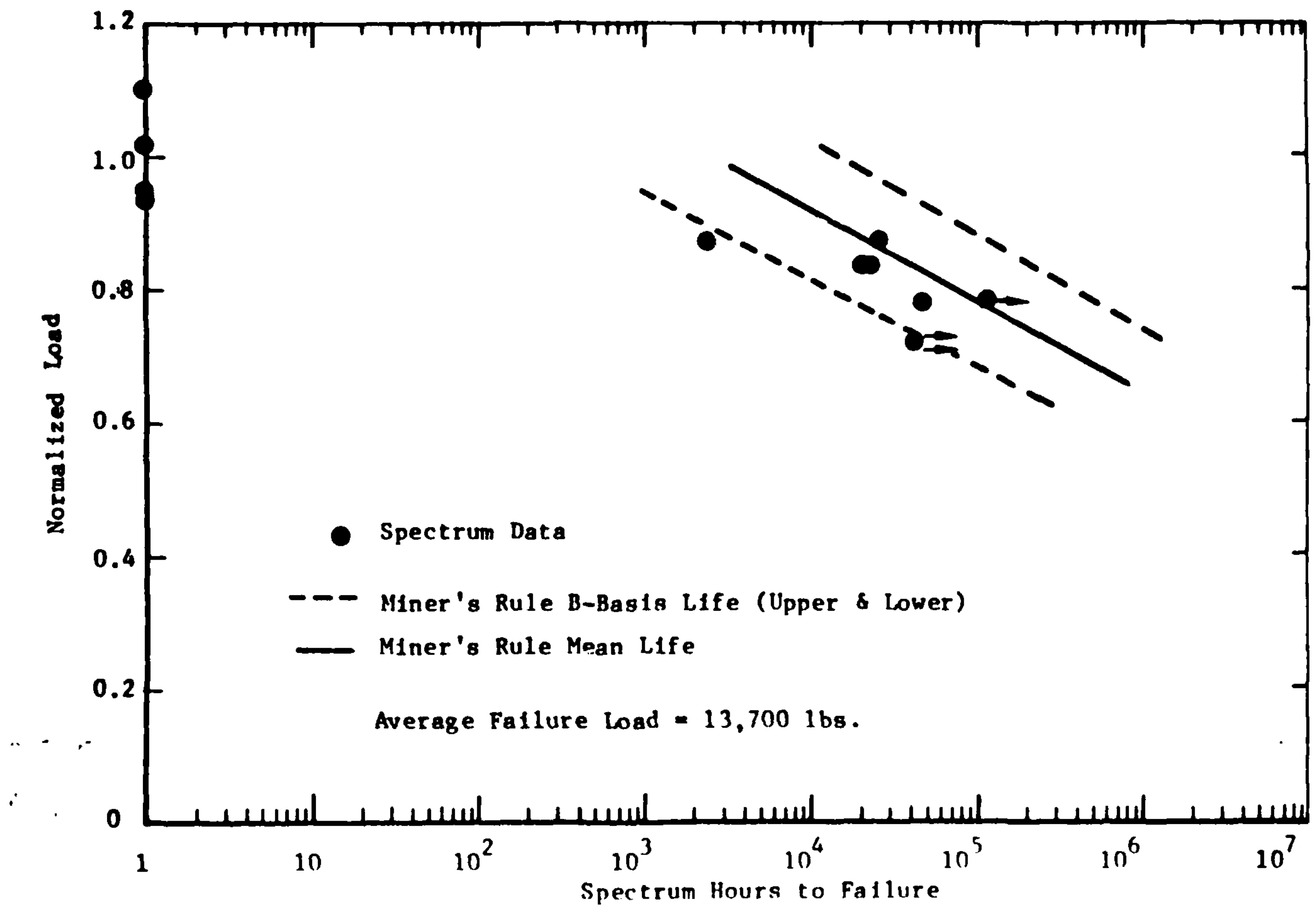
- PANEL SUBJECTED TO MAXIMUM LOAD LEVELS UPTO 80% OF ULTIMATE FAILURE LOAD SURVIVED TWO LIFETIMES OF FATIGUE LOADING WITH NO REDUCTION IN RESIDUAL STRENGTH
- FATIGUE FAILURES ARE IDENTICAL TO THOSE OF CONSTANT AMPLITUDE PANELS
- TEST DATA ADEQUATELY DEFINES THE FATIGUE LIFE CURVE

FATIGUE LIFE CURVE DUE TO

SPECTRUM LOADING



COMPARISON OF EXPERIMENTAL AND ANALYTICAL RESULTS



CONCLUSIONS

- POSTBUCKLED SHEAR PANELS HAVE EXCEPTIONALLY GOOD FATIGUE BEHAVIOR
- PANELS NEED ONLY BE DESIGNED TO SATISFY STATIC STRENGTH REQUIREMENTS.
- CONSTANT AMPLITUDE DATA IS USEFUL IN ESTIMATING FATIGUE LIFE UNDER SPECTRUM LOADING

DEVELOPMENT OF ANALYSIS FOR PREDICTING COMPRESSION FATIGUE LIFE AND RESIDUAL STRENGTH IN COMPOSITES

NADC CONTRACT - N62269-80-C-0265

CONTRACT MONITOR - MR. L. GAUSE

PRINCIPAL INVESTIGATORS - M. M. RATWANI AND H. P. KAN

NORTHROP CORPORATION
HAWTHORNE, CALIFORNIA

OBJECTIVE

DEVELOP MODEL FOR COMPRESSION FATIGUE AND RESIDUAL STRENGTH OF COMPOSITES,

INCORPORATING EFFECTS OF:

- o PLY ORIENTATIONS
- o STACKING SEQUENCE
- o LOADING CONDITIONS
  (CONSTANT AMPLITUDE, SPECTRUM)

CONCLUSIONS

1. THE COMPRESSION FATIGUE LIFE OF A COMPOSITE LAMINATE IS RELATED TO INTERLAMINAR STRESSES PRODUCED IN THE LAMINATE.

2. THE DELAMINATION PROPAGATION MODEL CAN BE USED TO PREDICT COMPRESSION FATIGUE LIFE AND RESIDUAL STRENGTH OF COMPOSITES IN WHICH DELAMINATION IS DOMINANT FAILURE MECHANISM.

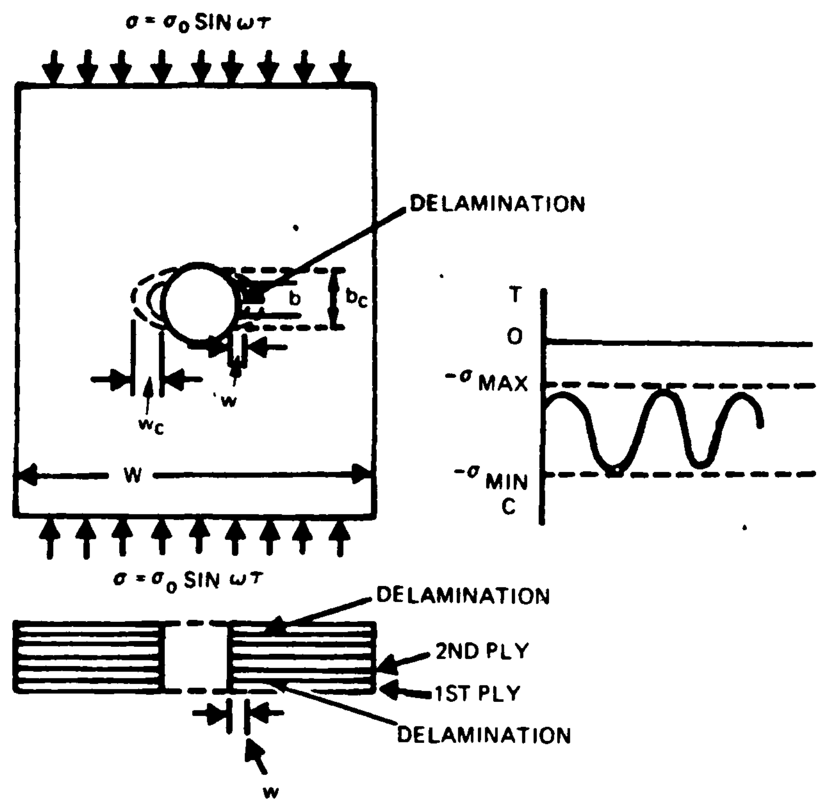
3. INTERLAMINAR NORMAL STRESSES HAVE SIGNIFICANT INFLUENCE ON THE COMPRESSION FATIGUE LIFE.

4. UNDER COMPRESSION DOMINATED SPECTRUM LOADING, THE MAJORITY OF THE LOAD CYCLES PRODUCE INTERLAMINAR STRESS RANGE BELOW THRESHOLD AND THE LAMINATE FATIGUE LIFE IS GENERALLY LONG.

5. DAMAGE GROWTH UNDER SPECTRUM LOADING DIFFERS SIGNIFICANTLY FROM THAT UNDER CONSTANT AMPLITUDE LOADING.

DELAMINATION PROPAGATION MODEL

PROPAGATION UNDER SHEAR STRESSES



COMPOSITE PANEL SUBJECTED TO FATIGUE LOADS

$$\frac{db}{dN} = c_1 \left(\tau_{zma} - \tau_{zmi} - \tau_{th}\right)^{n_1} b^{m_1}$$

where

- b is the length of the delamination,
- N is the number of fatigue cycles,
- τ_{zmi} is the minimum interlaminar shear stress,
- τ_{zma} is the maximum interlaminar shear stress,
- τ_{th} is the interlaminar threshold shear stress range below which delamination will not propagate,
- c_1, n_1 and m_1 are constants which depend on the resin system.

FATIGUE LIFE IS GIVEN BY

$$N_f = \frac{b_c^{1-0.5n_1} - b_o^{1-0.5n_1}}{c_1 \left(\tau_{zma} - \tau_{zmi} - \tau_{th}\right)^{n_1} (1 - 0.5n_1)}$$

DELAMINATION PROPAGATION UNDER SHEAR AND NORMAL STRESSES

$$\frac{db}{dN} = c_1 (\tau_{zma} - \tau_{zmi} - \tau_{th} + f)^{n_1} b^{m_1} + c_2 (\sigma_{zma} - \sigma_{zmi} - \sigma_{zth})^{n_2} b^{m_2}$$

σ_{zma} is the maximum interlaminar normal stress

σ_{zmi} is the minimum interlaminar normal stress

σ_{zth} is the threshold value of interlaminar normal stress range

f is the clamping stress, and is either zero or negative

c_2, n_2, m_2 are constants which depend on resin system and environment.

UNDER COMPRESSION-COMPRESSION LOADING

$f = 0$ if $a_2 = \sigma_z/\sigma_0 < 0$ and $f = \mu a_2 \sigma_{min}$ if $a_2 = \sigma_z/\sigma_0 > 0$

σ_z is the interlaminar normal stress

σ_0 is the remote applied stress

μ is the coefficient of friction between plies

UNDER TENSION-COMPRESSION LOADING THE VALUE OF "f" IS GIVEN BY

$$f = \mu a_2 \sigma_{min} \quad \text{if} \quad a_2 = \sigma_z/\sigma_o > 0$$

$$f = \mu a_2 \sigma_{max} \quad \text{if} \quad a_2 = \sigma_z/\sigma_o < 0$$

FATIGUE LIFE IS GIVEN BY

$$N_f = \frac{\dfrac{1}{b_o^{m-1}} - \dfrac{1}{b_c^{m-1}}}{(m-1)\left[c_1(\tau_{zma} - \tau_{zmi} - \tau_{th} + f)^{n_1} + c_2(\sigma_{zma} - \sigma_{zmi} - \sigma_{zth})^{n_2}\right]}$$

CUMULATIVE DAMAGE IS GIVEN BY

$$\sum \frac{N_i}{N_{fi}} \left[1 - \left(\frac{b_o}{b_{ci}}\right)^{m-1}\right] + \left(\frac{b_o}{b_{cn}}\right)^{m-1} = 1$$

MINER'S RULE IS GIVEN BY

$$\sum \frac{N_i}{N_{fi}} = 1$$

RESIDUAL STRENGTH PREDICTION MODEL

RESIDUAL STRENGTH IS EXPRESSED AS

$$R(N) = R(0)\,\Phi(b_N)$$

WHERE R(N) IS THE RESIDUAL STRENGTH AFTER N CYCLES OF FATIGUE LOADING, R(O) IS THE STATIC STRENGTH, AND $\Phi(b_N)$ IS AN ARBITRARY FUNCTION OF THE DELAMINATION SIZE, b_N, AFTER N NUMBER OF FATIGUE CYCLES.

IT CAN BE SHOWN THAT

$$R(N) = R(0)\left[1 - \left(1 - \frac{\sigma_{min}}{R(0)}\right)\left(\frac{b_N - b_o}{b_c - b_o}\right)\right]$$

UNDER SPECTRUM LOADING b_N IS GIVEN BY

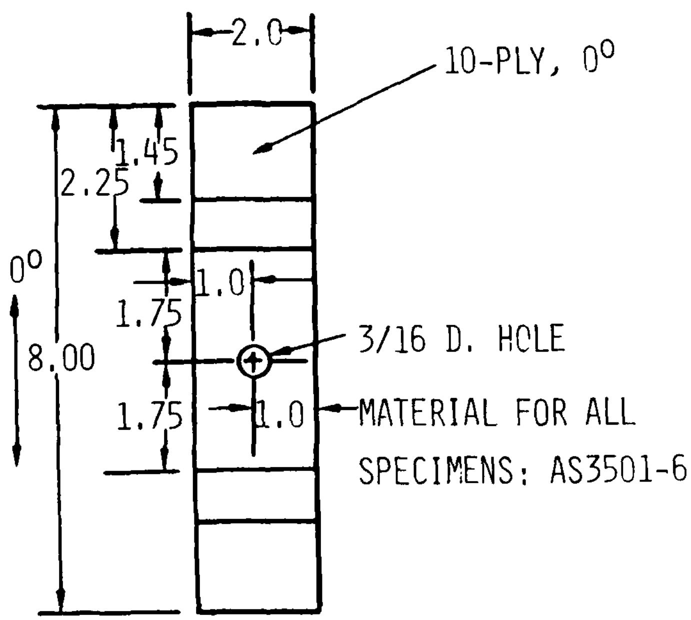
$$b_N = \frac{b_o}{\left[\left(1 - \sum \frac{N_i}{N_{fi}}\right) + \sum \frac{N_i}{N_{fi}}\left(\frac{b_o}{b_{ci}}\right)^{m_i - 1}\right]^{(1/m_i - 1)}}$$

EXPERIMENTAL PROGRAM

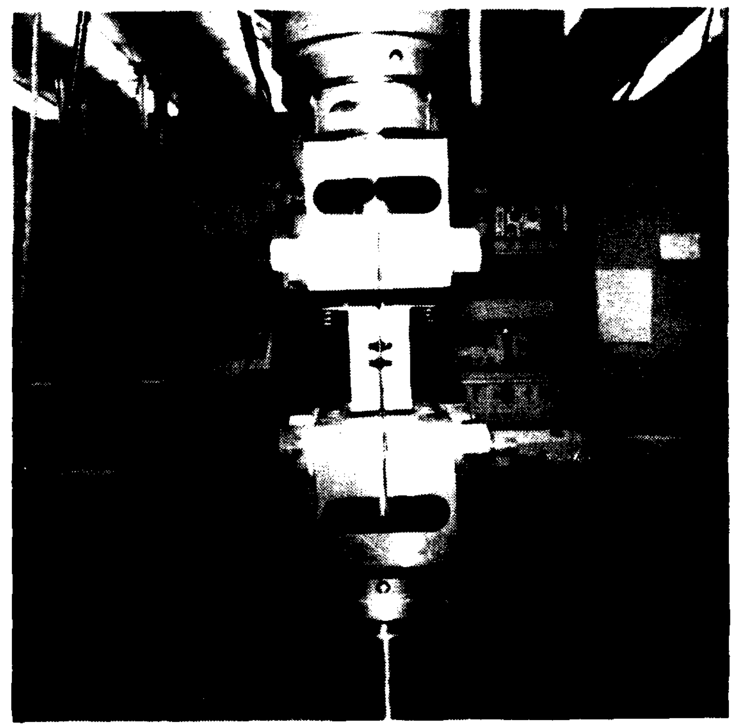
TWO LAMINATES - 10 PLY $(0/90/\pm45/0)_S$

16 PLY $(\pm45/0_4/90/0)_S$

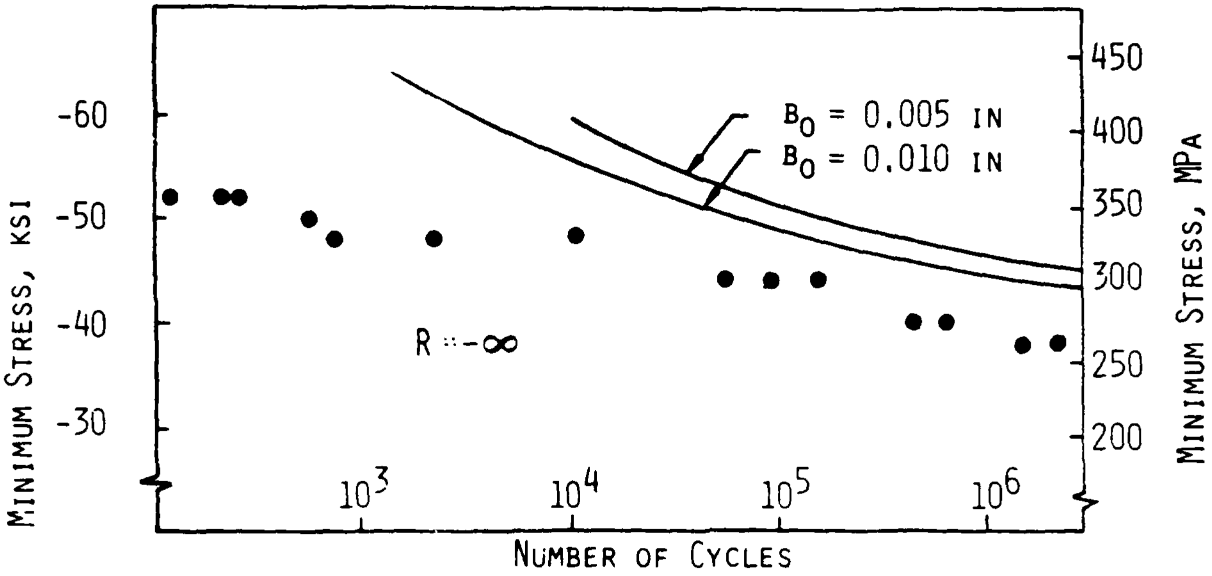
FATIGUE TESTING AT R = -1 AND $-\infty$ AND UNDER FLIGHT-BY-FLIGHT COMPRESSION-DOMINATED SPECTRUM.



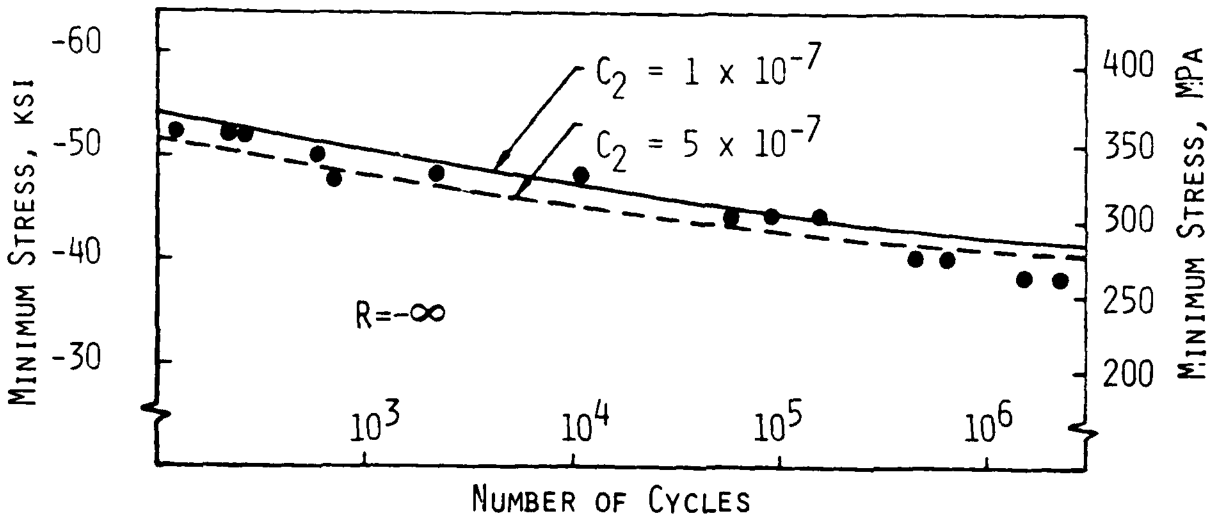
TEST SPECIMEN



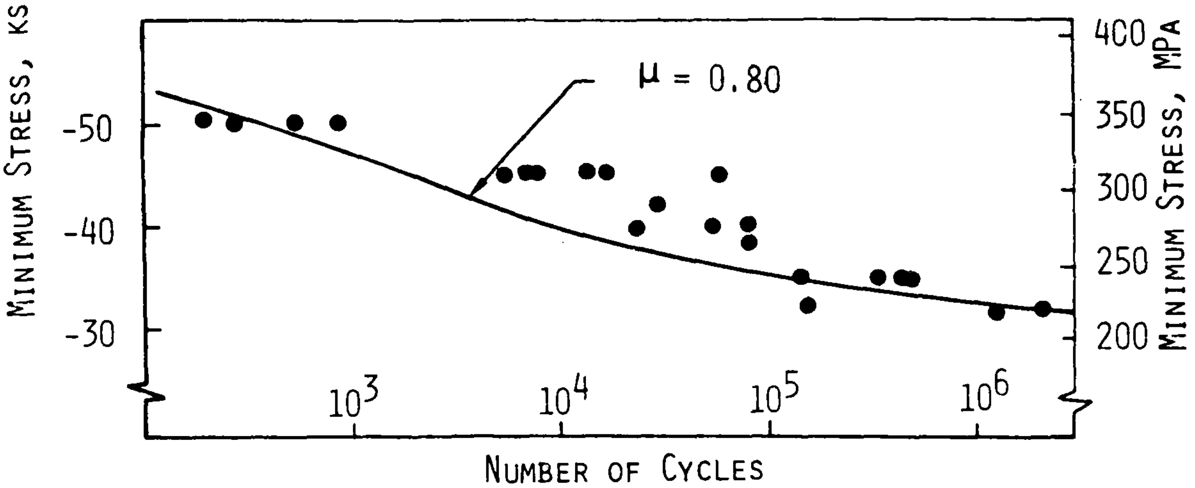
SPECIMEN IN TEST MACHINE



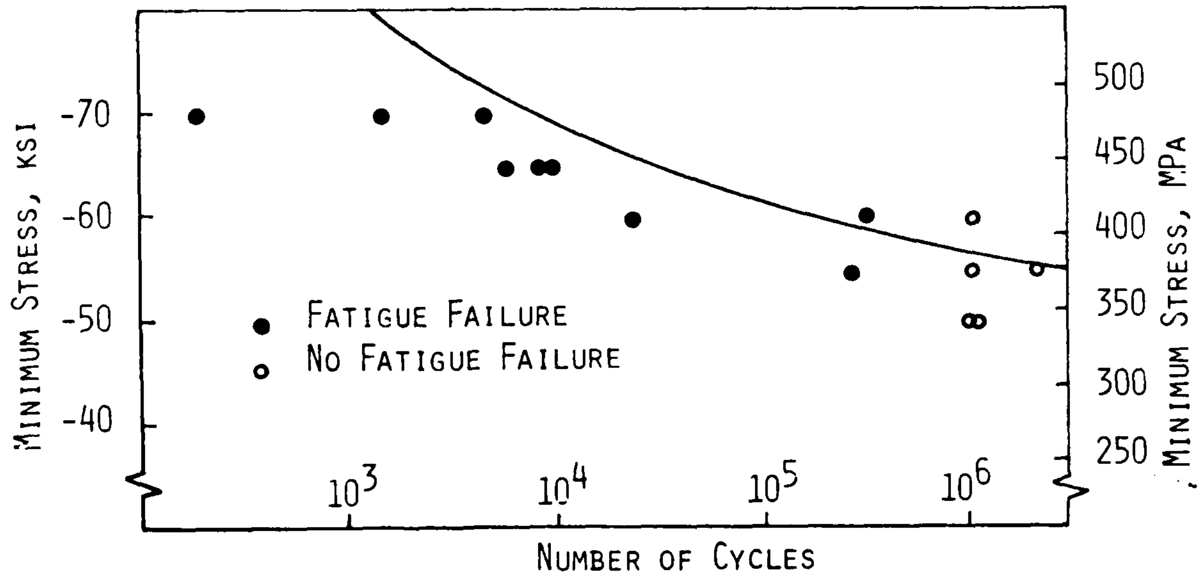
COMPARISON OF OBSERVED AND PREDICTED FATIGUE BEHAVIOR OF THE 10-PLY LAMINATE. PREDICTION BASED ON SHEAR STRESSES ONLY.



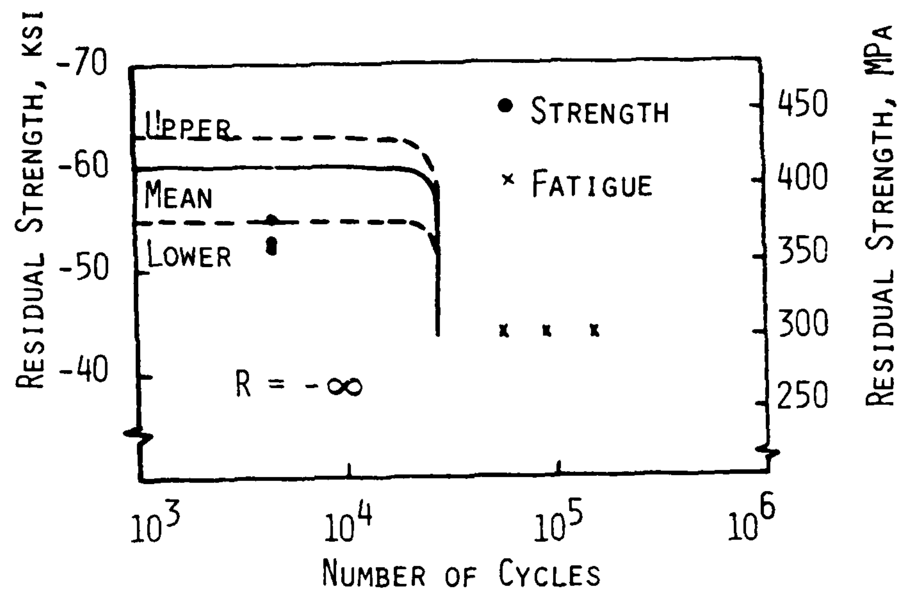
COMPARISON OF OBSERVED AND PREDICTED FATIGUE BEHAVIOR OF THE 10-PLY LAMINATE. PREDICTION WITH THE PRESENT MODEL.



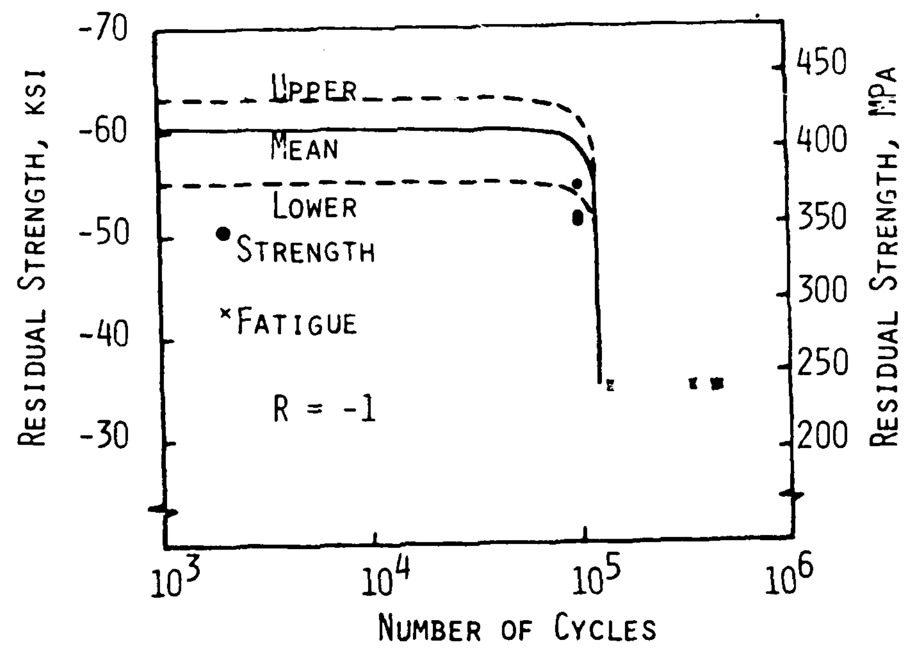
COMPARISON OF OBSERVED AND PREDICTED FATIGUE BEHAVIOR OF THE 10-PLY LAMINATES AT R = -1.



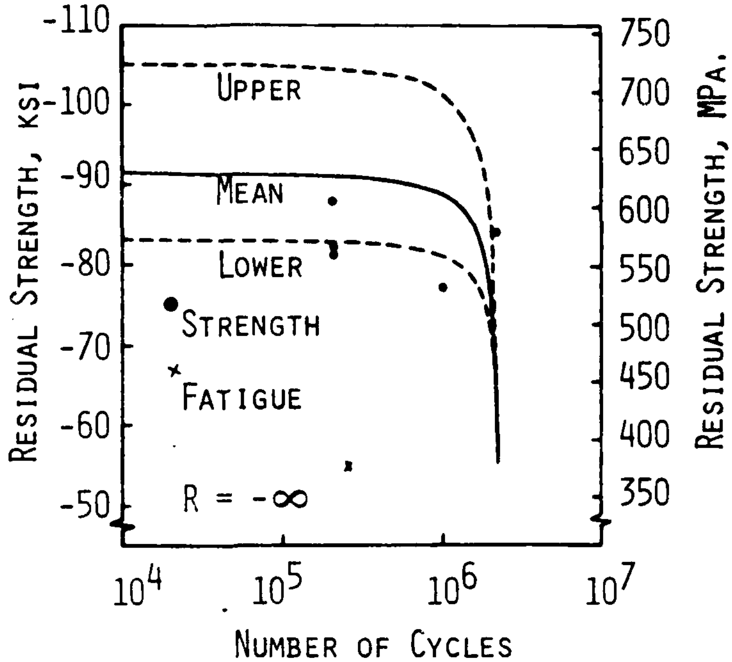
COMPARISON OF OBSERVED AND PREDICTED FATIGUE BEHAVIOR OF THE 16-PLY LAMINATE AT R = $-\infty$.



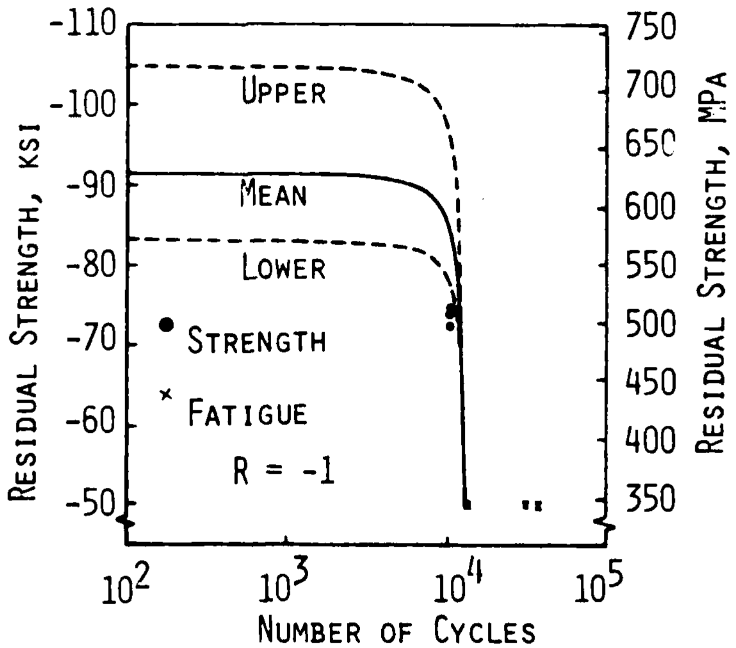
Comparison of Cbserved and Predicted Residual Strength of the 10-Ply Laminate Tested at σ_{MIN} = -44 ksi.



Comparison of Observed and Predicted Residual Strength of the 10-Ply Laminate Tested at σ_{MIN} = -35 ksi.



Comparison of Observed and Predicted Residual Strength of the 16-Ply Laminate Tested at σ_{MIN} = -55 ksi.



Comparison of Observed and Predicted Residual Strength of the 16-Ply Laminate Tested at σ_{MIN} = -50 ksi,

MICROBUCKLING INITIATED FAILURE IN TOUGH RESIN LAMINATES

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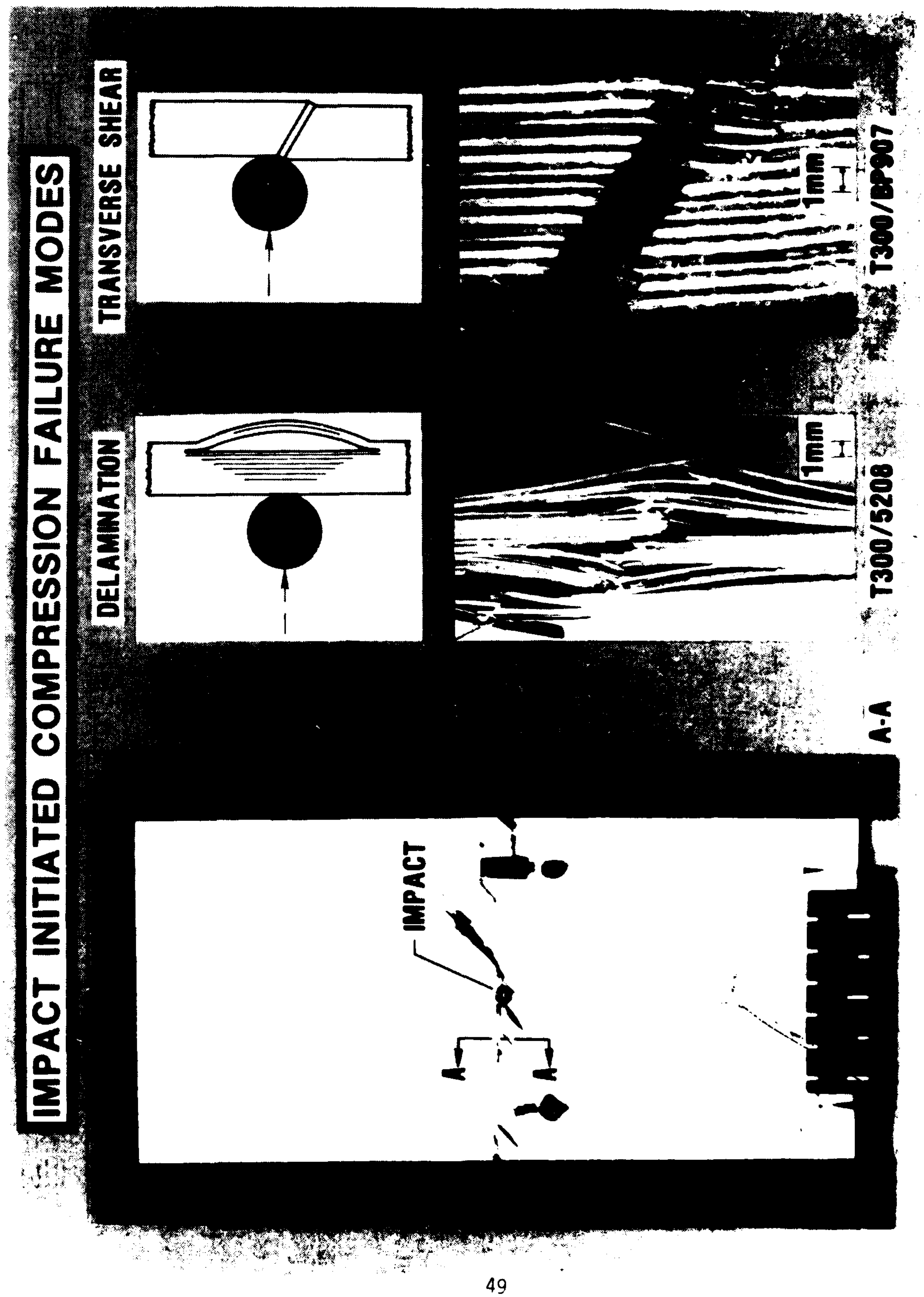
NASA LANGLEY RESEARCH CENTER

RESEARCH OBJECTIVES

- TO IDENTIFY FAILURE MECHANISMS WHICH INITIATE FAILURE IN COMPRESSION LOADED GRAPHITE-EPOXY LAMINATES AND STRUCTURAL COMPONENTS
- TO UNDERSTAND WHY TOUGH RESINS IMPROVE THE COMPRESSIVE STRENGTH OF GRAPHITE-EPOXY LAMINATES WITH IMPACT DAMAGE BUT NOT WITH HOLES,
- TO DEVELOP WAYS TO MAKE COMPRESSION LOADED LAMINATES LESS NOTCH SENSITIVE

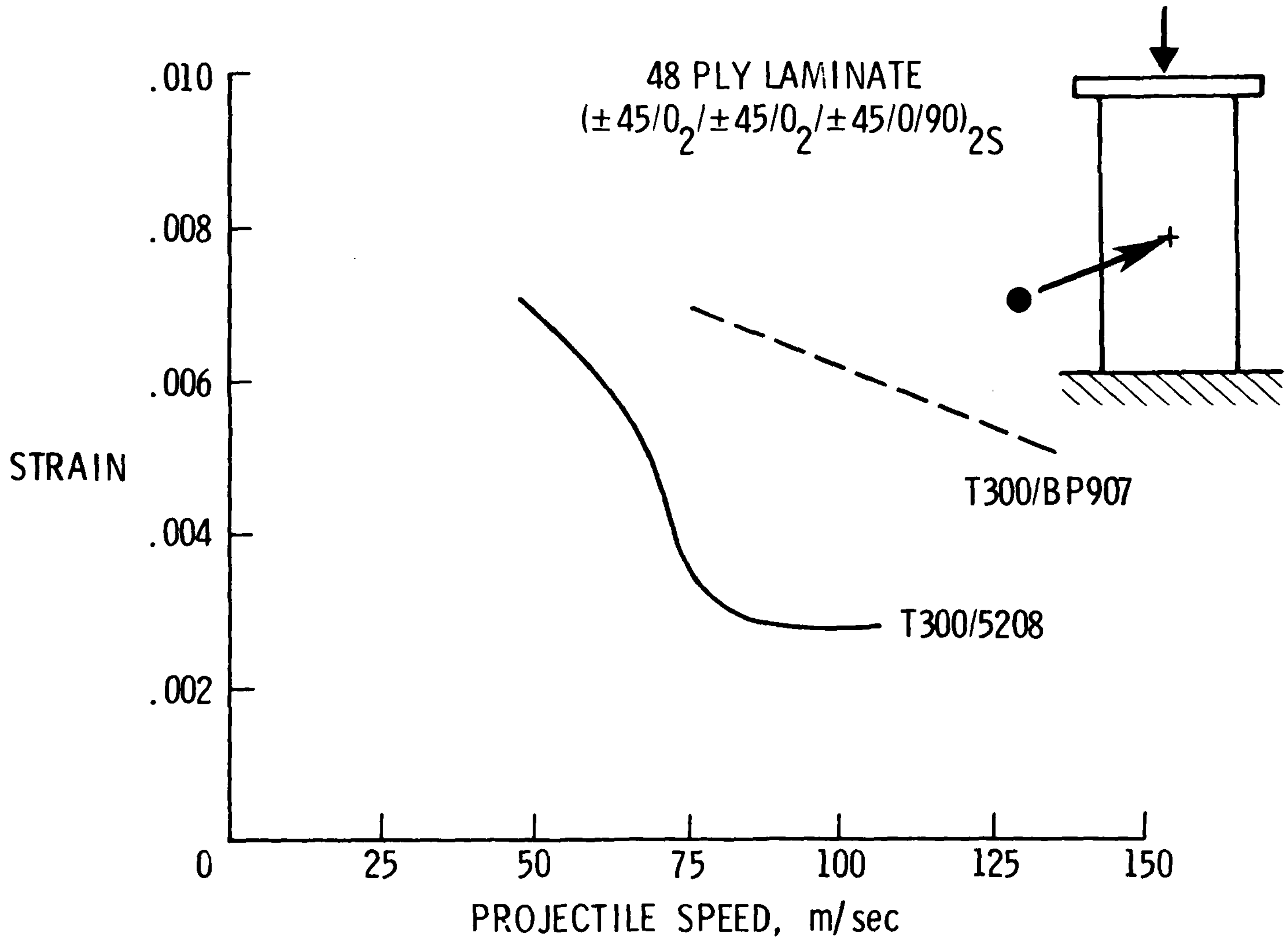
CONCLUSIONS

- USING TOUGH RESIN OR TRANSVERSE STITCHING CAN SUPPRESS DELAMINATION FAILURE MODE
- MACROSCOPIC AND MICROSCOPIC TRANSVERSE SHEAR CRIPPLING ARE DOMINANT COMPRESSION FAILURE MODES FOR DELAMINATION-RESISTANT GRAPHITE-EPOXY LAMINATES
- FIBER MICROBUCKLING HYPOTHESIZED TO BE THE MECHANISM WHICH INITIATES TRANSVERSE SHEAR CRIPPLING MODE
- TEST RESULTS INDICATE DELAMINATION-RESISTANT RESIN MATERIAL CAN SUPPRESS DAMAGE PROPAGATION FOR STIFFENED PANEL FOLLOWING STIFFENER FAILURE

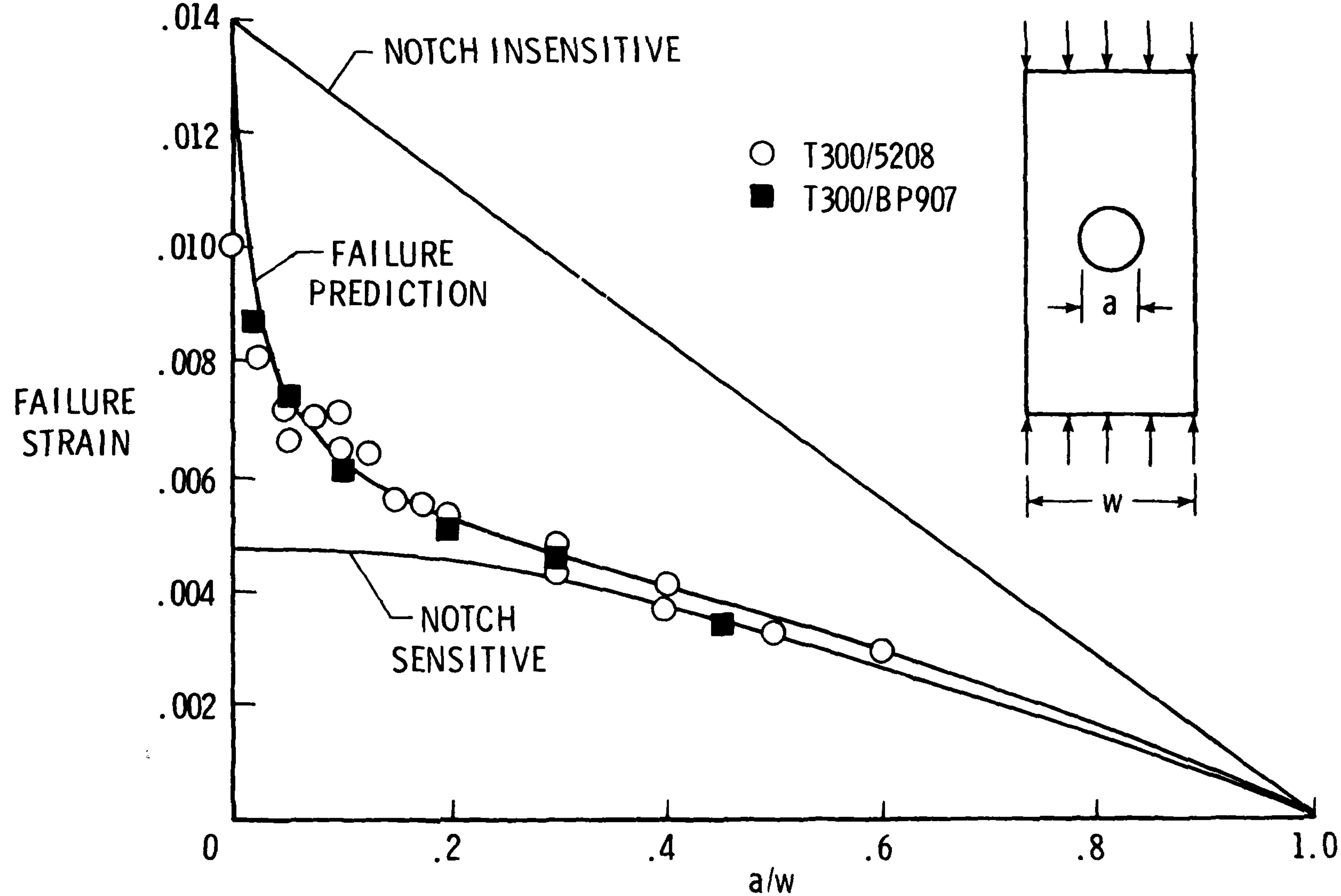


IMPACT INITIATED COMPRESSION FAILURE MODES
DELAMINATION
TRANSVERSE SHEAR
IMPACT
A
A
A-A
T300/5208
T300/BP907
1mm
1mm

IMPACT DAMAGE FAILURE THRESHOLD CURVES

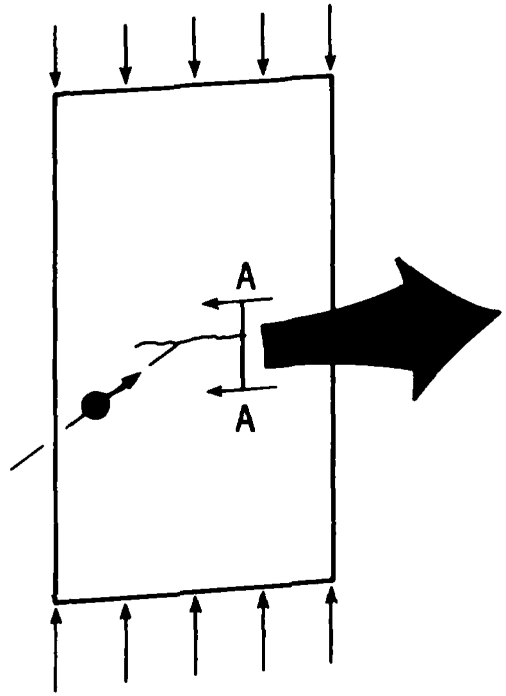


EFFECTS OF CIRCULAR HOLES ON COMPRESSIVE STRENGTH



PROPAGATION OF IMPACT INDUCED DELAMINATION

SHEAR CRIPPLING FAILURE MODE

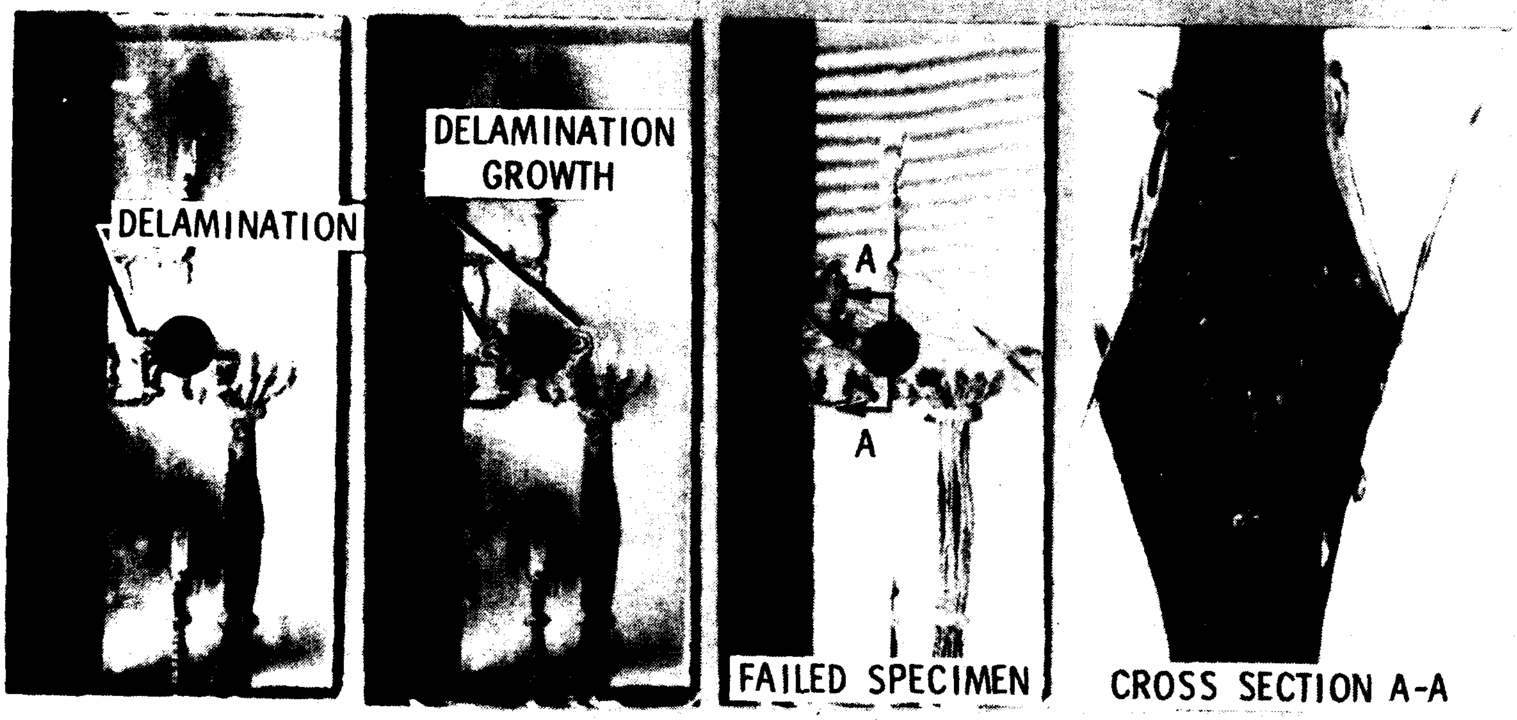


48 PLY LAMINATE
$(\pm 45/0_2/\pm 45/0_2/\pm 45/0/90)_{2S}$

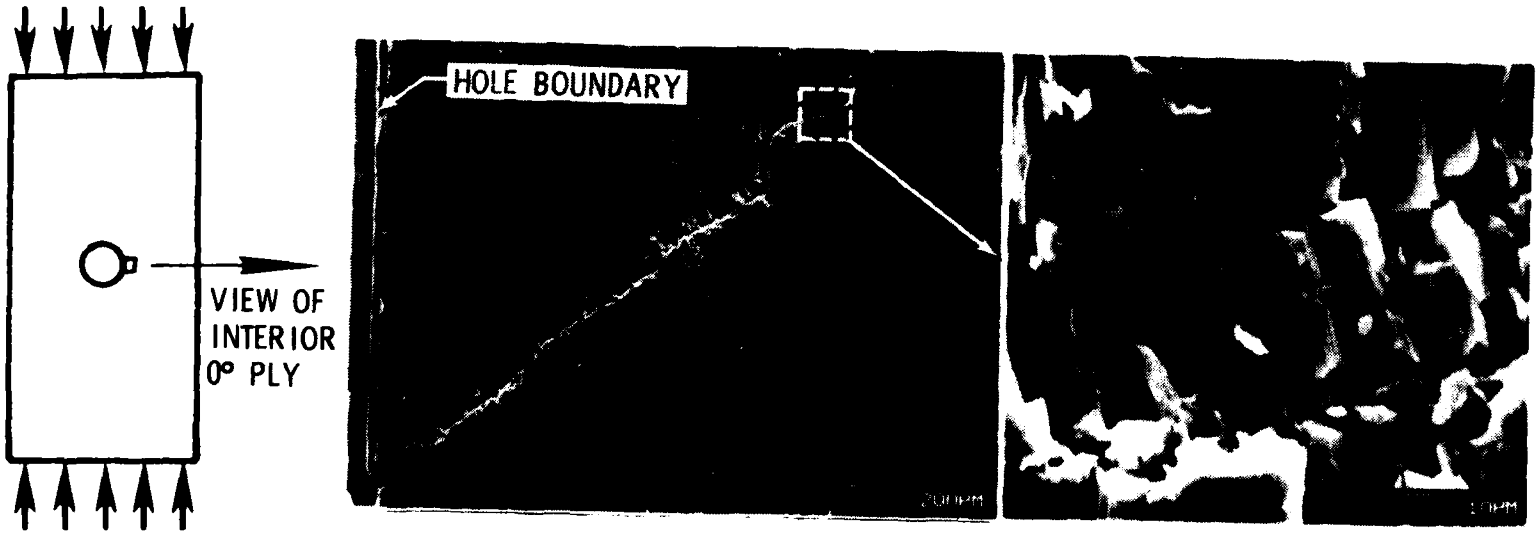
SECTION AA-TYPICAL 0_2 REGION

LOCAL FAILURE PROPAGATION FOR A 48-PLY LAMINATE

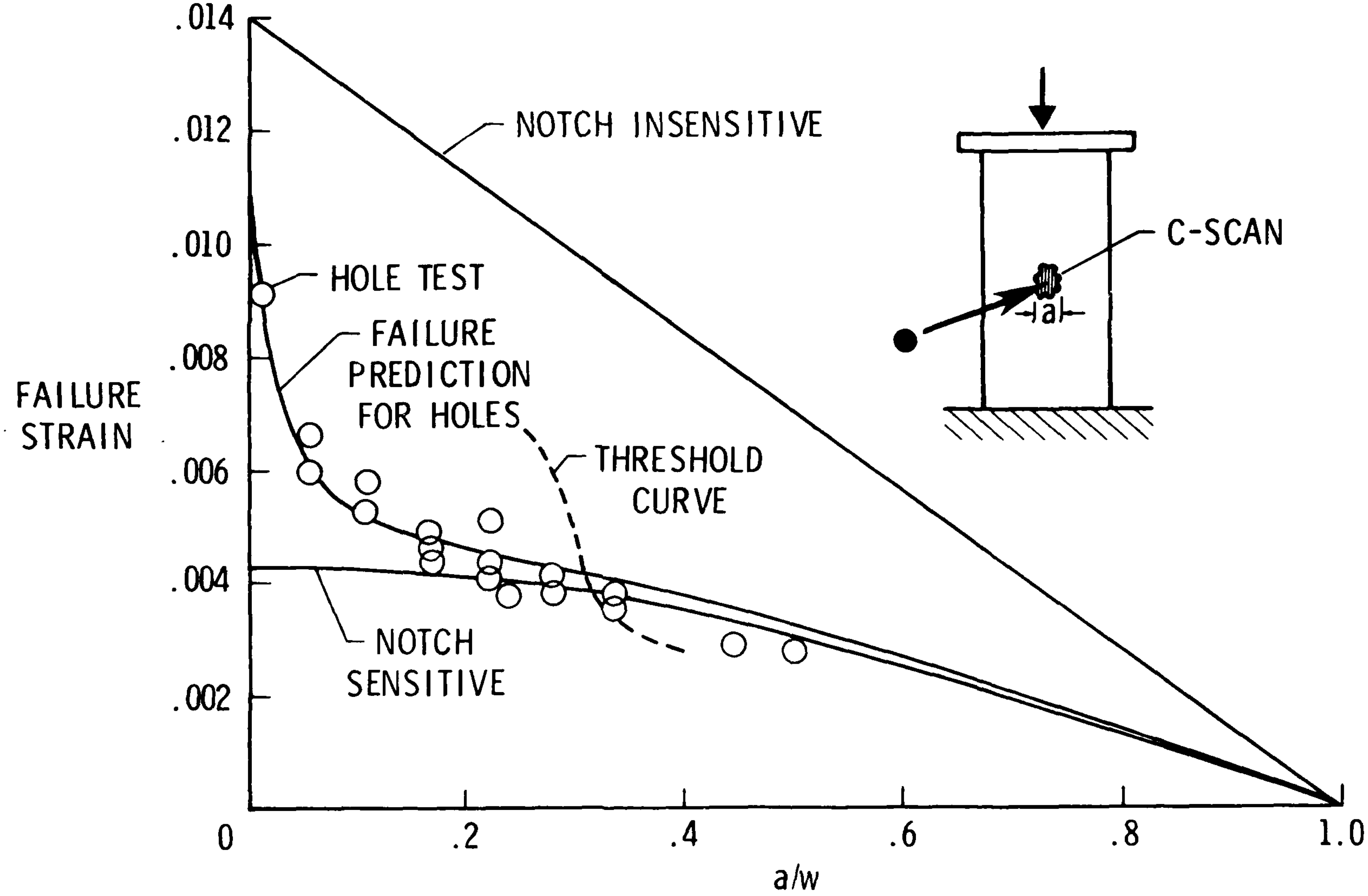
1.91-cm-DIAMETER HOLE

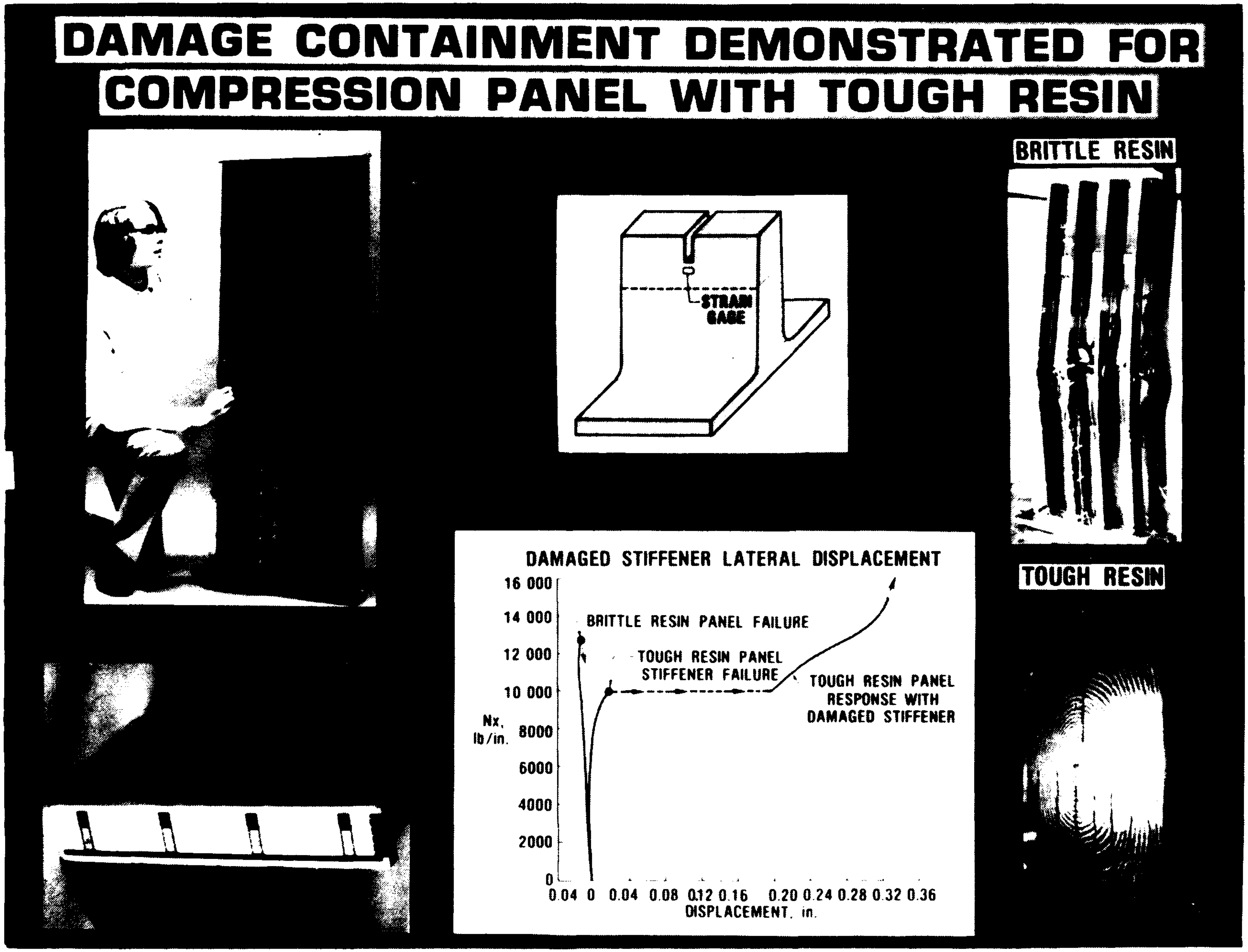


SHEAR CRIPPLING FAILURE MODE



COMPARISON OF IMPACT DAMAGE AND CIRCULAR HOLES ON COMPRESSIVE STRENGTH





DAMAGE CONTAINMENT DEMONSTRATED FOR
COMPRESSION PANEL WITH TOUGH RESIN
STRAIN GAGE
BRITTLE RESIN
TOUGH RESIN
DAMAGED STIFFENER LATERAL DISPLACEMENT
BRITTLE RESIN PANEL FAILURE
TOUGH RESIN PANEL STIFFENER FAILURE
TOUGH RESIN PANEL RESPONSE WITH DAMAGED STIFFENER
Nx, lb/in.
16 000
14 000
12 000
10 000
8000
6000
4000
2000
0
0.04 0 0.04 0.08 0.12 0.16 0.20 0.24 0.28 0.32 0.36
DISPLACEMENT, in.

SUMMARY OF IMPACT WORK

FATIGUE & FRACTURE BRANCH

Walter Illg

NASA Langley

OCTOBER 1982

OBJECTIVES

- TO ESTABLISH, THROUGH SLOW TRANSVERSE LOADING, THE ENERGY CAPABILITIES OF VARIOUS COMPOSITES;
- TO DEVELOP A SIMPLE ANALYTIC METHOD TO PREDICT PROGRESSIVE IMPACT DAMAGE IN QUASI-ISOTROPIC COMPOSITE LAMINATES;
- TO DETERMINE THE EFFECT OF DELIBERATELY FABRICATED PARTIAL BONDING ON THE IMPACT RESISTANCE OF COMPOSITE LAMINATES UNDER PRESTRAIN.
- TO PREDICT THE RESIDUAL TENSILE STRENGTH OF IMPACTED LAMINATES.

CONCLUSIONS

- FOR 8-PLY 1/2-INCH-DIAMETER GRAPHITE LAMINATES UNDER SLOW LOADING:
  - POLYSULFONE MATRIX RESULTS IN MUCH LESS DELAMINATION AND SUSTAINS SOMEWHAT LARGER TRANSVERSE LOAD THAN EPOXY MATRIX.
  - ENERGY ABSORBED TO PENETRATION IS 50 PERCENT MORE IN POLYSULFONE MATRIX THAN IN EPOXY MATRIX.

- A SIMPLE ENERGY METHOD PREDICTS REALISTIC IMPACT DAMAGE PROGRESSION. IT PREDICTS INITIAL DAMAGE IN THE LOWER PLIES.

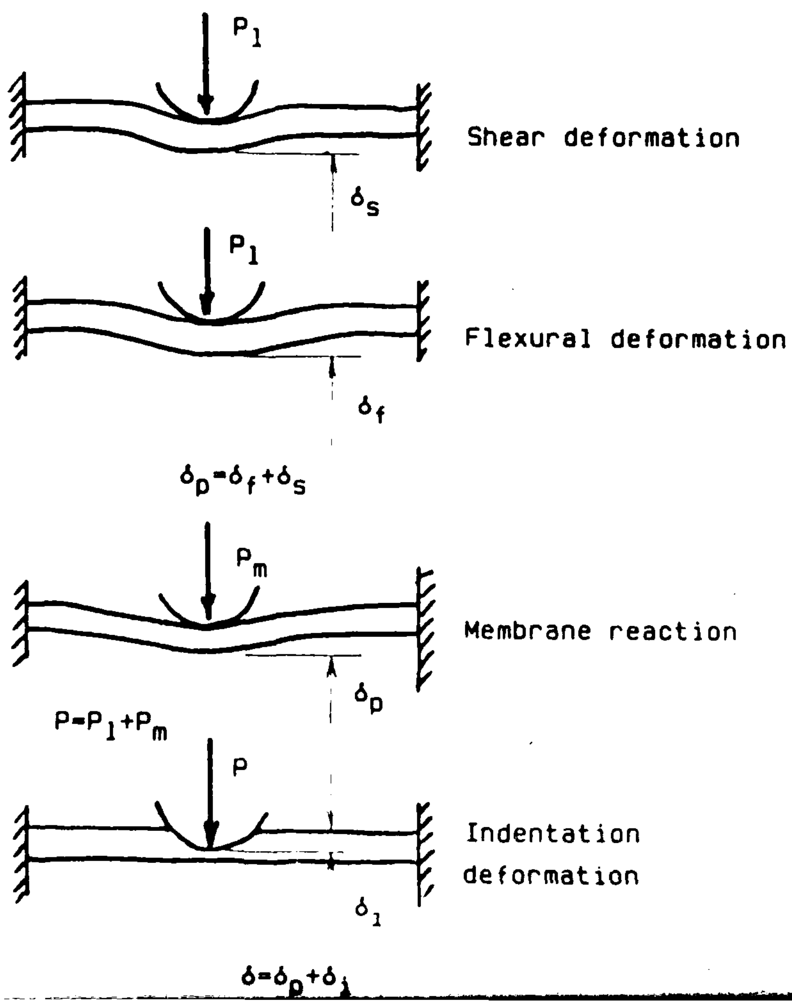
- THE TENSILE PRELOAD FOR THE THRESHOLD OF IMPACT-COLLAPSE OF 24-PLY LAMINATES IS UNAFFECTED BY PARTIAL BONDING. BUT FOR COMPRESSIVE PRELOAD THE THRESHOLD IS DRASTICALLY REDUCED.

- THE DELAMINATION PATTERN IS OVAL AND SYMMETRIC FOR COMPRESSIVE PRELOAD, BUT IS SKEWED AND UNSYMMETRIC FOR TENSILE PRELOAD.

- AN ENERGY METHOD PREDICTS A LOWER BOUND FOR RESIDUAL TENSILE STRENGTHS OF IMPACTED LAMINATES.

PLATE DEFORMATION MECHANICS

including membrane action



Shear and flexural deformations are combined to obtain linear deformations for the plate reaction.

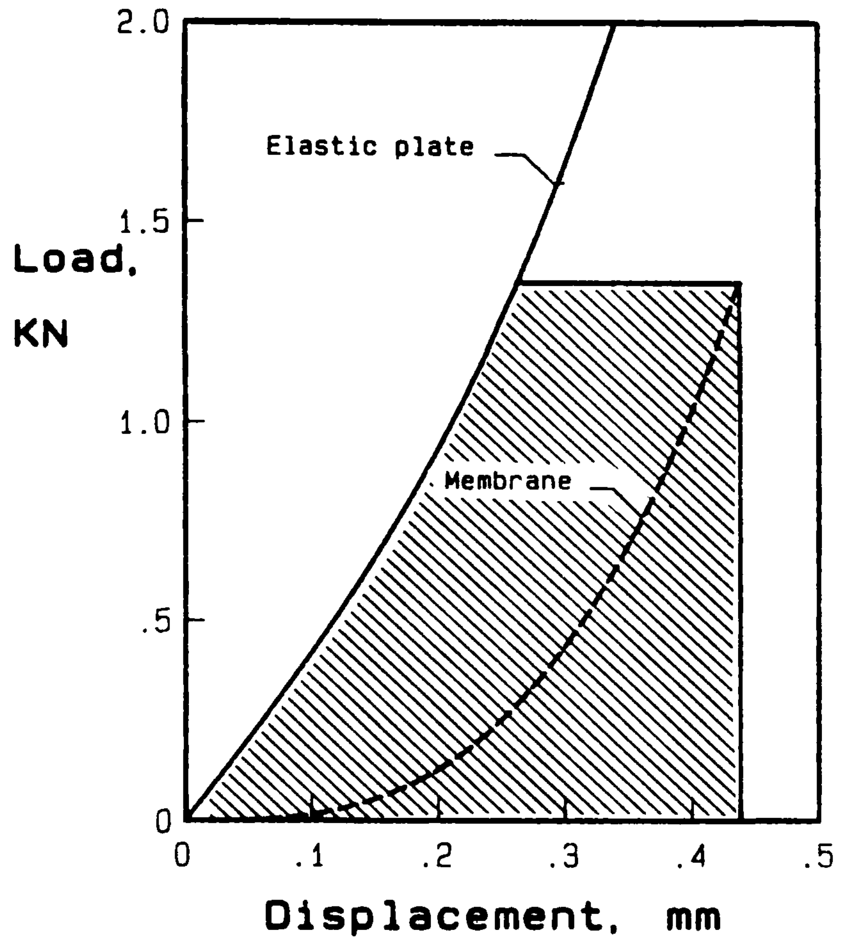
Membrane reaction is calculated for this total deformation.

Total reaction is the sum of the plate reaction and the membrane reaction.

Total deformation is the sum of the plate deformation and the indentation deformation.

STRAIN ENERGY LIMITS

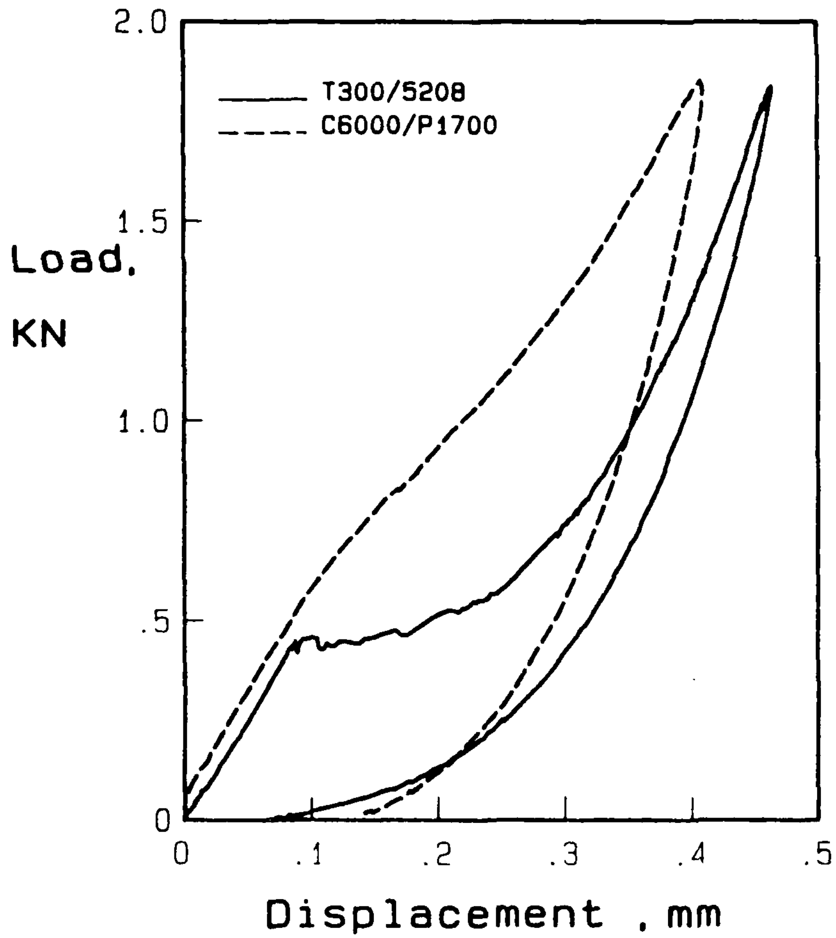
8-Ply Graphite-Epoxy plates



* *With an ideal matrix, plate could remain perfectly elastic up to the fiber/membrane failure load.*
* *Releasing all matrix shear strain energy at that load results in the maximum strain energy (cross-hatched area).*
* *In large plates, most energy is in the fibers. Fibers fail without much delamination.*
* *In small plates most strain energy is in the matrix. Significant delamination occurs before fiber failure.*

EFFECT OF MATRIX ON HYSTERESIS

1/2 INCH APERTURE 8-PLY PLATES



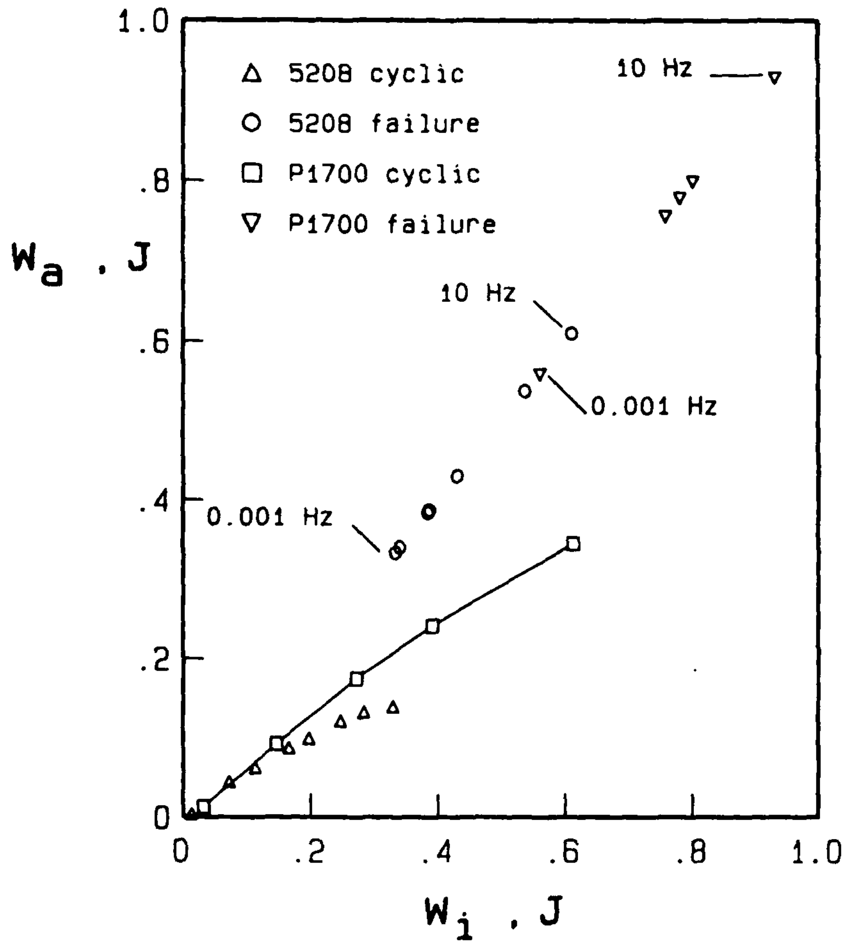
5208 Epoxy

- *Has well defined elastic range*
- *Grows delaminations*
- *Loses local flexural stiffness*
- *Small residual displacement*

P1700 Polysulfone

- *Significant creep*
- *Shows ductile yield*
- *Retains significant residual displacement*
- *Retains local stiffness*

IMPACT ENERGY ABSORBED BY EPOXY AND POLYSULFONE



o Both materials absorb more energy at high loading rates.

o P1700 absorbs 2/3 of maximum energy by yielding.

o 5208 absorbs 1/2 of maximum energy by delaminating.

o Max penetration energy for C6000/P1700 is 0.93 J @ 10 Hz loading rate.

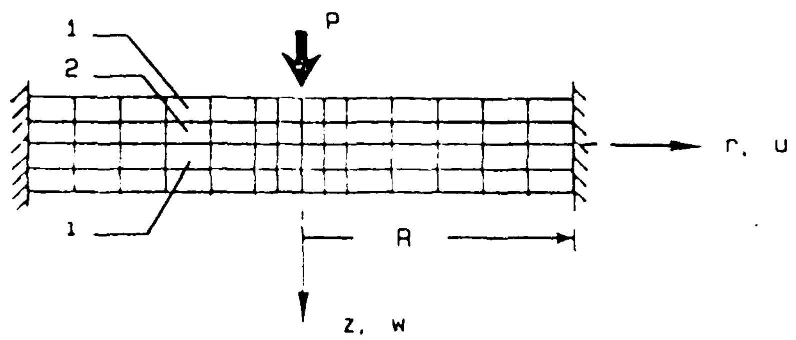
o Max penetration energy for T300/5208 is 0.63 J @ 10 Hz.

Wi, impact energy

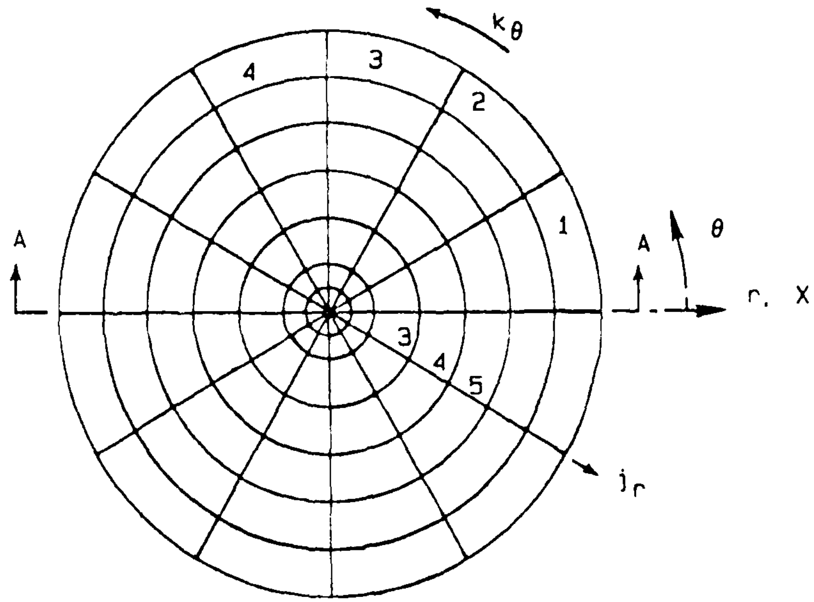
Wa, energy absorbed

POTENTIAL ENERGY MODEL

Circular Laminated Plates

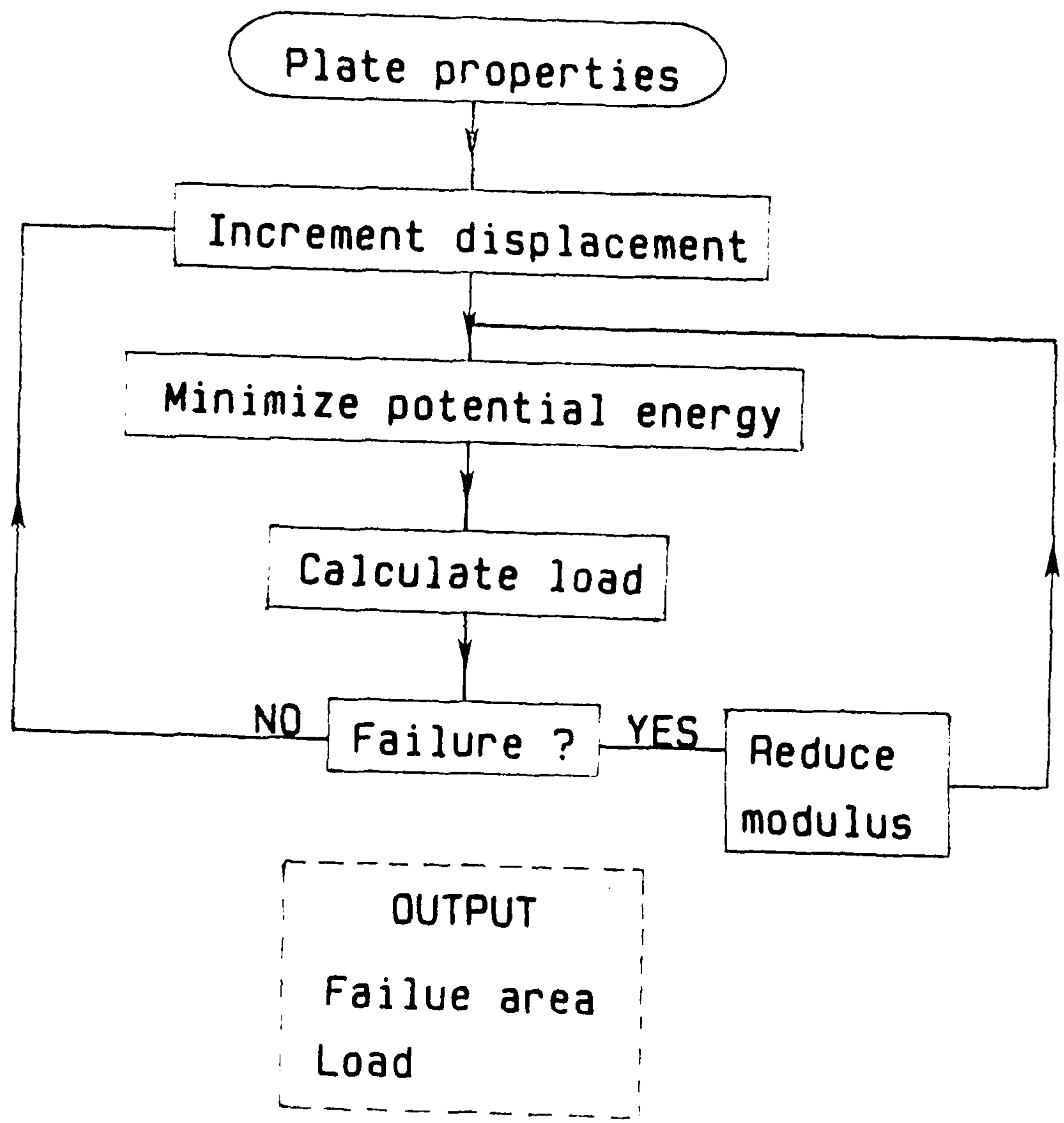


Section A-A



i^{th} layer idealization

DAMAGE ANALYSIS FLOW CHART



PLY-BY-PLY DAMAGE AT P=502N

T300/5208 Gr/Ep. R=34.9 mm. 8 plys

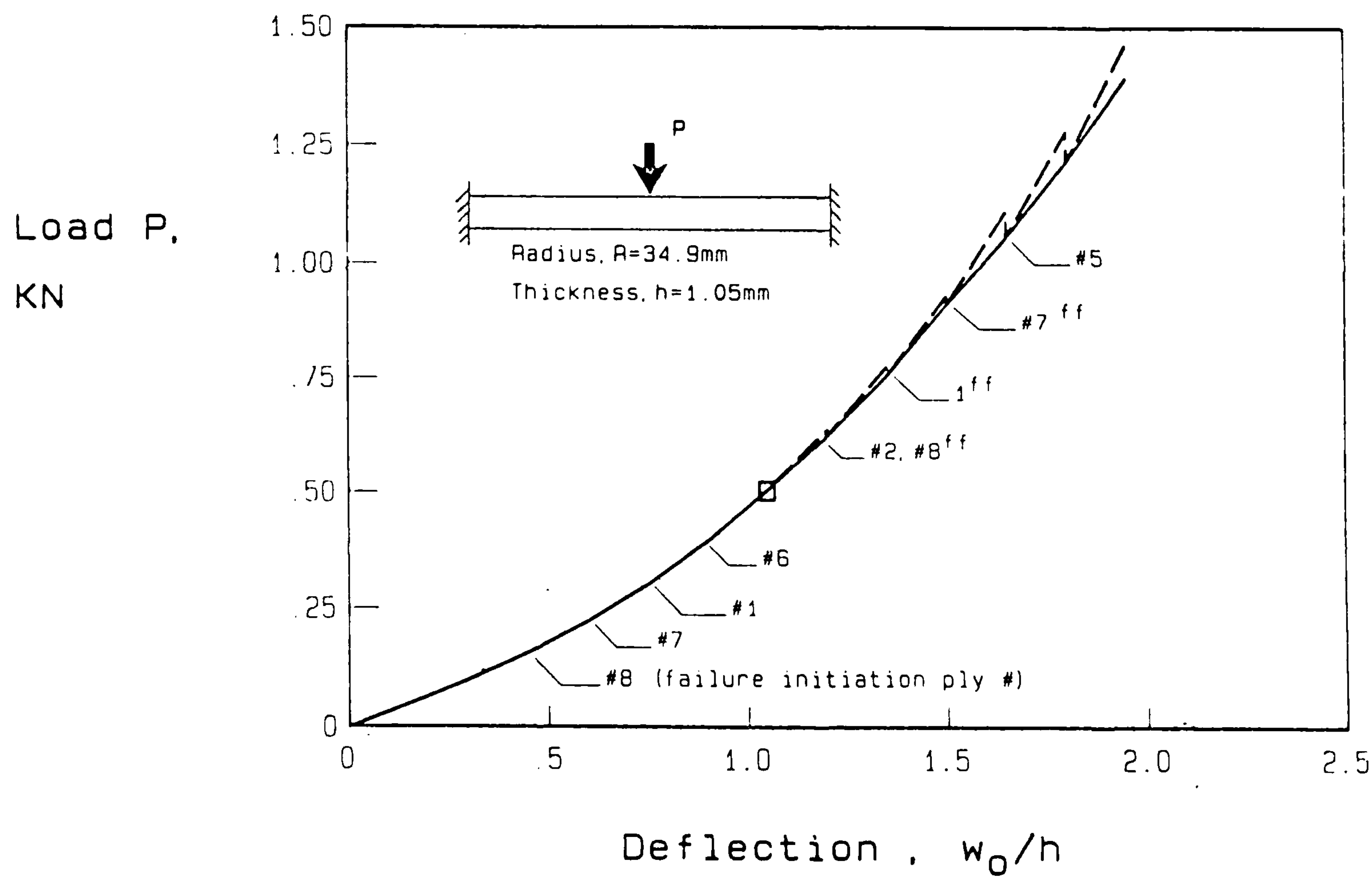
Damage scale 10 mm

Ply #	Fiber angle	Damage
1	45°	
2	0°	
3	-45°	
4	90°	

Ply #	Fiber angle	Damage
5	90°	
6	-45°	
7	0°	
8	45°	

PLATE FAILURE SEQUENCE

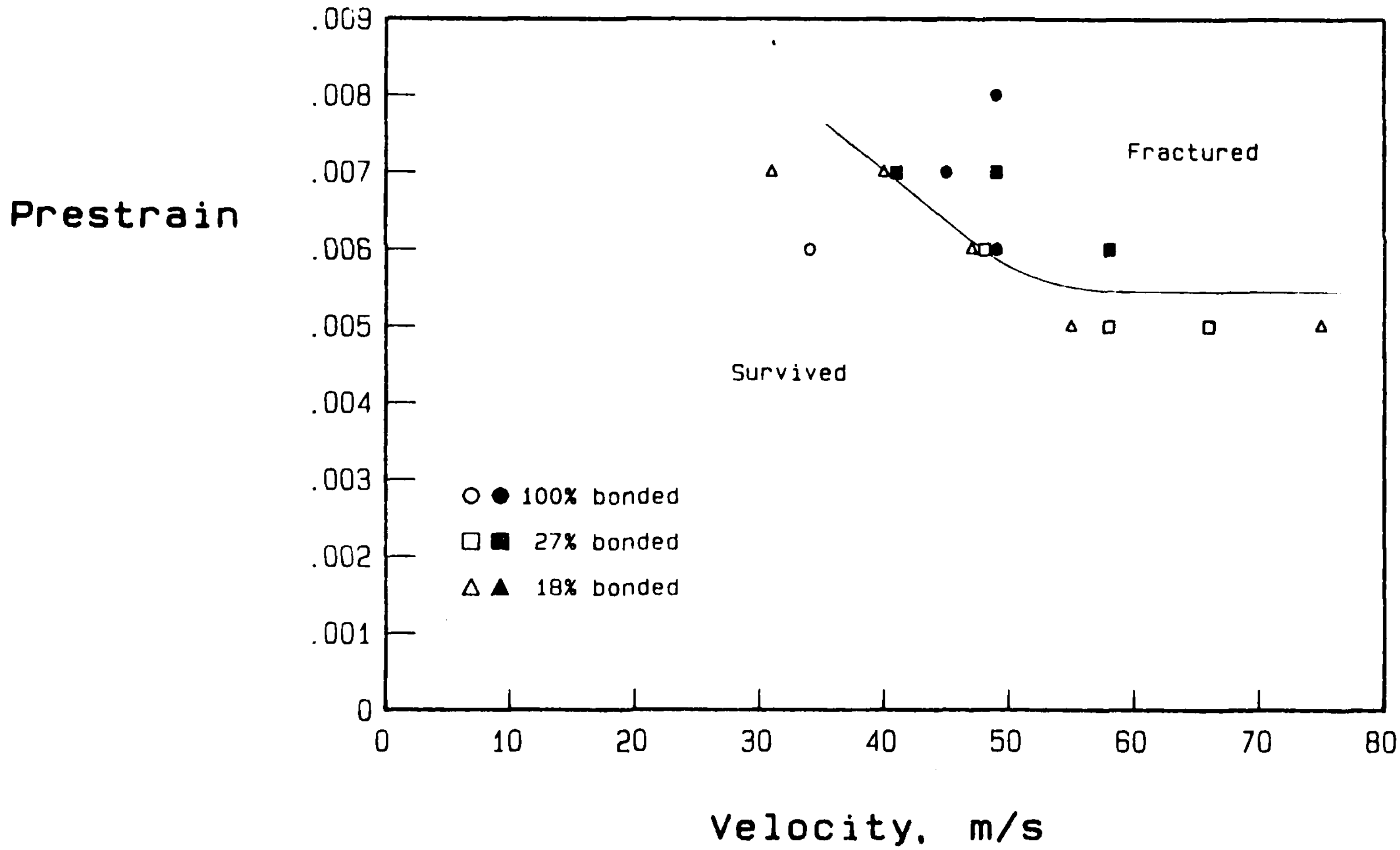
Gr/Ep 5208/T300, $[45/0/-45/90]_S$



IMPACT UNDER TENSION

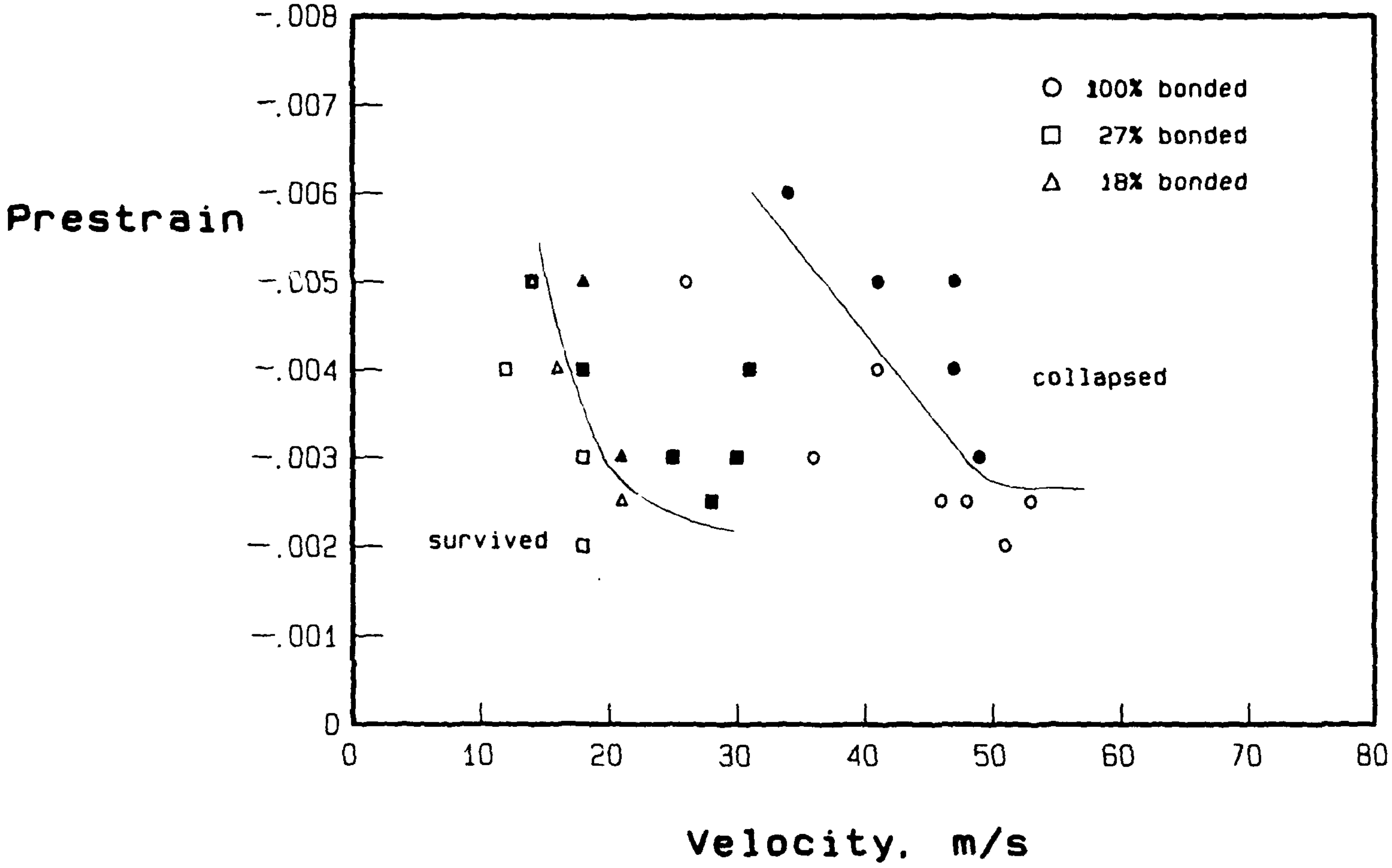
24 PLIES T300/5208

Partially Bonded



IMPACT UNDER COMPRESSION

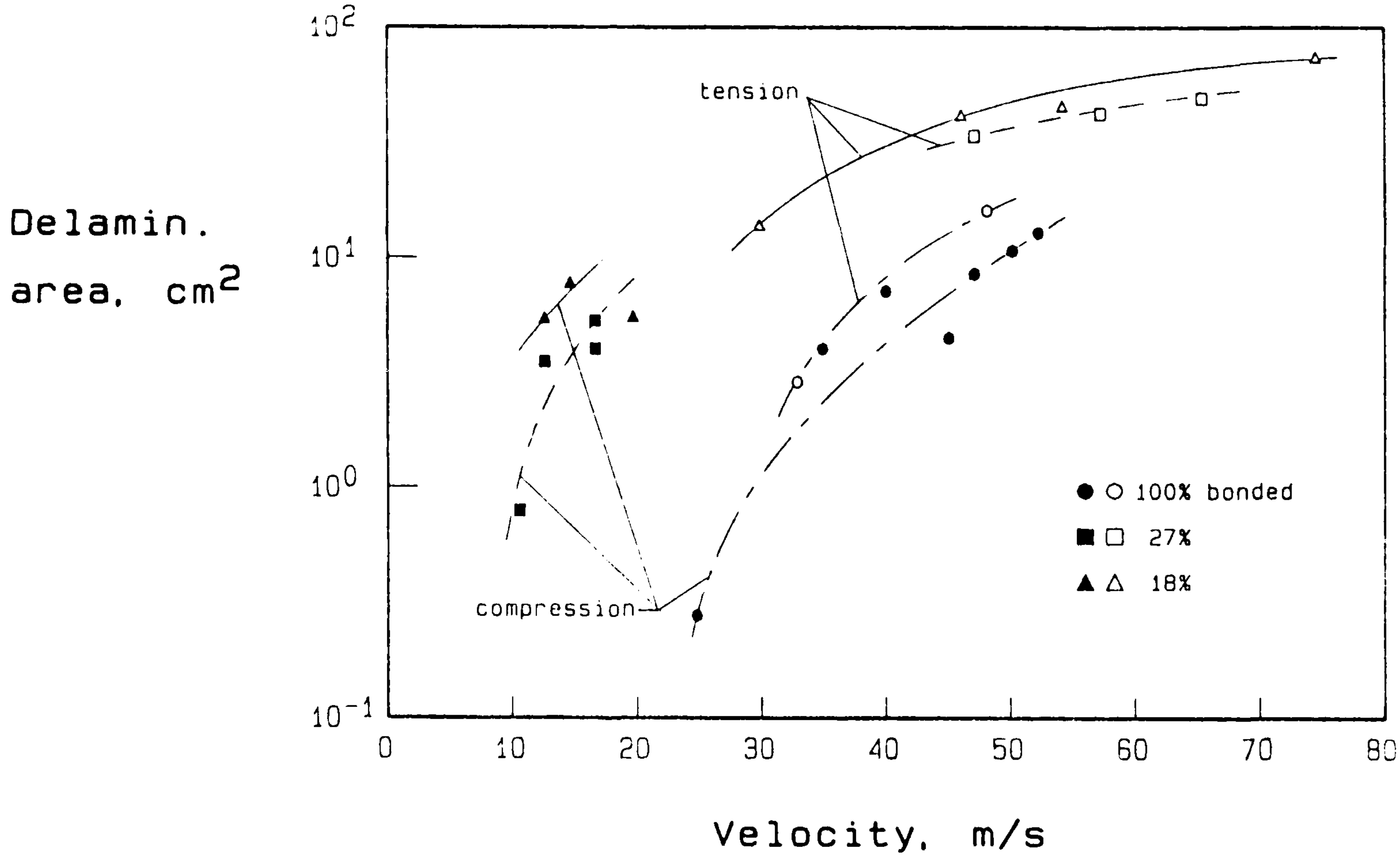
24 PLIES T300/5208



IMPACT DELAMINATION UNDER LOAD

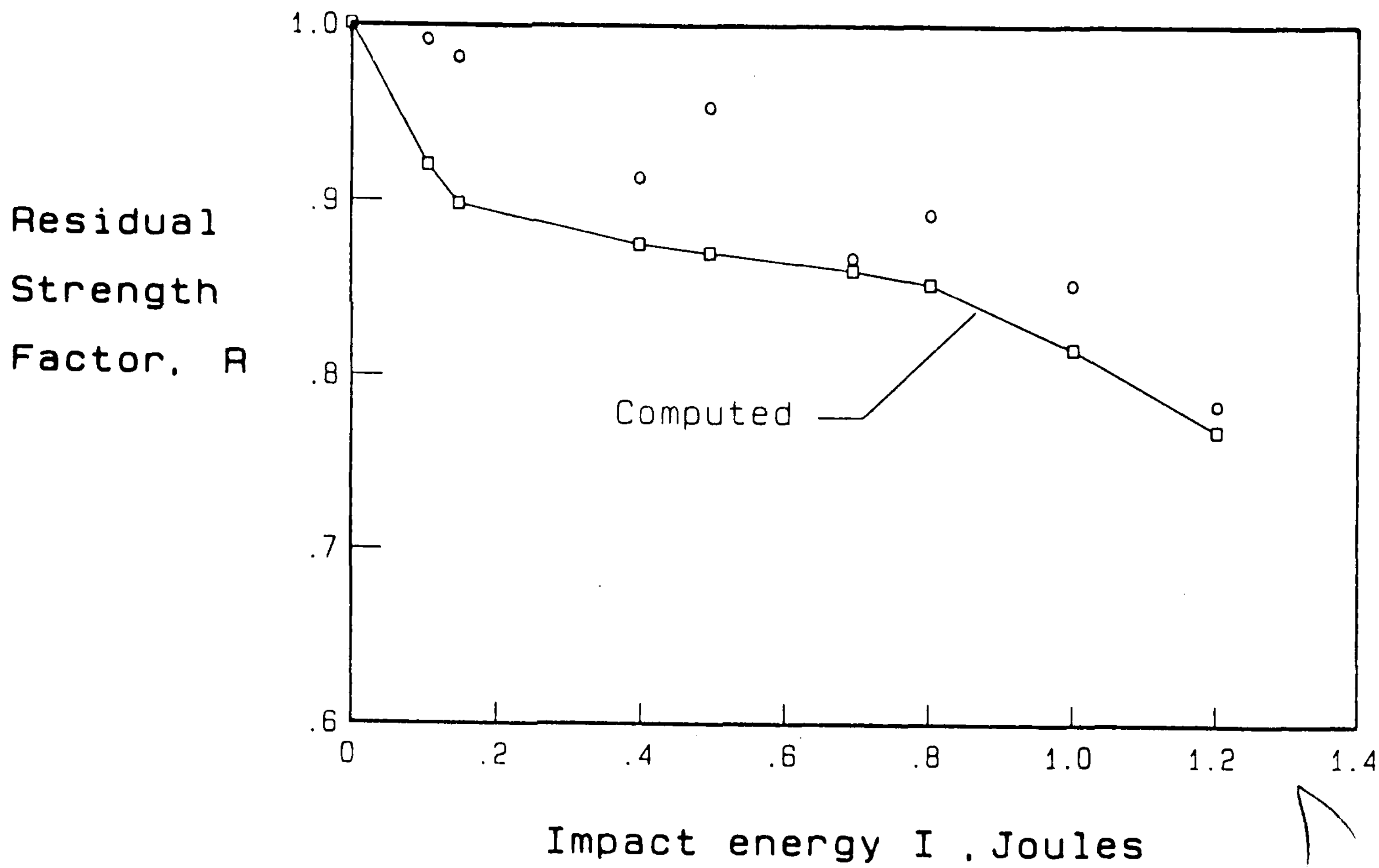
24 PLIES T300/5208

Survivors



EFFECT OF IMPACT ON STRENGTH

T300/5208 graphite epoxy 8-ply plates



AD P001249

CHARACTERIZATION OF INTERLAMINAR FRACTURE TOUGHNESS

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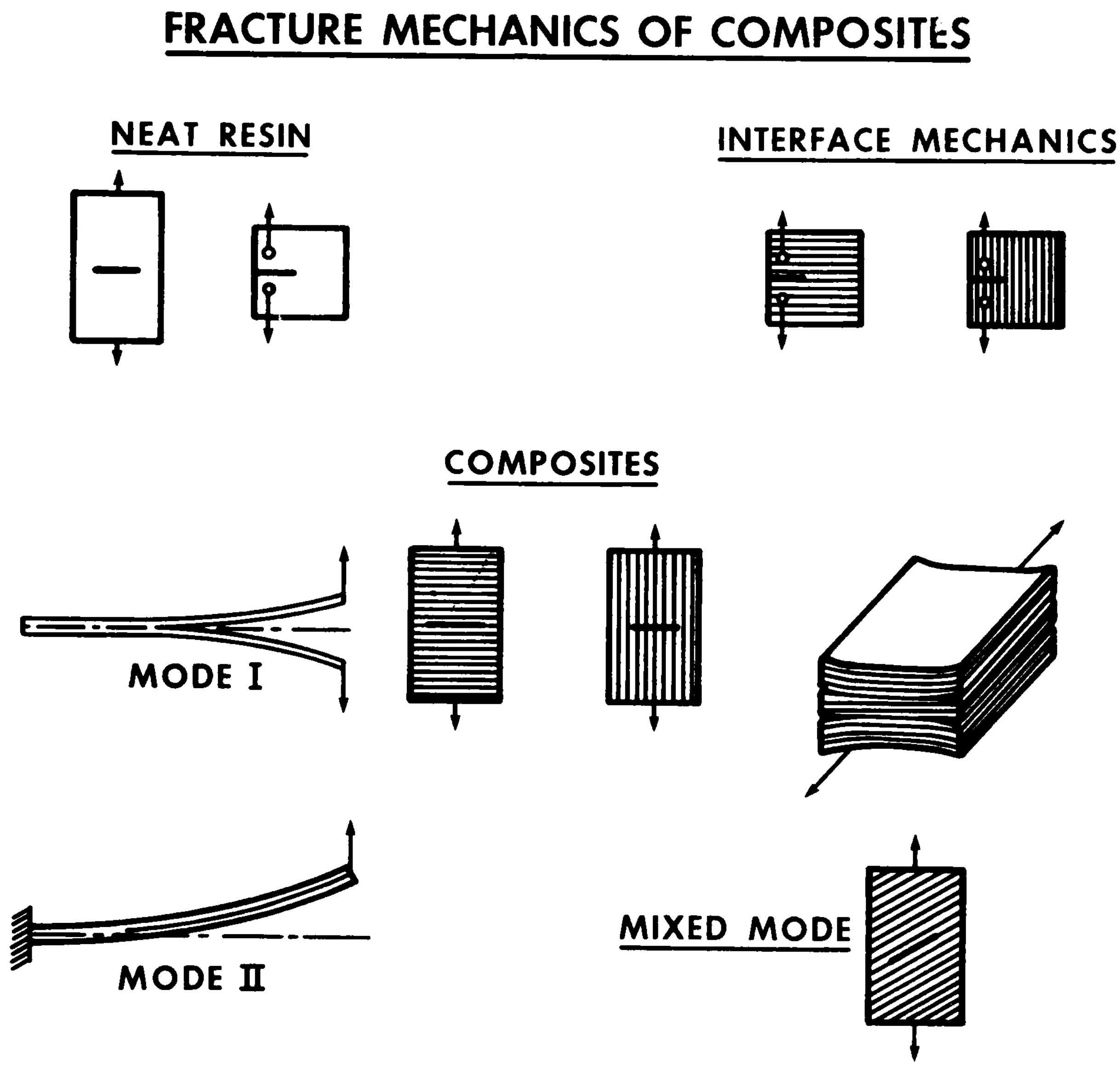
OBJECTIVE: TO DEVELOP TEST METHODS FOR CHARACTERIZING INTERLAMINAR FRACTURE OF UNIDIRECTIONAL AND MULTI-DIRECTIONAL FIBER REINFORCED POLYMERIC MATRIX COMPOSITES.

APPROACH: ASSESS BOTH EXPERIMENTALLY AND ANALYTICALLY VARIOUS APPROACHES TO DEVELOPING MODE I, MODE II, AND MIXED MODE FRACTURE CHARACTERIZATION TEST METHODS FOR SCREENING NEW AND/OR IMPROVED MATRIX RESINS AND COMPOSITES. NEAT RESIN TEST METHODS ARE ALSO INCLUDED. CURRENT EFFORT IS FOCUSED ON DOUBLE CANTILEVER BEAM TEST FOR MODE I.

CONCLUSIONS

- DOUBLE CANTILEVER BEAM TEST IS A VIABLE CANDIDATE FOR MEASURING INTERLAMINAR MODE I FRACTURE RESISTANCE
- THE AREA METHOD OF DATA REDUCTION PROVIDES A VERY DIRECT APPROACH FOR ESTIMATING G_{IC} FOR MATERIALS UNDERGOING ELASTIC RESPONSE
- FUTURE WORK SHOULD FOCUS ON MULTIDIRECTIONAL LAMINATES WITH THE DEB TEST BEING COMPARED TO RESULTS OBTAINED FROM THE FREE-EDGE DELAMINATION COUPON
- CONSIDERATION SHOULD BE GIVEN TO DATA REDUCTION PROCEDURES FOR MATERIALS DISPLAYING VISCOELASTIC RESPONSE

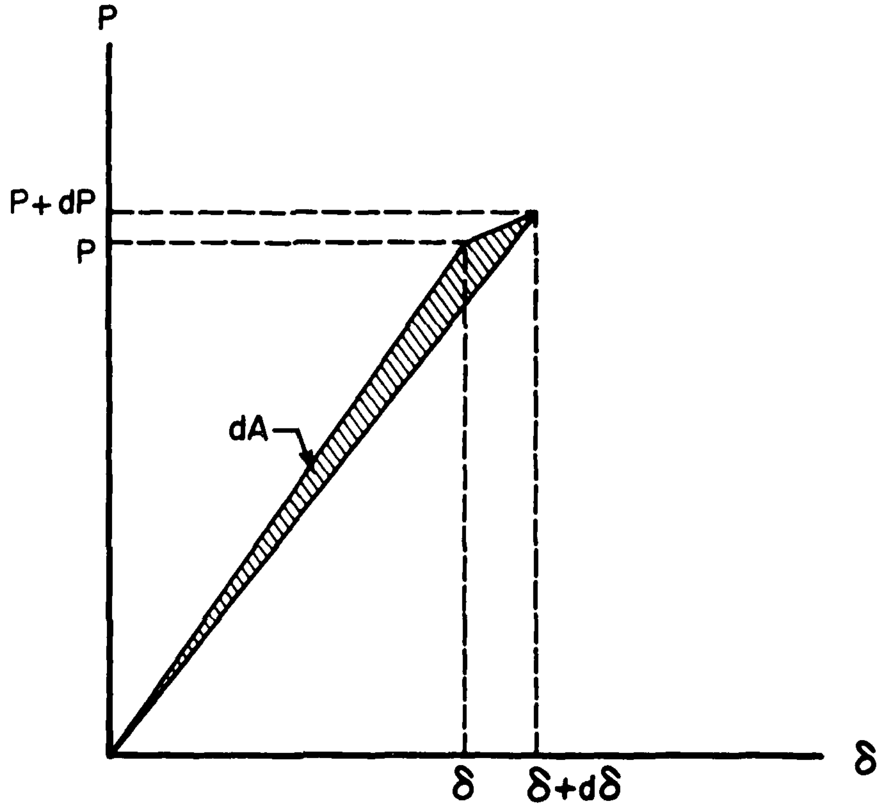

FRACTURE MECHANICS OF COMPOSITES
NEAT RESIN
INTERFACE MECHANICS
COMPOSITES
MODE I
MODE II
MIXED MODE

STRAIN ENERGY RELEASE RATE

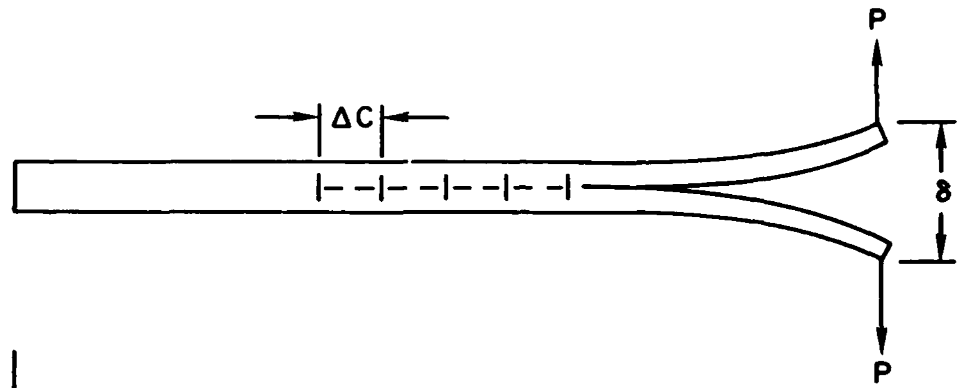
$$G_I = -\frac{1}{b}\frac{dU}{da}$$

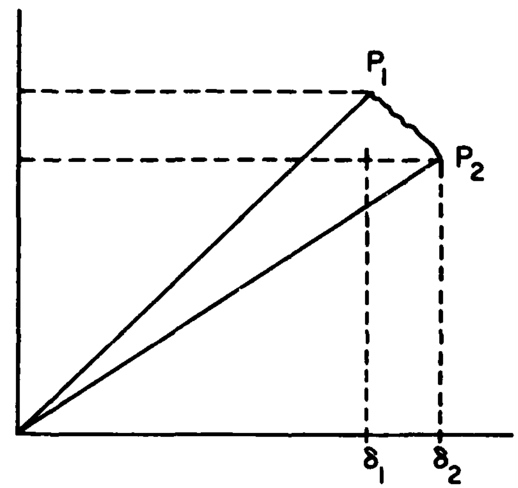
$$-dU = dA = \frac{1}{2}(Pd\delta - \delta dP)$$

$$G_I = \frac{1}{b}\frac{dA}{da} = \frac{1}{2b}\left(P\frac{d\delta}{da} - \delta\frac{dP}{da}\right)$$



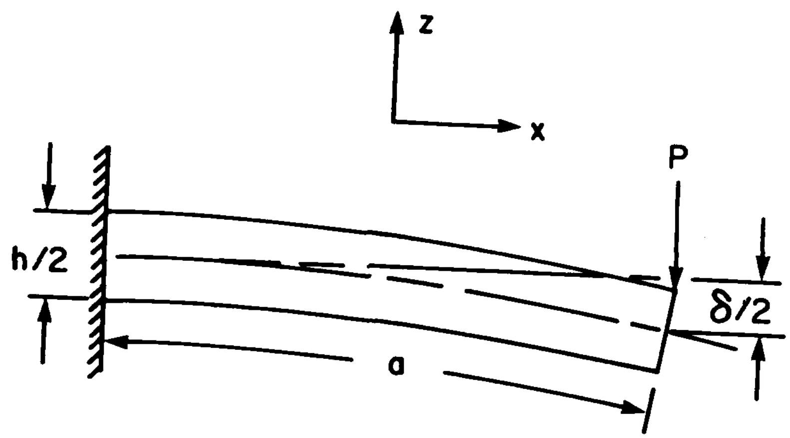
MODE I TEST FOR INTERLAMINAR TOUGHNESS





$$G_{IC} = \frac{1}{2b\,\Delta c}\,(P_1\,\delta_2 - P_2\,\delta_1)$$

OTHER DATA REDUCTION SCHEMES

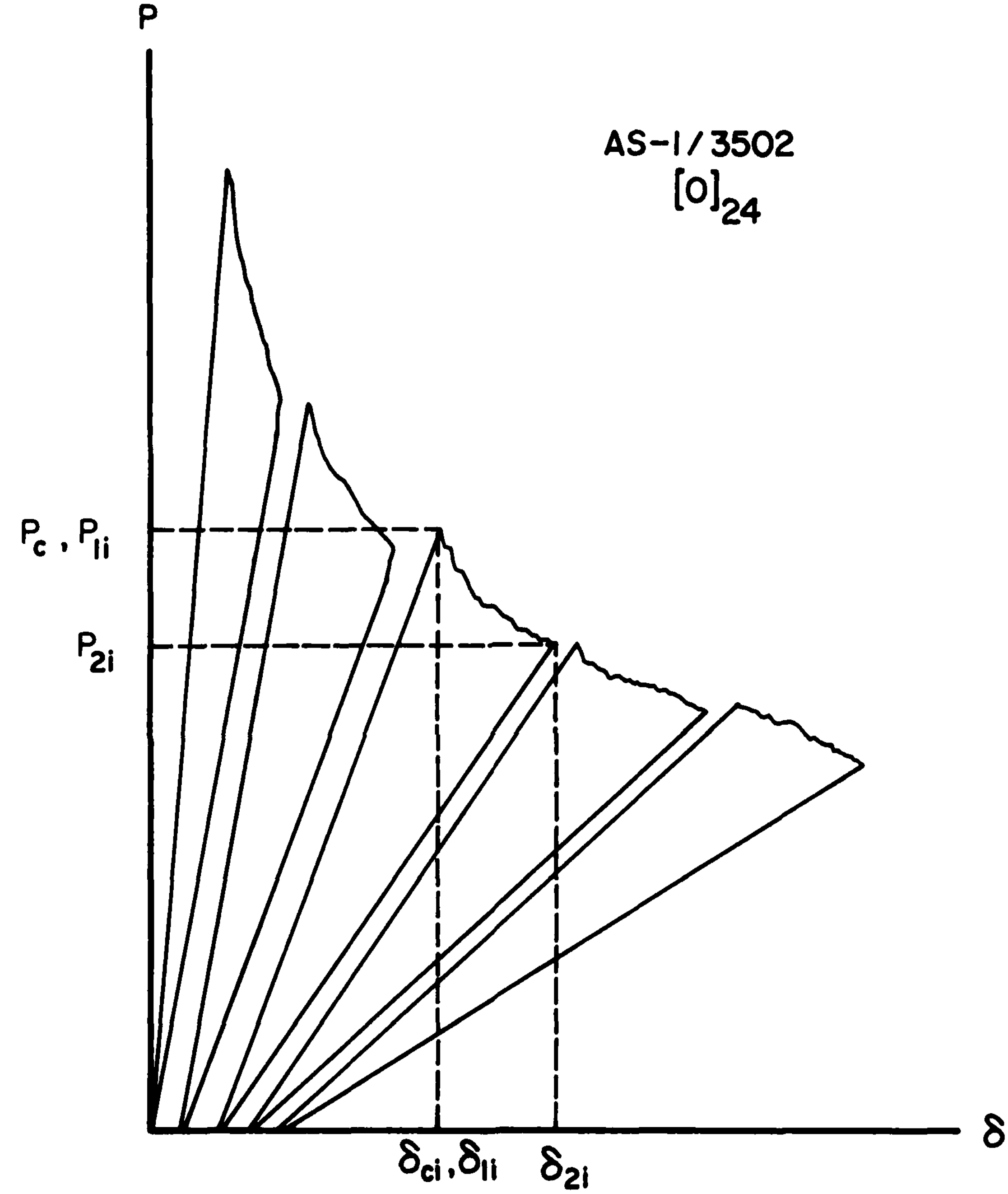


BEAM ANALYSIS: $\delta = BPa^3$; $B = \dfrac{64}{E_x^b bh^3}$

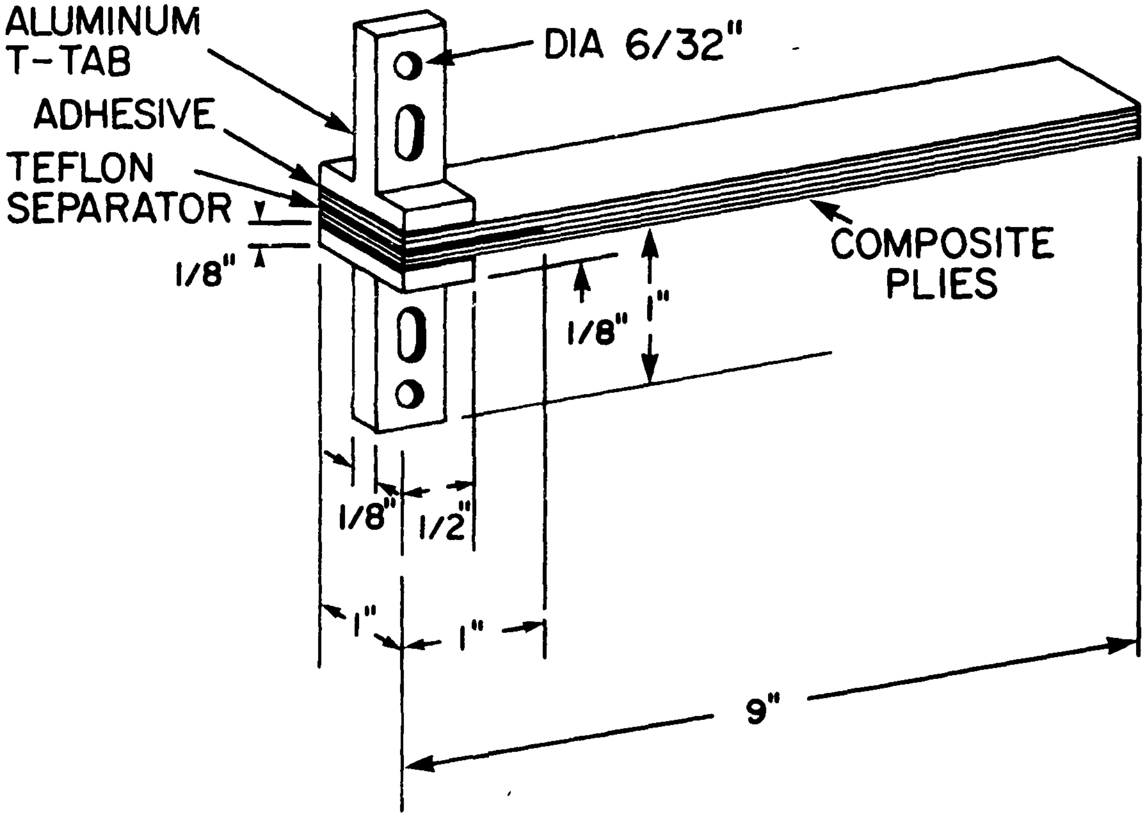
$$G_{Ic} = \frac{3P_c\delta_c}{2bc}$$

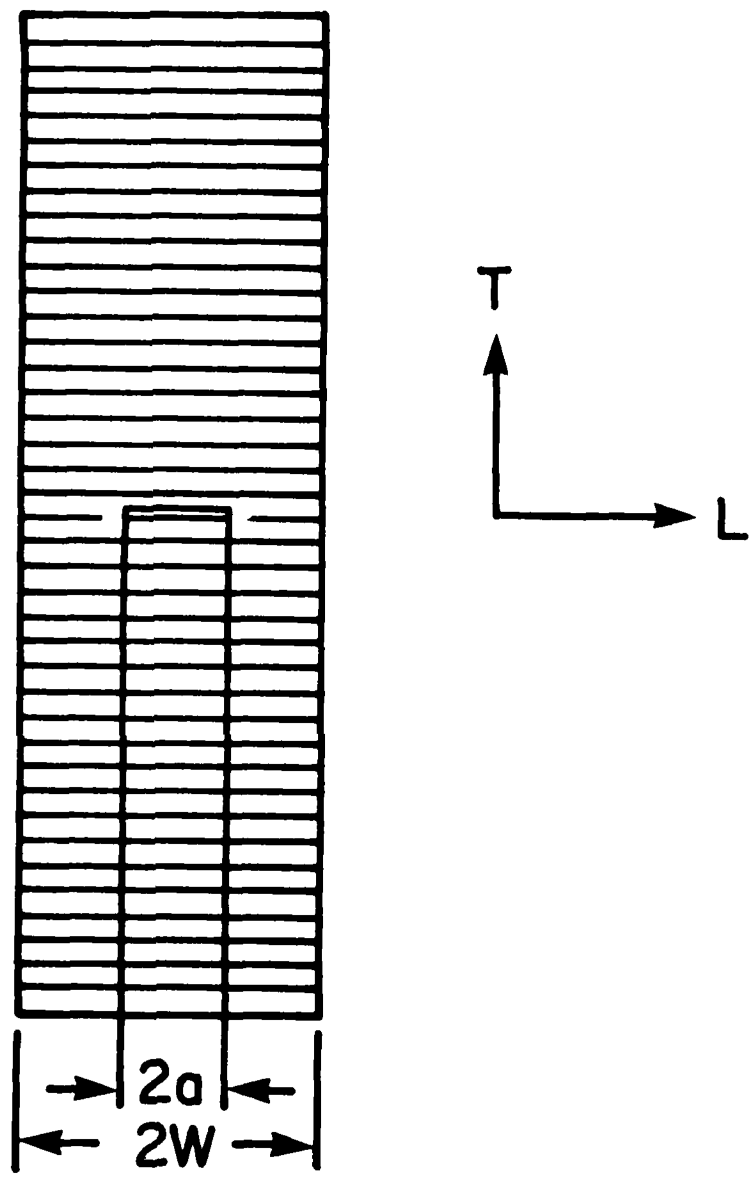
EMPIRICAL BEAM ANALYSIS: $\delta = RPa^n$

$$G_{Ic} = \frac{nP_c\delta_c}{2ba}$$



P
AS-1/3502
[0]24
Pc , P1i
P2i
δci , δ1i
δ2i
δ


ALUMINUM
T-TAB
DIA 6/32"
ADHESIVE
TEFLON
SEPARATOR
1/8"
COMPOSITE
PLIES
1/8"
1"
1/8"
1/2"
1"
1"
9"


T
L
2a
2W

COMPARISON OF METHODS FOR DETERMINING G_{IC}

AS-1/3502 UNIDIRECTIONAL	
METHOD	G_{IC} (LB/IN)
DCB, $[0]_{24}$, AREA METHOD	0.801
DCB, $[0]_{24}$, BEAM ANALYSIS	0.691
DCB, $[0]_{24}$, EMPIRICAL ANALYSIS	0.861
CENTER NOTCH, $[90]_{16}$	0.881

MODE I DCB TEST

MATERIAL	G_{IC} (LB/IN)
AS1/3502	0.756
AS1/3502 MINI PANELS	0.828
AS1/3502 MINI PANEL (MATCHED METAL MOLDS)	1.266
AS1/POLYSULFONE	3.740
T300/V378-A	0.452
AS1/3502 GLASS SCRIM	1.472
AS1/3502 KEVLAR DRY (PAPER) INLAY	2.286
AS1/3502 AF-163U ADHESIVE INLAY	7.283

Materials Sciences Corporation

COMPOSITE DEFECT SIGNIFICANCE

BY

S.N. CHATTERJEE, MATERIALS SCIENCES CORPORATION

AND

R.B. PIPES, UNIVERSITY OF DELAWARE

FOR

NAVAL AIR DEVELOPMENT CENTER, WARMINSTER, PA

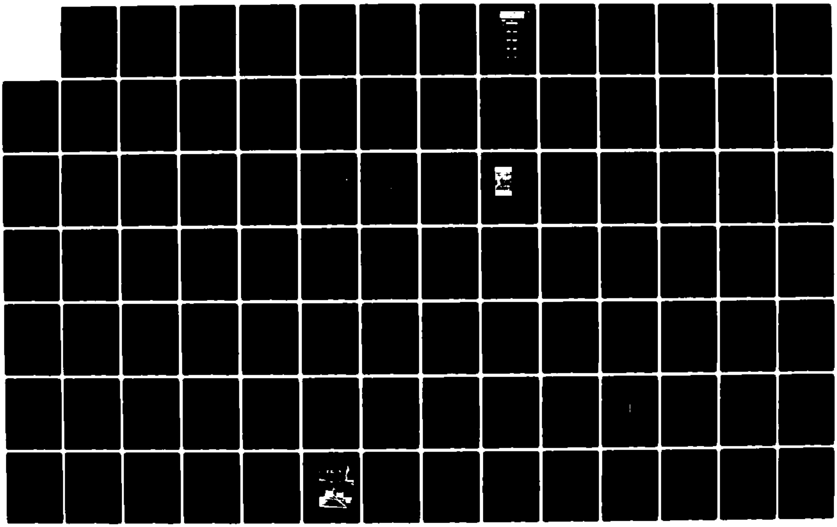
OBJECTIVE

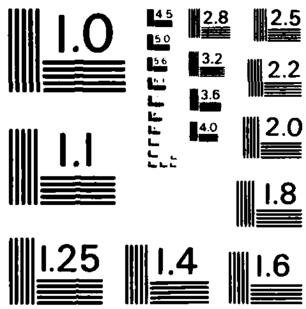
- o REVIEW STATE OF THE ART FOR NONDESTRUCTIVE AND ANALYTICAL EVALUATION OF DEFECT CRITICALITY IN LAMINATED COMPOSITES AND STRUCTURES
- o TEST THE VALIDITY OF CRITICALITY CRITERIA DEVELOPED IN PREVIOUS PROGRAMS FOR DISBONDS IN BEAM AND PLATE TYPE MEMBERS SUBJECTED TO QUASISTATIC COMPRESSION AND TRANSVERSE SHEAR
- o STUDY GROWTH OF ISOLATED DISBONDS AND DELAMINATIONS NEAR PLY DROPS UNDER CYCLIC LOADS
- o DEVELOP ANALYTICAL METHODOLOGIES FOR CALCULATION OF ENERGY RELEASE RATES FOR MULTIPLE DISBONDS IN PLATE OR BEAM TYPE MEMBERS UNDER GENERALIZED PLANE STRAIN OR PLANE STRESS CONDITIONS

CONCLUSIONS

STATE OF THE ART:	THE FIELD IS RAPIDLY EXPANDING. NDI TECHNIQUES ARE AVAILABLE FOR QUANTIFYING MANY KINDS OF DEFECTS. VARIOUS MODELING AND STRESS ANALYSIS METHODS ARE BEING EMPLOYED FOR CRITICALITY ASSESSMENT, ALTHOUGH VALIDITY OF THESE METHODS HAVE NOT BEEN FULLY DEMONSTRATED
DISBONDS UNDER TRANSVERSE SHEAR:	APPROPRIATE STRESS ANALYSIS METHODS AND STRAIN ENERGY RELEASE RATE CONCEPTS CAN BE USED TO ASSESS CRITICALITY. SEMI-EMPIRICAL GROWTH LAWS CAN BE USED TO MODEL GROWTH UNDER CYCLIC LOADS
DISBONDS UNDER COMPRESSION:	ISOLATED DISBONDS ARE CRITICAL WHEN BUCKLING CAN OCCUR UNDER STATIC LOAD. SLOW GROWTH UNDER FATIGUE IS POSSIBLE NEAR GEOMETRIC DISCONTINUITIES LIKE PLY DROPS BEFORE FINAL FAILURE

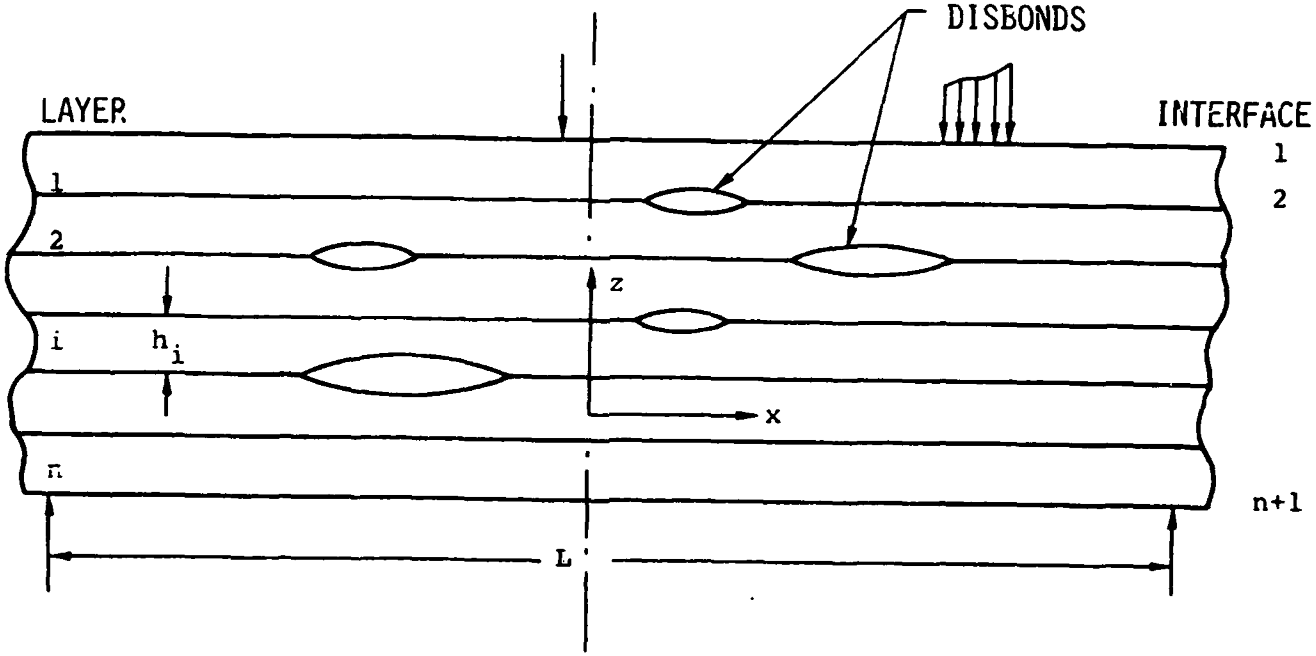
AD-A130 750 PROCEEDINGS OF THE ANNUAL MECHANICS OF COMPOSITES REVIEW (8TH) HELD AT WR..(U) AIR FORCE WRIGHT AERONAUTICAL LABS WRIGHT-PATTERSON AFB OH L A WILSON 2/5
UNCLASSIFIED APR 83 AFWAL-TR-83-4005 F/G 11/4 NL





MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

DISBONDS IN BEAM TYPE STRUCTURES



STRESS ANALYSIS FOR ISOLATED OR MULTIPLE DISBONDS IN BEAMS OR PLATES UNDER 2-D PLANE STRESS OR QUASI-3D PLANE STRAIN CONDITIONS

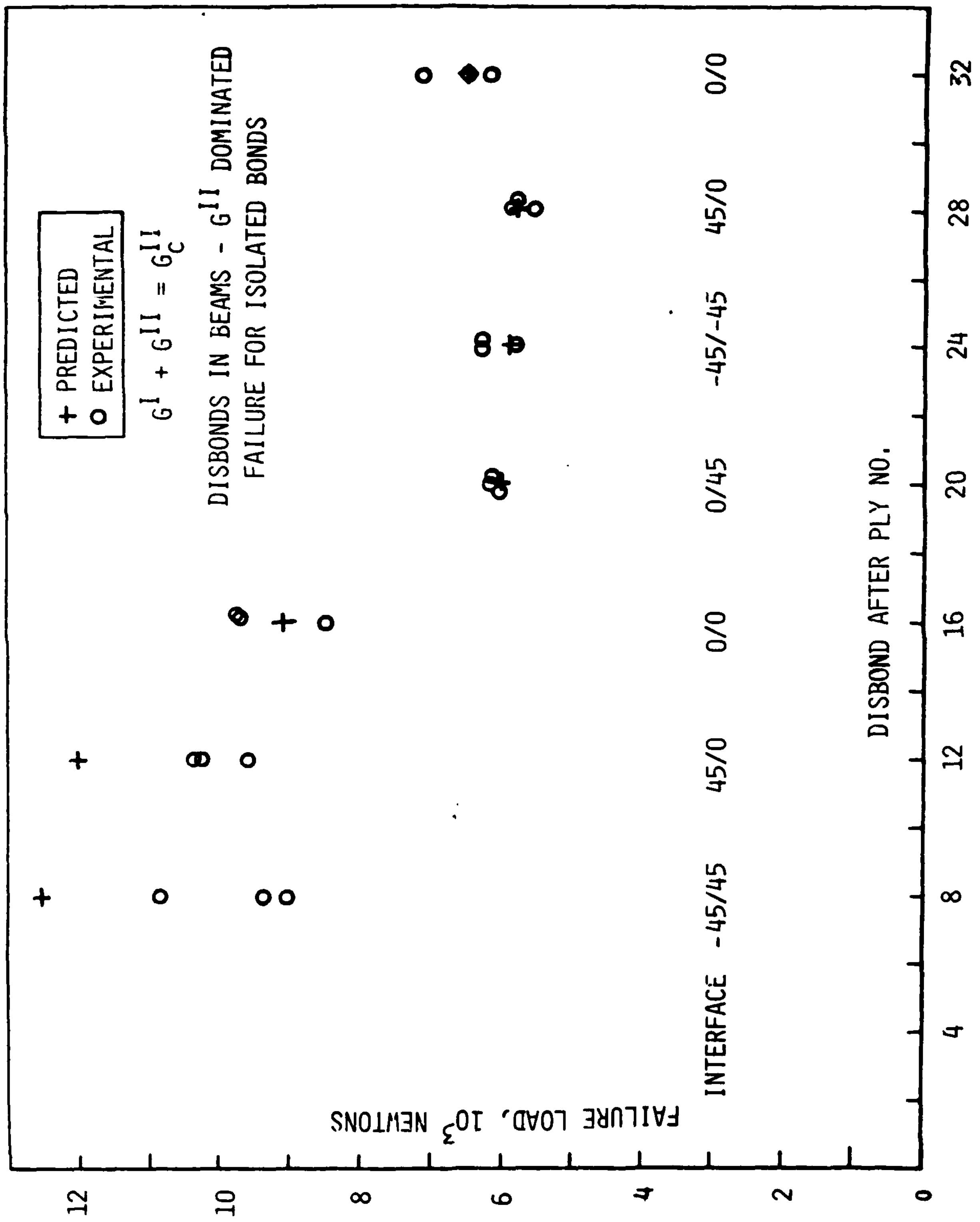
1. ELASTICITY SOLJTION FOR EACH LAYER VIA FOURIER SERIES OR FOURIER TRANSFORMS

2. OBTAIN SINGULAR INTEGRAL EQUATIONS IN TERMS OF DISPLACEMENT DISCONTINUITIES ACROSS EACH DISBOND

3. EXTRACT CHARACTERISTIC STRESS SINGULARITIES

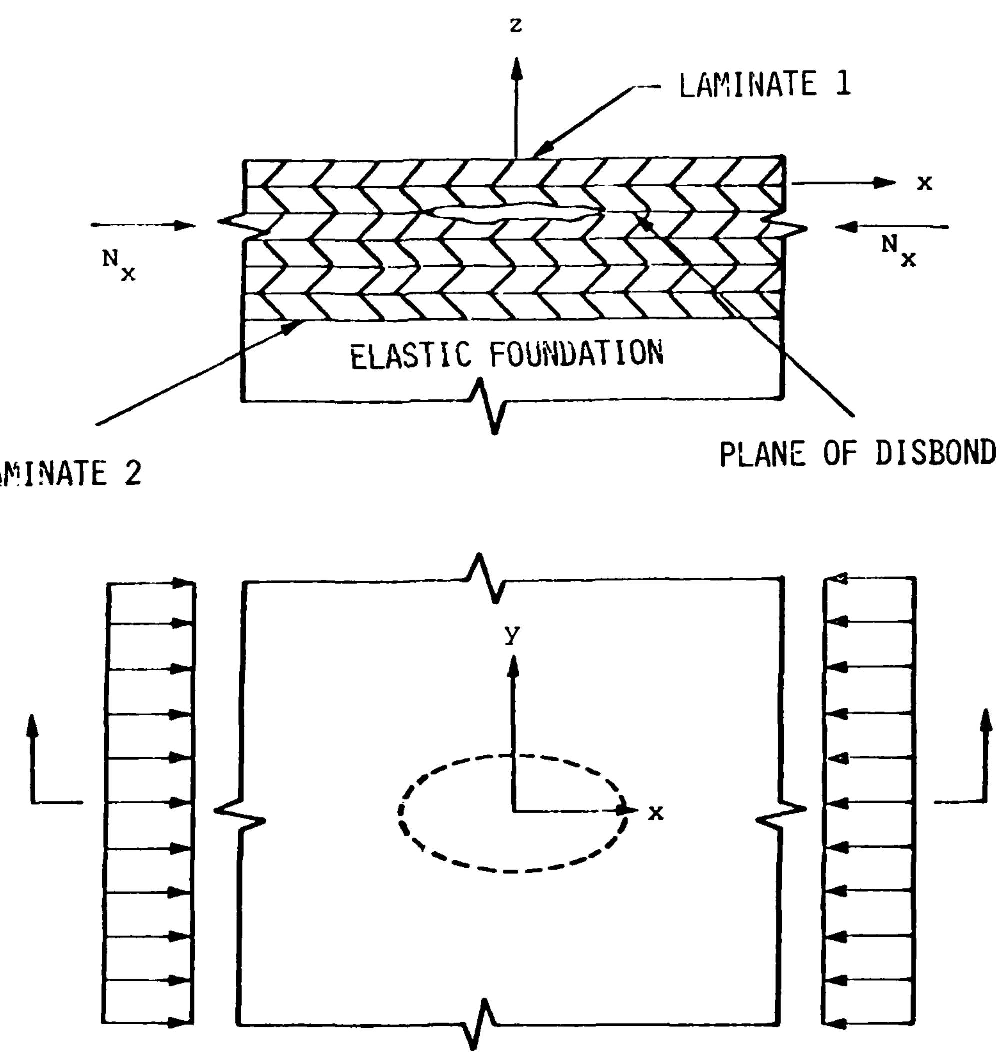
4. NUMERICAL SOLUTION OF THE SET OF EQUATIONS

5. DETERMINE STRENGTH OF SINGULARITIES AND ENERGY RELEASE RATES AT EACH TIP



+ PREDICTED
o EXPERIMENTAL
$G^{I} + G^{II} = G_{C}^{II}$
DISBONDS IN BEAMS - G^{II} DOMINATED
FAILURE FOR ISOLATED BONDS
INTERFACE -45/45
45/0
0/0
0/45
-45/-45
45/0
0/0
DISBOND AFTER PLY NO.
4
8
12
16
20
24
28
32
FAILURE LOAD, 10^3 NEWTONS
0
2
4
6
8
10
12

DISBOND IN A PLATE



STRESS ANALYSIS - DISBOND WITH ELLIPTIC PLAN FORM BETWEEN TWO LAMINATED PLATES

1. USE MODIFIED LAMINATED PLATE THEORY INCLUDING

   (i) EFFECTS OF TRANSVERSE SHEAR DEFORMATION

   (ii) TRACTIONS ACTING ON THE PLANE OF THE DISBOND

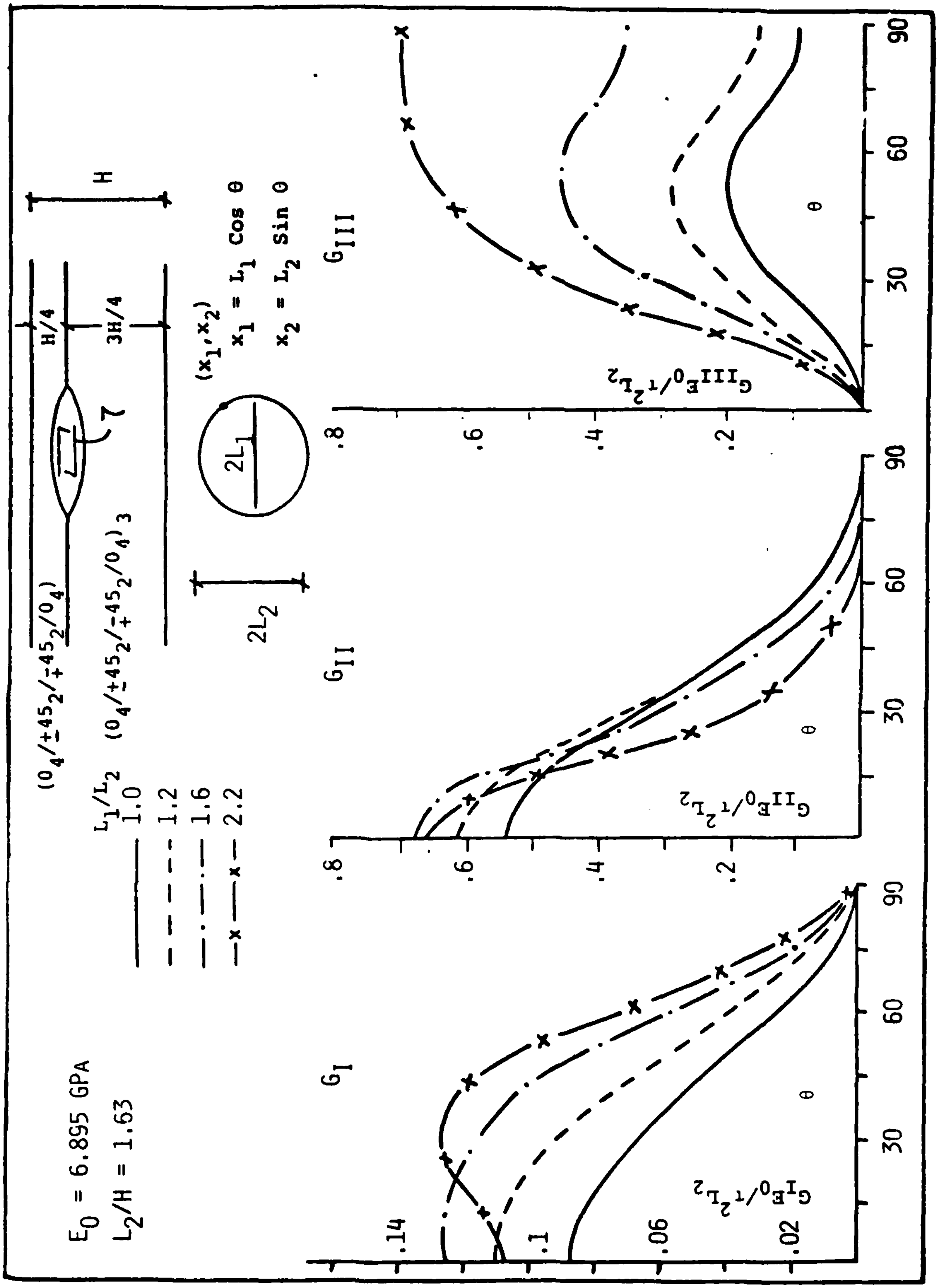
   (iii) PRESTRESS AND ELASTIC FOUNDATION

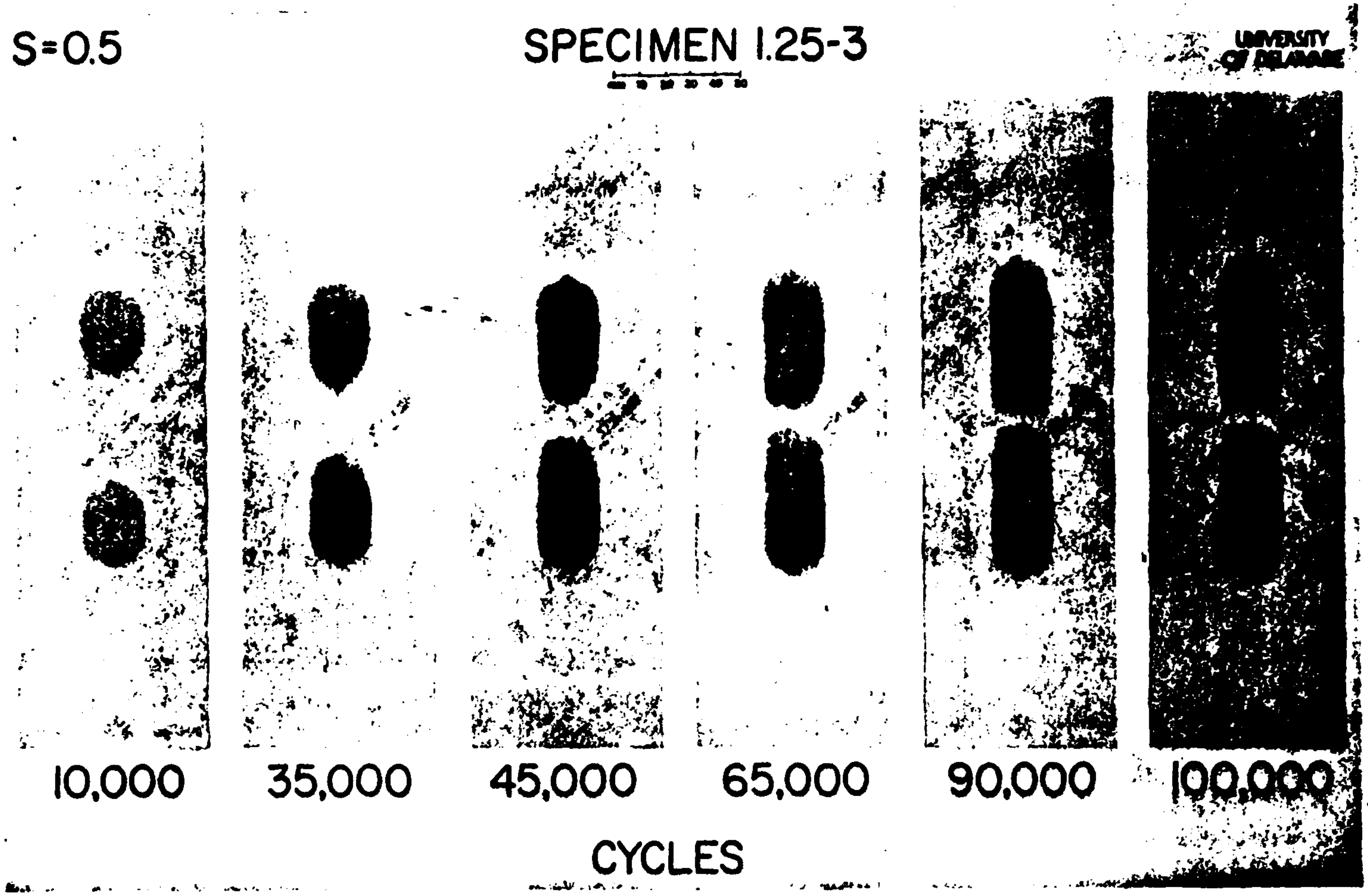
2. USE TRANSFORM METHODS TO REDUCE THE SYSTEM OF DIFFERENTIAL EQUATIONS TO THREE COUPLED INTEGRAL EQUATIONS IN TERMS OF DISPLACEMENT DISCONTINUITIES ACROSS THE DISBOND

3. USE NUMERICAL INTEGRATION SCHEMES AND DISCRETIZATION TECHNIQUES TO OBTAIN A SYSTEM OF ALGEBRAIC EQUATIONS

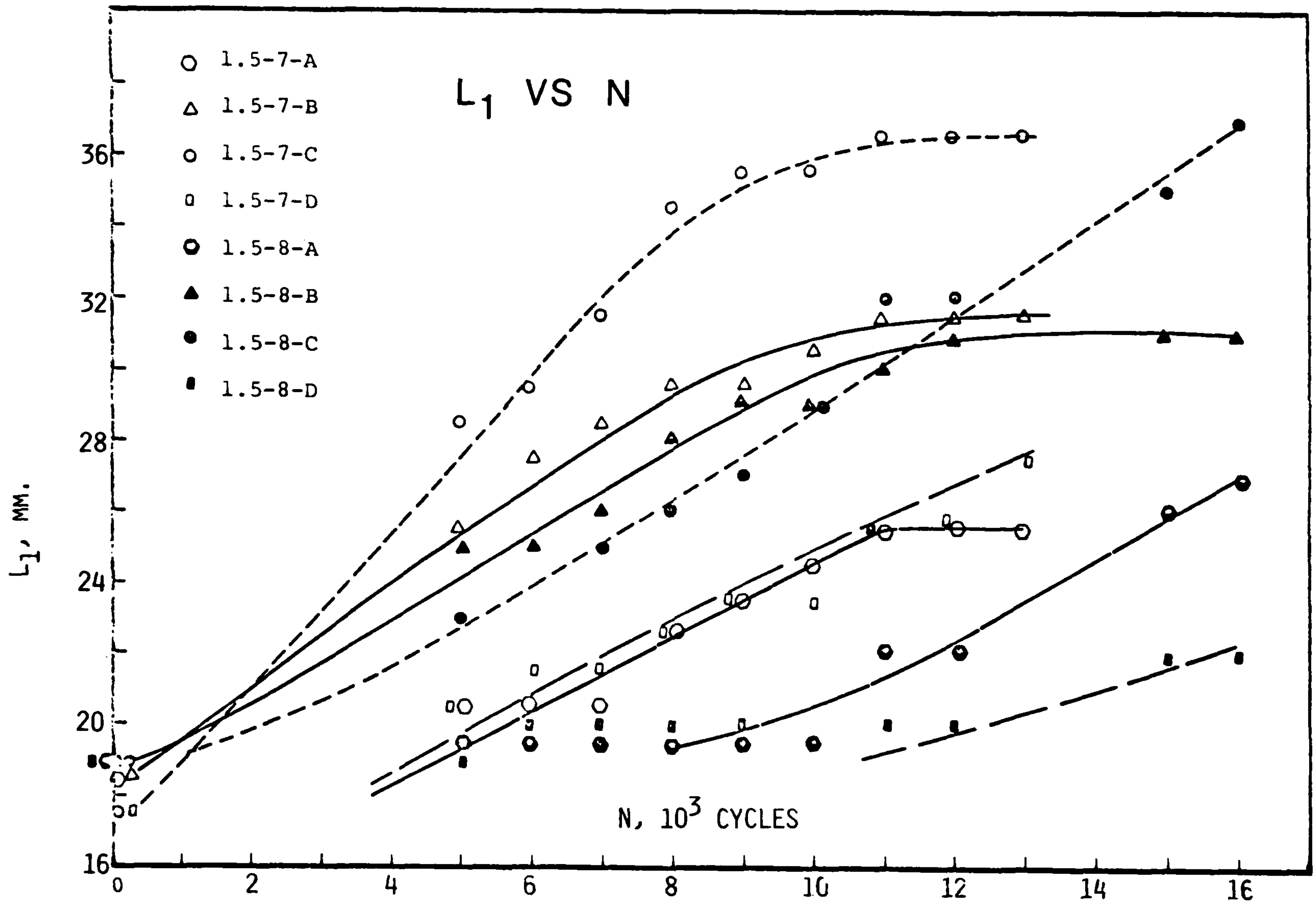
4. CALCULATE STRAIN ENERGY RELEASE RATES (MODE II AND MODE III CONTRIBUTIONS) FROM INTERACTIVE LINE FORCES AT THE PERIPHERY OF THE DISBOND. CHOOSE SEMI-EMPIRICAL GROWTH LAW FOR CYCLIC LOADING.

5. EQUATE DETERMINANT OF THE DISCRETIZED SYSTEM TO ZERO TO OBTAIN CRITICAL LOAD FOR BUCKLING FAILURE.

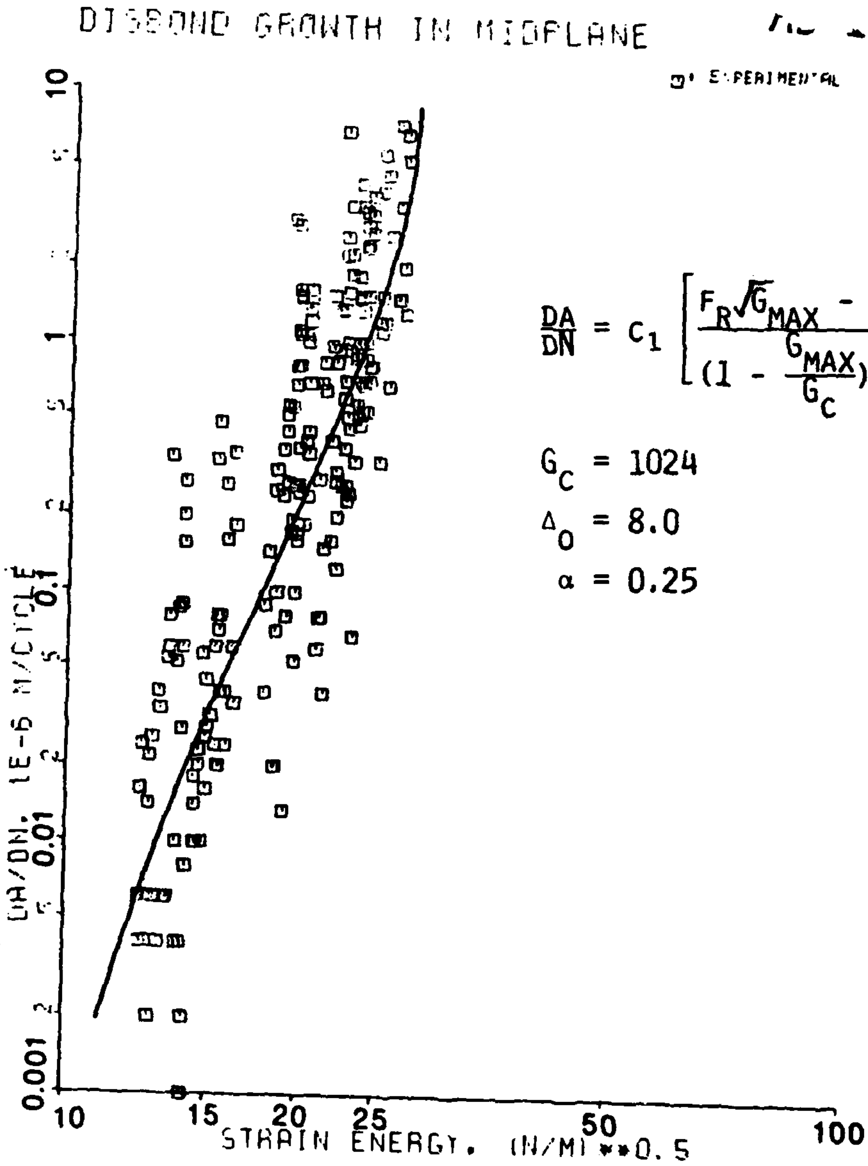

E0 = 6.895 GPA
L2/H = 1.63
(04/±452/∓452/04)
(04/±452/∓452/04)3
H/4
3H/4
H
2L1
2L2
(x1,x2)
x1 = L1 Cos θ
x2 = L2 Sin θ
L1/L2
1.0
1.2
1.6
2.2
GI
GII
GIII
GIE0/τ2L2
GIIE0/τ2L2
GIIIE0/τ2L2
θ
.14
.1
.06
.02
.8
.6
.4
.2
30
60
90



ULTRASONIC C-SCAN-FATIGUE SPECIMEN



L_1 VS. N FOR SAMPLES 1.5-7 AND 1.5-8, S=0.6



$$\frac{DA}{DN} = C_1 \left[\frac{F_R\sqrt{G}_{MAX} - \Delta_O}{(1 - \frac{G_{MAX}}{G_C})^{\alpha}} \right]^{C_2}$$

$G_C = 1024$

$\Delta_O = 8.0$

$\alpha = 0.25$

CURVE FIT TO FATIGUE DATA

AD P001250

SUPERPOSITION METHOD FOR ANALYSIS OF FREE EDGE STRESSES

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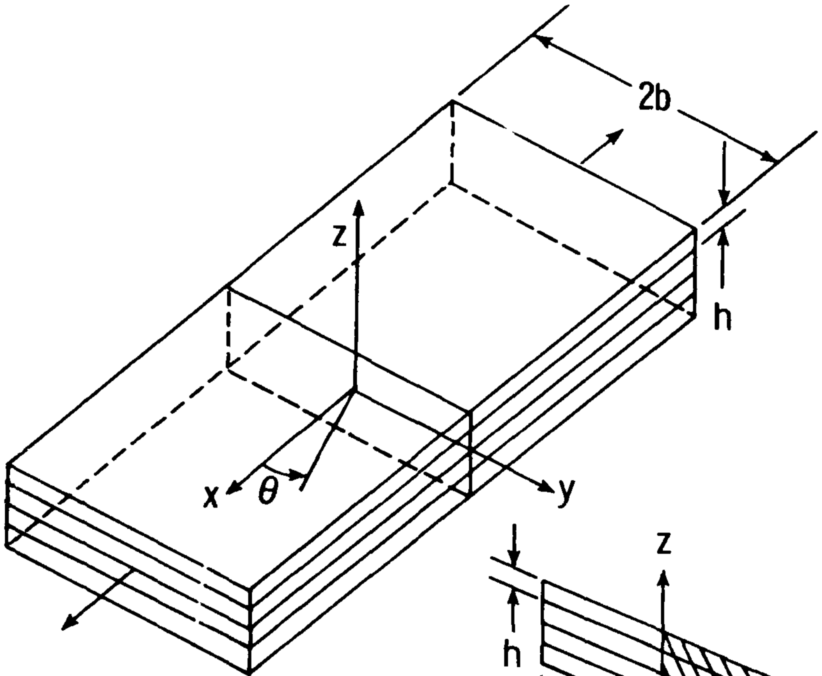
OBJECTIVES

- TO SIMPLIFY FREE EDGE STRESS ANALYSIS BY USING SUPERPOSITION PRINCIPLES
- TO DEVELOP A 2-D ANALYSIS FOR CALCULATION OF FREE EDGE STRESSES IN COMPOSITE LAMINATES
- TO EVALUATE THE ACCURACY OF THE 2-D ANALYSIS FOR MECHANICAL, THERMAL AND HYGROSCOPIC LOADS

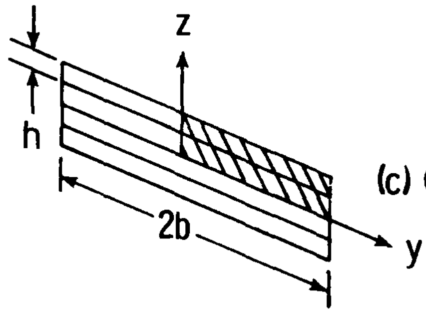
CONCLUSIONS

- SUPERPOSITION TECHNIQUE SIMPLIFIES FREE EDGE STRESS ANALYSIS OF LAMINATED COMPOSITE PLATES
- SUPERPOSITION TECHNIQUE CLARIFIES RELATIONSHIP BETWEEN CLT AND RIGOROUS ANALYSIS OF LAMINATED COMPOSITE PLATES
- SUPERPOSITION TECHNIQUE WORKS WELL FOR MECHANICAL, THERMAL AND HYGROSCOPIC LOADS
- FREE EDGE STRESS PROBLEM IS A PLANE-STRAIN PROBLEM ($\varepsilon_x = 0$) IF NO PLIES HAVE EXTENSION-SHEAR COUPLING. THEREFORE, A 2-D ANALYSIS IS SUFFICIENT
- THE INTERLAMINAR NORMAL STRESS (σ_z) CAN BE CALCULATED ACCURATLEY FOR QI LAMINATES USING A 2-D ANALYSIS BY ASSUMING $\varepsilon_{xy} = 0$ FOR ALL PLIES

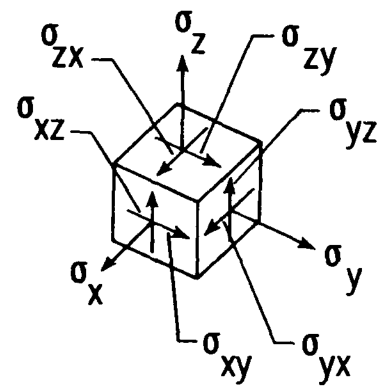
THE EDGE-STRESS PROBLEM OF A COMPOSITE LAMINATE



(a) FOUR-PLY LAMINATE



(b) AN x = CONSTANT PLANE



(d) 3-D STRESS COMPONENTS

$u = \varepsilon_0 x + U(y, z)$

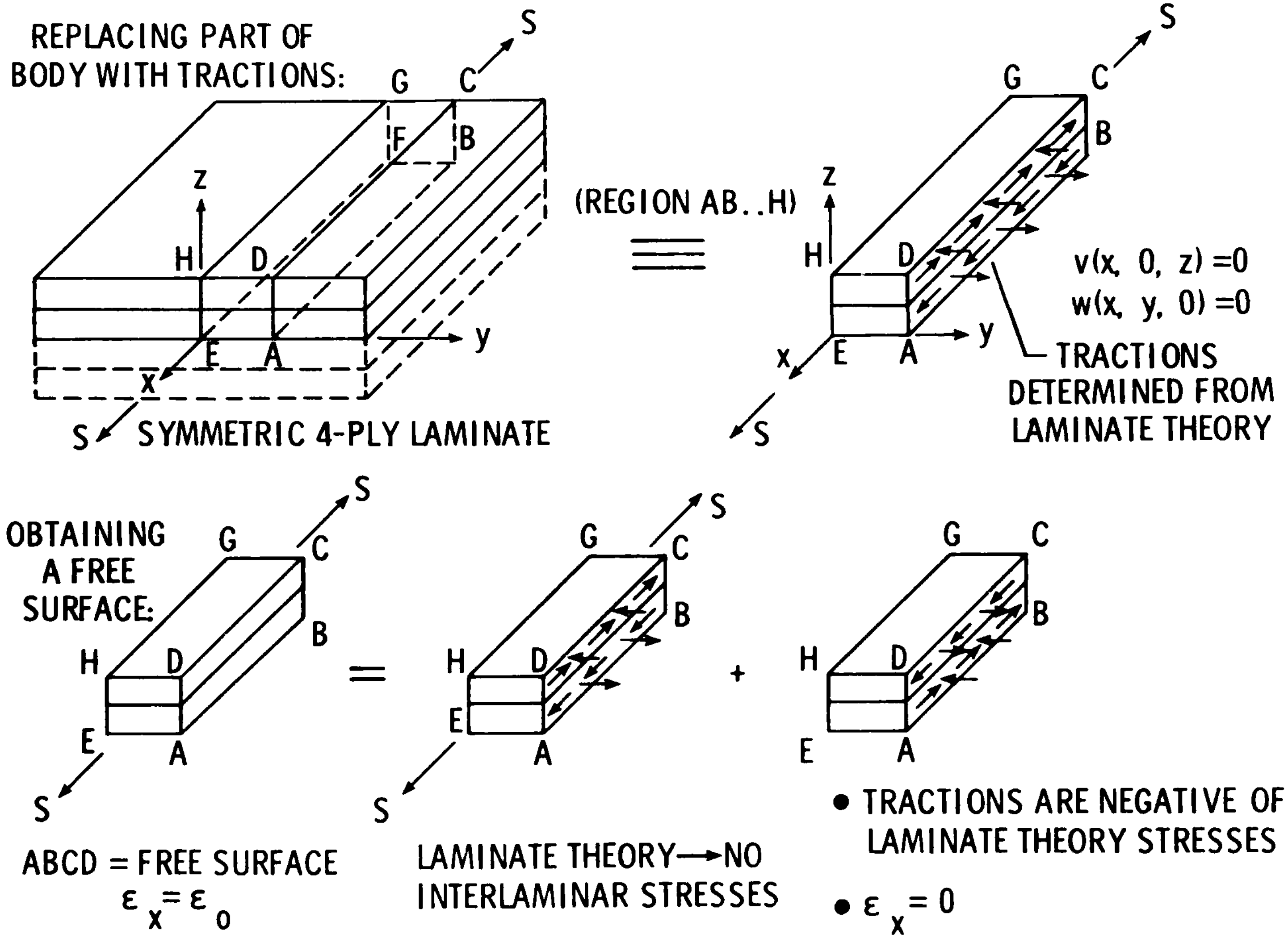
$v = V(y, z)$

$w = W(y, z)$

(c) QUASI 3-D APPROXIMATION

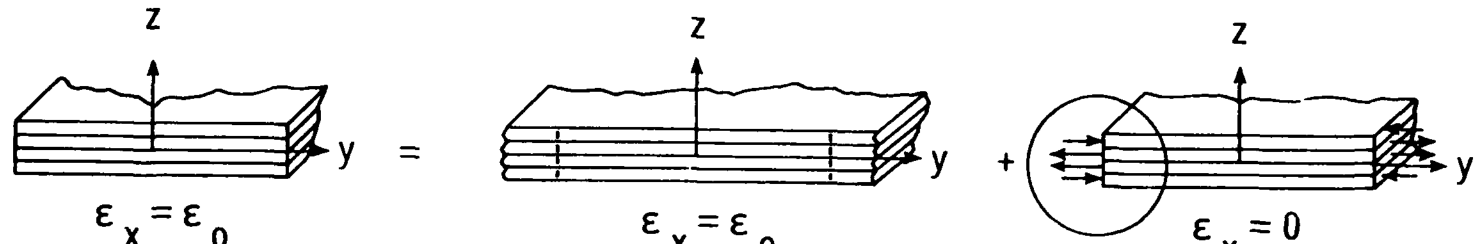
$\varepsilon_0 = 0.001$

OBTAINING A FREE SURFACE BY SUPERPOSITION



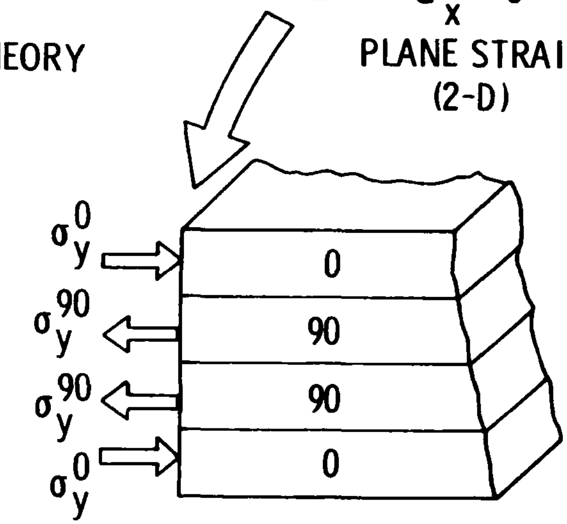
USE OF SUPERPOSITION FOR $[0/90]_s$ LAMINATE

MECHANICAL LOAD



LAMINATE THEORY STRESSES

PLY	σ_y, MPa	σ_{xy}, MPa
0^0	2.474	0
90^0	-2.474	0



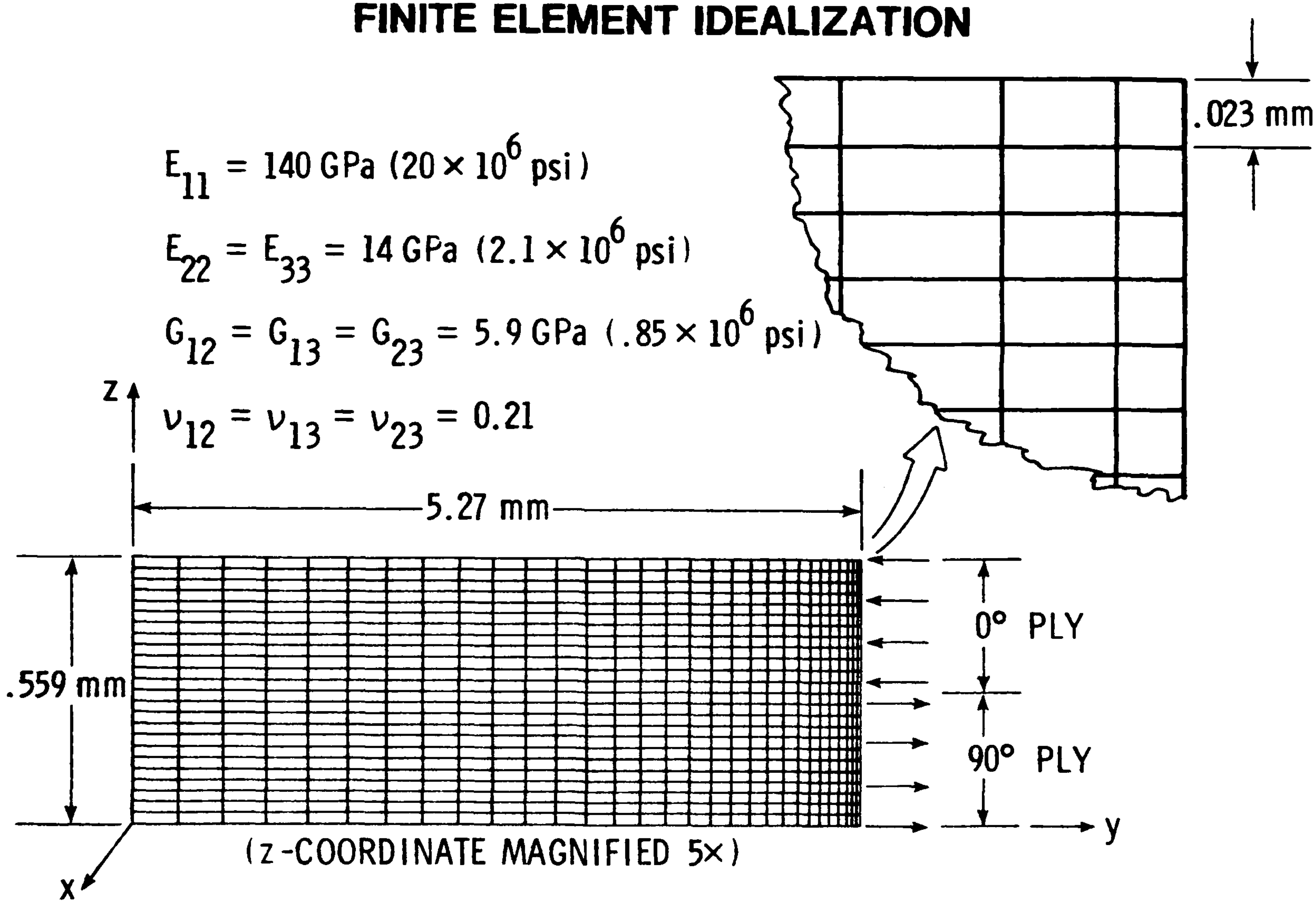
FINITE ELEMENT IDEALIZATION

$E_{11} = 140 \text{ GPa } (20 \times 10^6 \text{ psi})$

$E_{22} = E_{33} = 14 \text{ GPa } (2.1 \times 10^6 \text{ psi})$

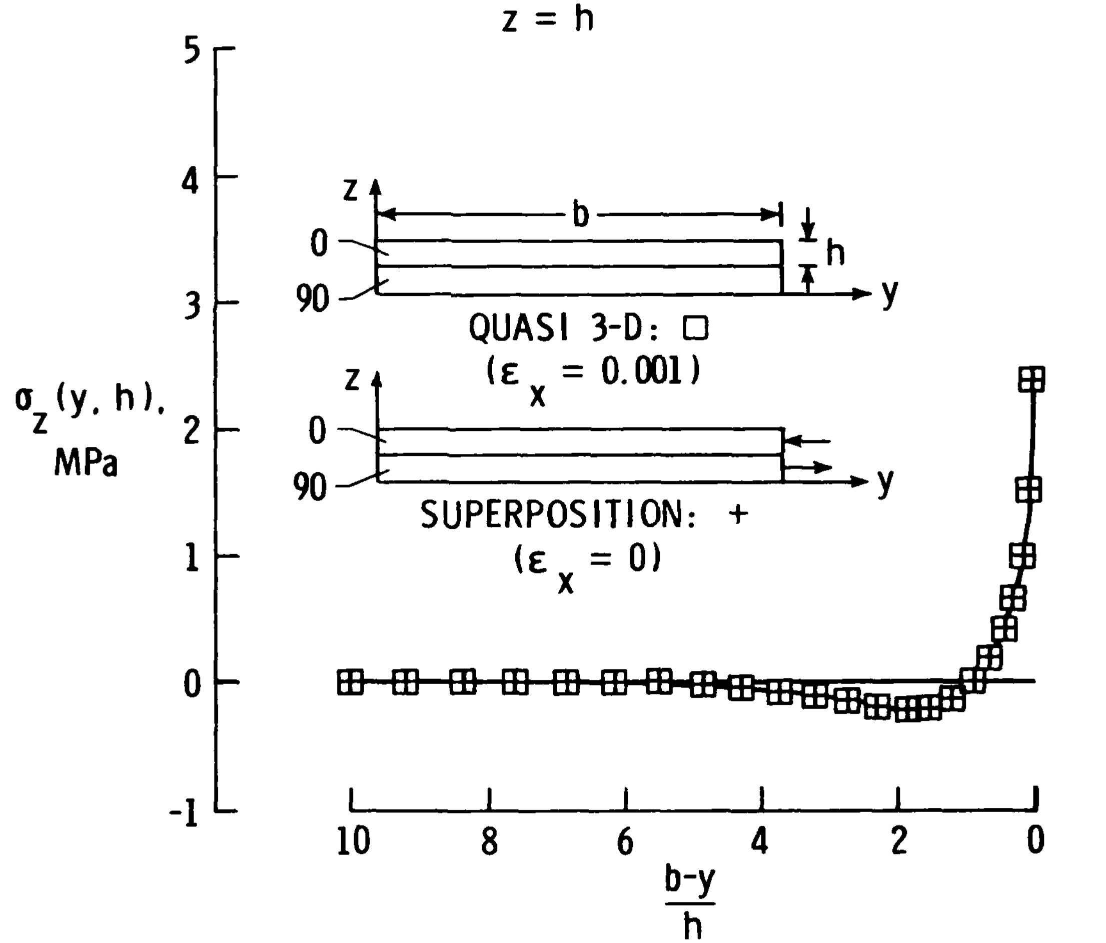
$G_{12} = G_{13} = G_{23} = 5.9 \text{ GPa } (.85 \times 10^6 \text{ psi})$

$\nu_{12} = \nu_{13} = \nu_{23} = 0.21$



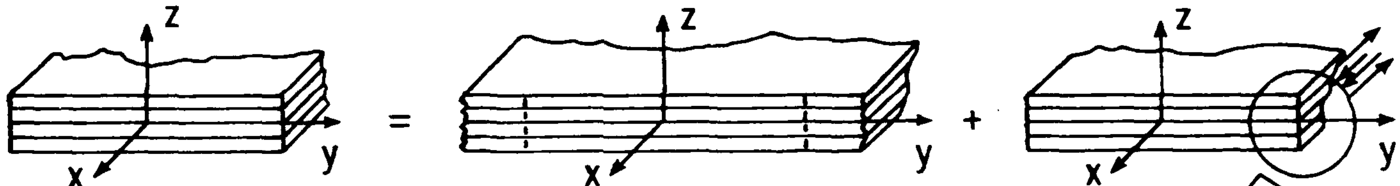
775 NODES
720 ELEMENTS

INTERLAMINAR NORMAL STRESS IN $[0/90]_s$ LAMINATE



USE OF SUPERPOSITION FOR $[\pm 45]_s$ LAMINATE

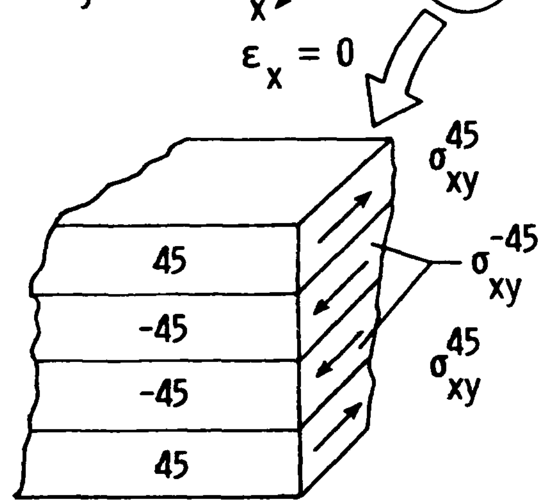
MECHANICAL LOAD



$\varepsilon_x = \varepsilon_0$ QUASI 3-D

$\varepsilon_x = \varepsilon_0$ LAMINATE THEORY

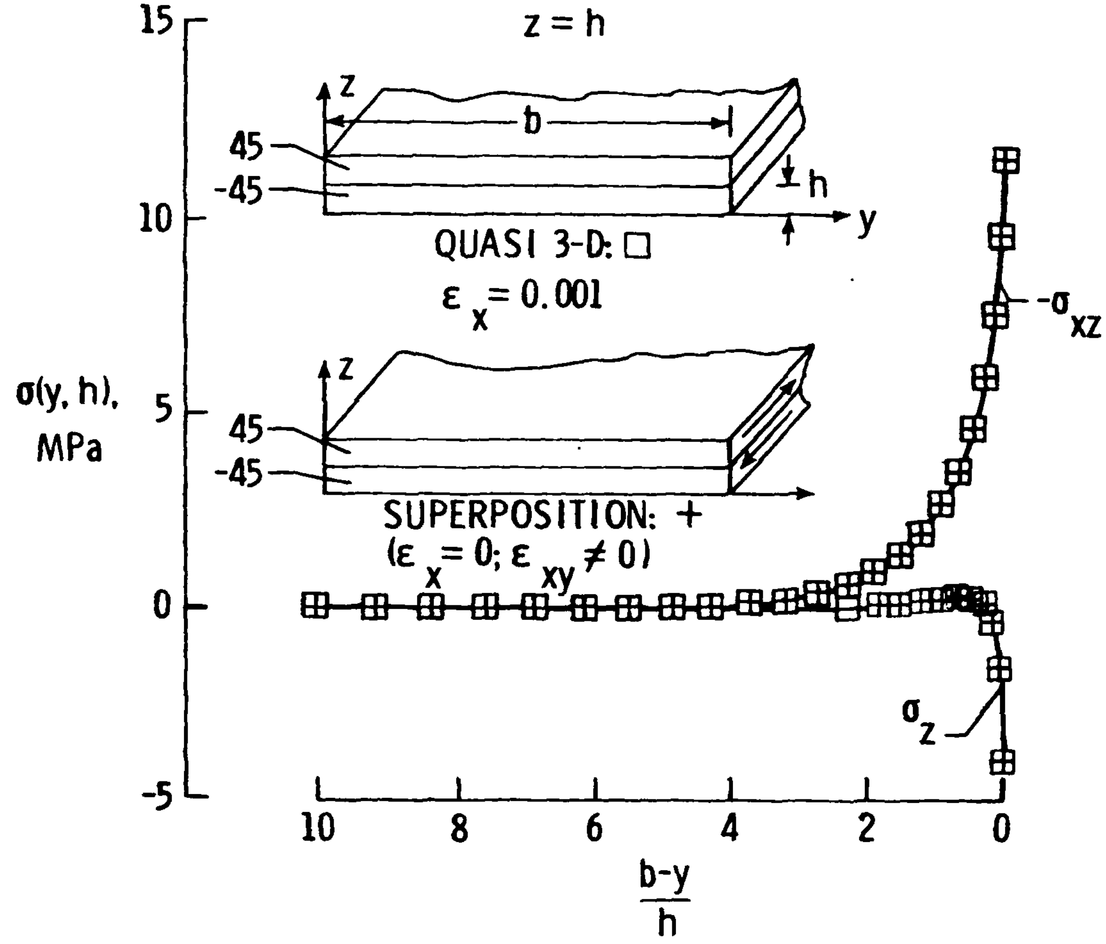
$\varepsilon_x = 0$



LAMINATE THEORY STRESSES

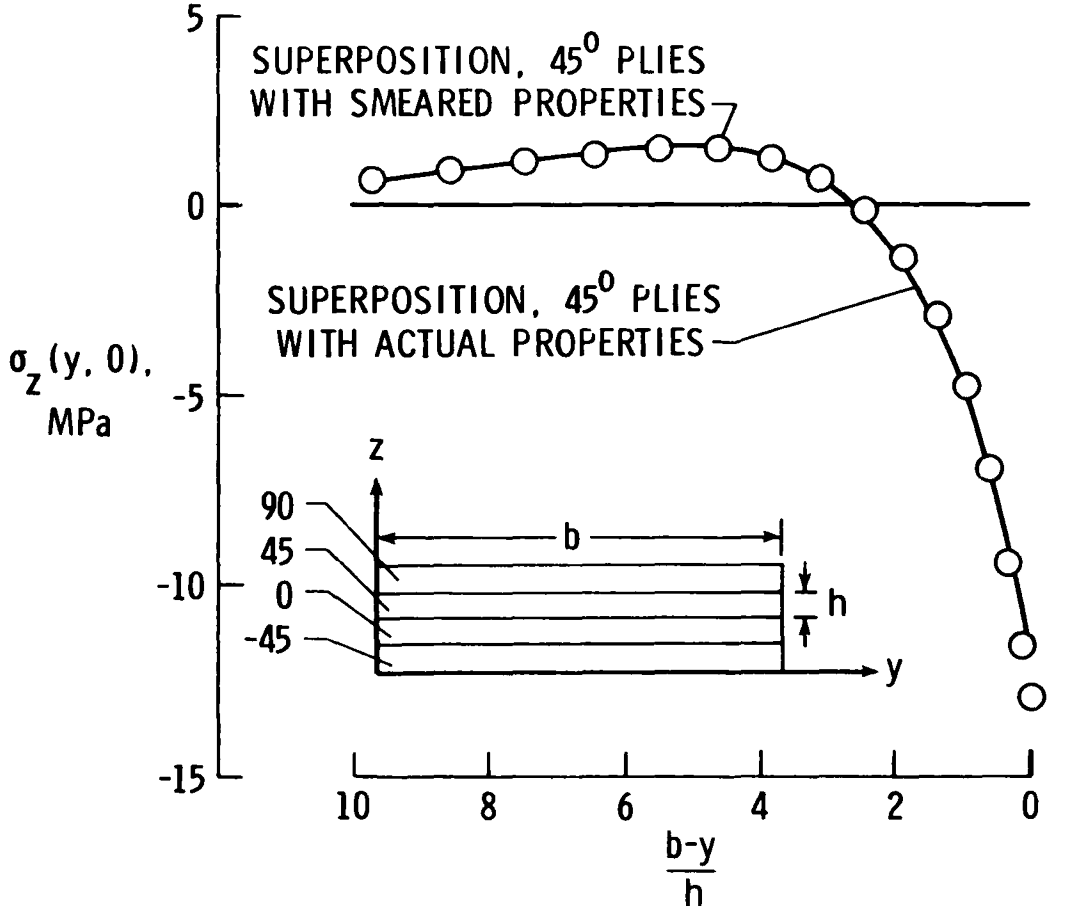
PLY	σ_y, MPa	σ_{xy}, MPa
45^0	0	7.957
-45^0	0	-7.957

INTERLAMINAR NORMAL AND SHEAR STRESSES IN $[\pm 45]_s$ LAMINATE



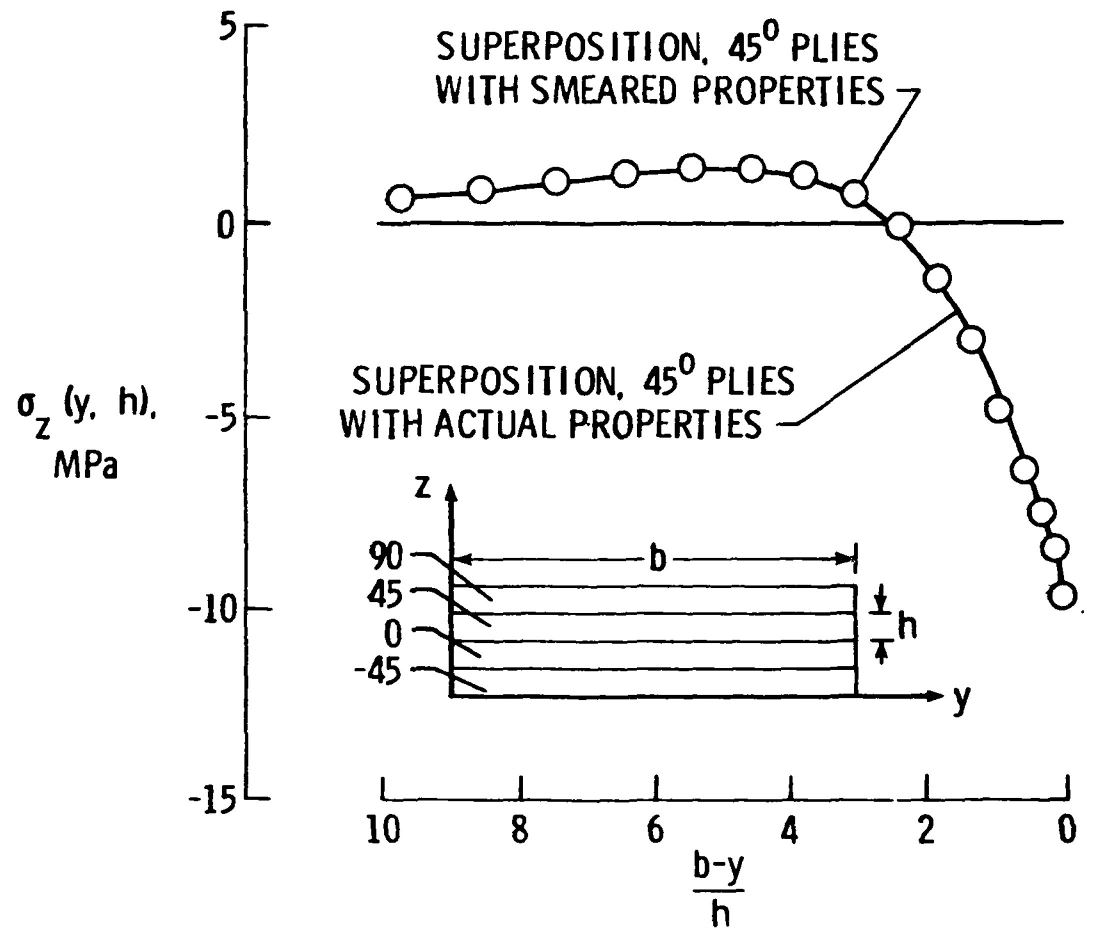
INTERLAMINAR NORMAL STRESS IN $[90/45/0/-45]_s$ LAMINATE ALONG MIDPLANE

$\varepsilon_x = 0.001$



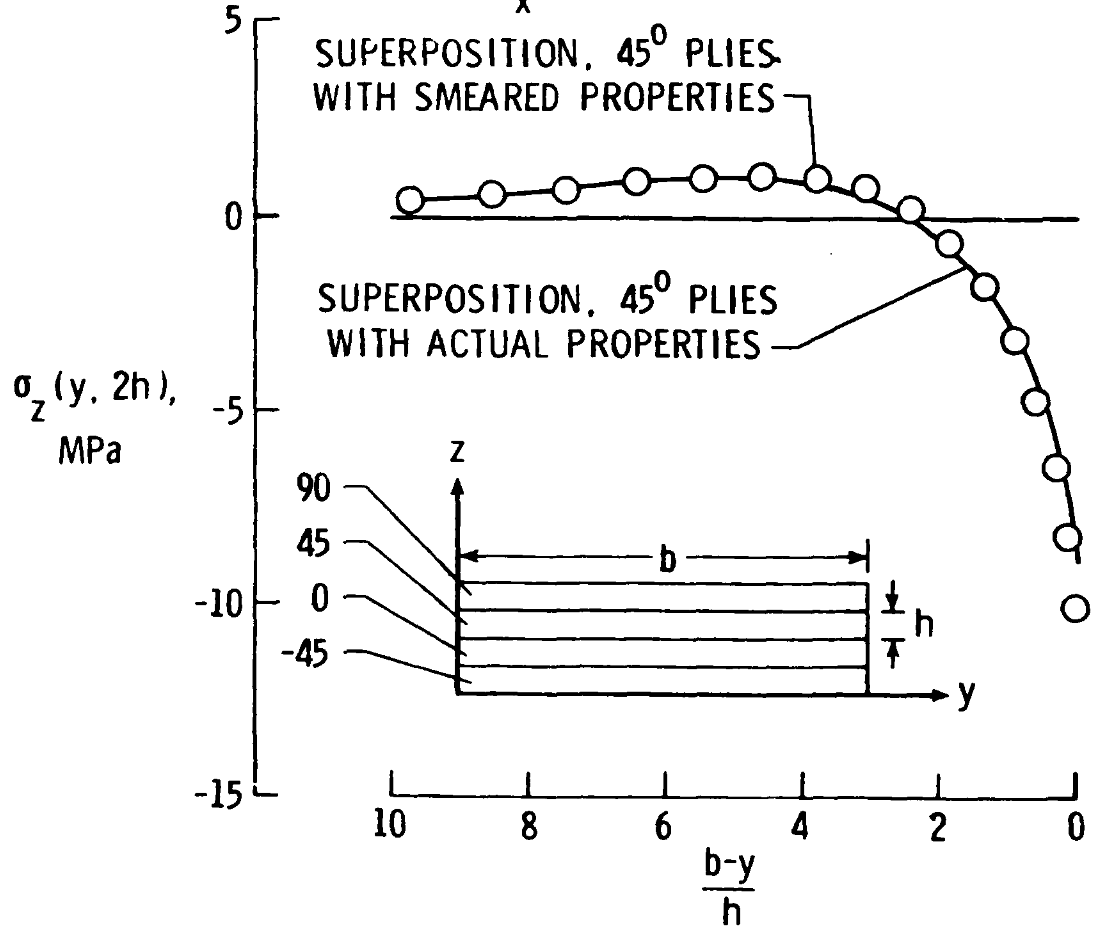
INTERLAMINAR NORMAL STRESS IN $[90/45/0/-45]_s$ LAMINATE ALONG $z = h$

$\varepsilon_x = 0.001$

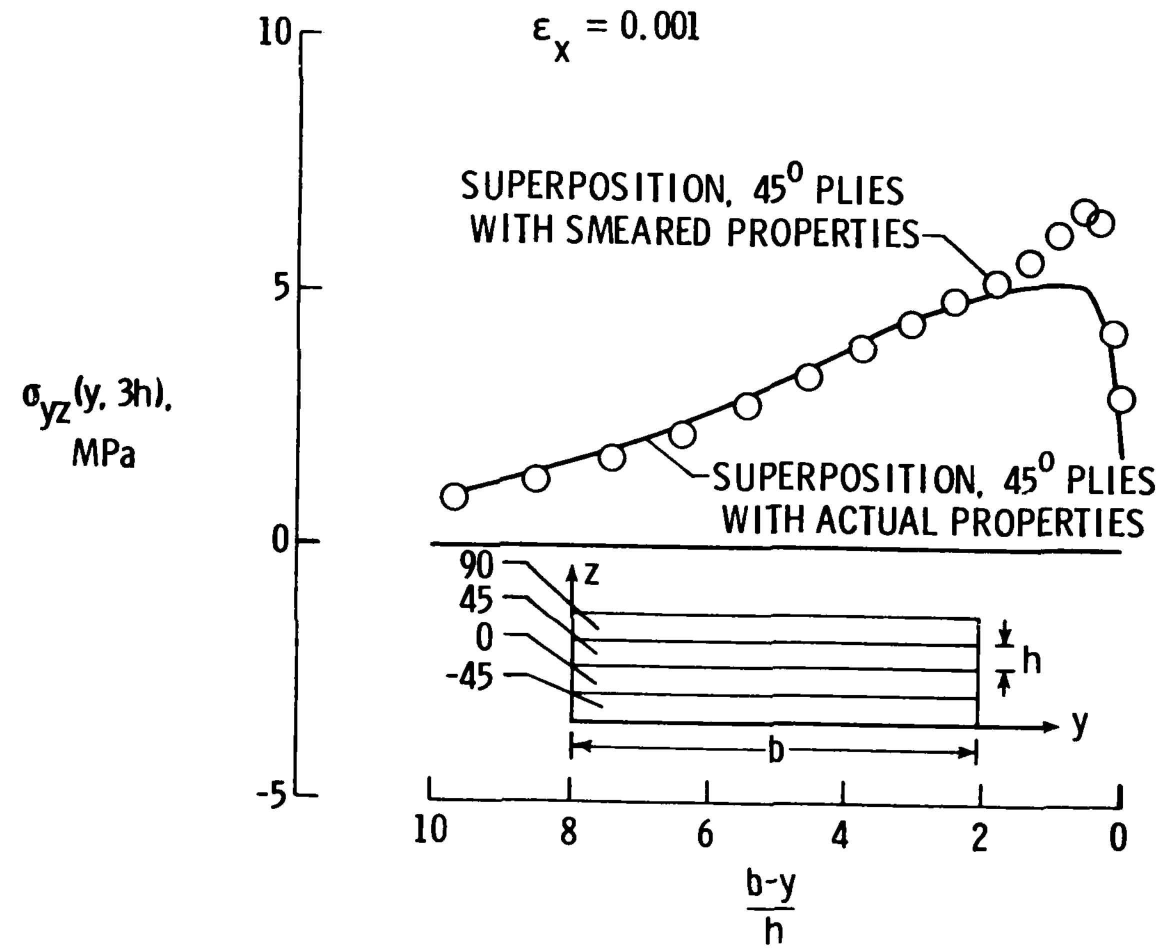


INTERLAMINAR NORMAL STRESS IN $[90/45/0/-45]_s$ LAMINATE ALONG $z = 2h$

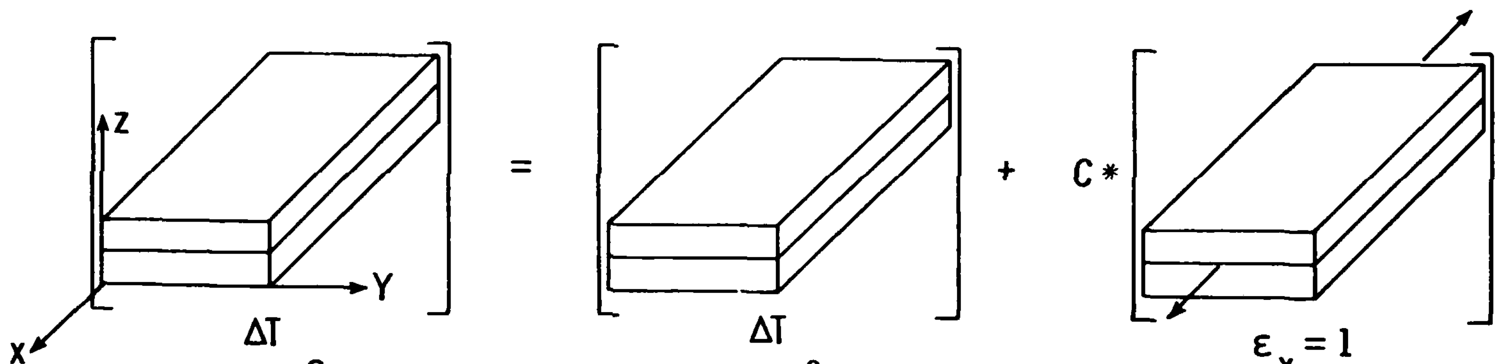
$\varepsilon_x = 0.001$



INTERLAMINAR SHEAR STRESS IN $[90/45/0/-45]_s$ LAMINATE ALONG $z = 3h$



ANALYSIS OF INTERLAMINAR STRESSES DUE TO THERMAL LOADS



ΔT, $\varepsilon_x = ?$, $F_x = \int \sigma_x dA = 0$ $\quad = \quad$ ΔT, $\varepsilon_x = 0$, $F_{x1} = \int \sigma_x dA$ $\quad + \; C * \quad$ $\varepsilon_x = 1$, $F_{x2} = \int \sigma_x dA$

"C" CHOSEN SUCH THAT $F_x = 0 = F_{x1} + C * F_{x2}$

$$C = \frac{-F_{x1}}{F_{x2}} \approx \alpha_x \Delta T \Big|_{CLT}$$

MECHANICAL VERSUS THERMAL LOADS

MECHANICAL LOADS

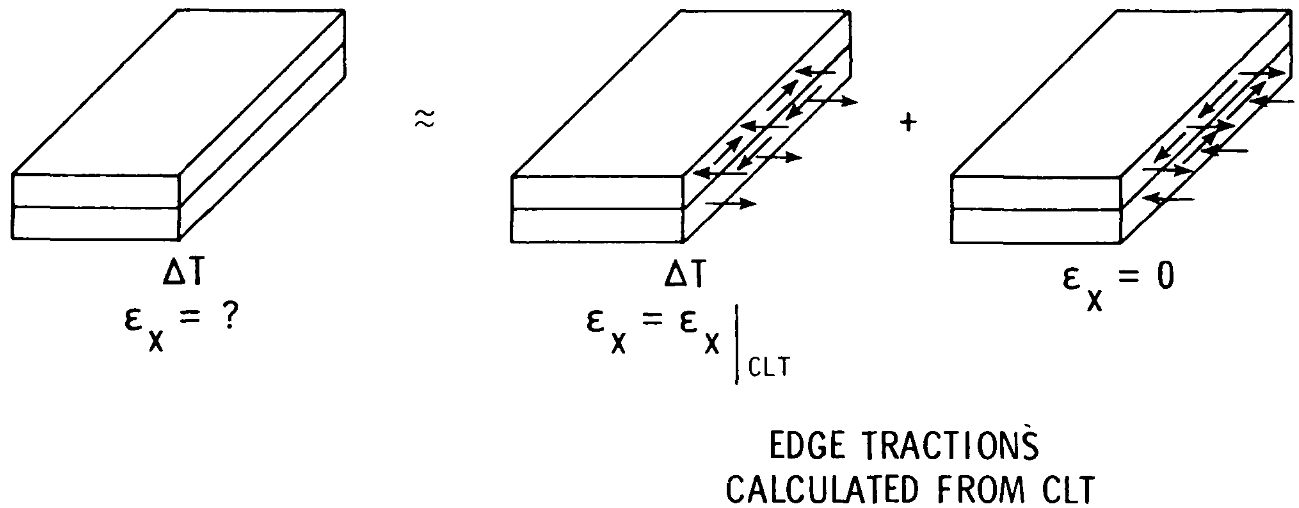
- SPECIFY ε_x
- CLT IN-PLANE STRESSES RECOVERED WHEN INTERLAMINAR STRESSES → 0
- $\int \sigma_x dA \neq F_x\big|_{CLT}$

THERMAL LOADS

- SPECIFY ΔT
- CONSTRAINT $\int \sigma_x dA = 0$
- CLT IN-PLANE STRESSES NOT RECOVERED WHEN INTERLAMINAR STRESSES → 0
- $\varepsilon_x \neq \varepsilon_x\big|_{CLT}$

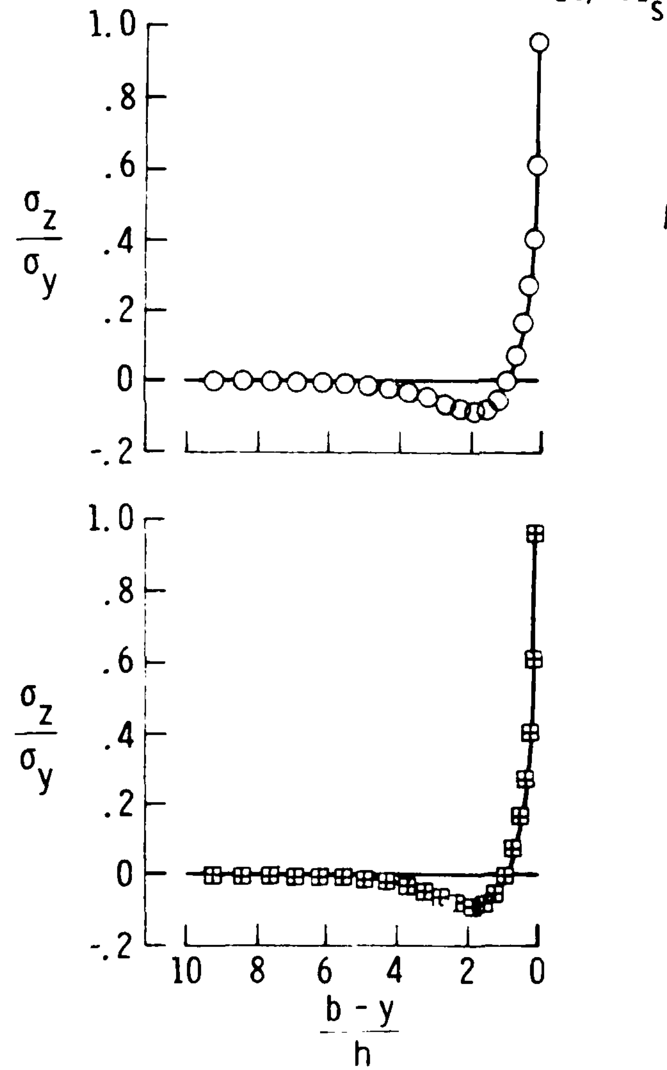
- IN-PLANE STRESSES AND STRAINS FROM LAMINATE THEORY VERY ACCURATE EXCEPT FOR VERY NARROW LAMINATES
  - GREATEST DIFFERENCE IS FOR ε_x DUE TO ΔT

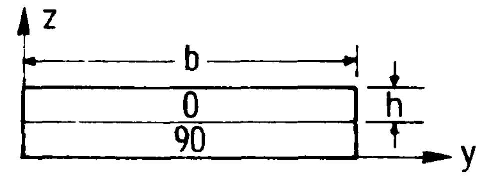
APPROXIMATE SUPERPOSITION TECHNIQUE FOR THERMAL ANALYSIS



INTERLAMINAR NORMAL STRESS DUE TO MECHANICAL AND THERMAL LOADS

$[0/90]_s$





MECHANICAL LOADING, $\varepsilon_x = 0.001$
LAMINATE THEORY STRESSES

PLY	$\sigma_{y'}$, MPa	$\sigma_{xy'}$, MPa
0°	2.474	0
90°	-2.474	0

THERMAL LOADING, 1°F
LAMINATE THEORY STRESSES

PLY	$\sigma_{y'}$, MPa	$\sigma_{xy'}$, MPa
0°	-199.09 (-199.14)*	0 (0)
90°	199.09 (199.17)*	0 (0)
$\varepsilon_x = 1.9469 * 10^{-6}$ $(1.9264 * 10^{-6})$*		

* $\int \sigma_x \, da = 0$

MECHANICAL VERSUS THERMAL LOADS

MECHANICAL LOADS

- SPECIFY ε_x
- CLT IN-PLANE STRESSES RECOVERED WHEN INTERLAMINAR STRESSES $\rightarrow 0$
- $\int \sigma_x \, dA \neq F_x \Big|_{CLT}$

THERMAL LOADS

- SPECIFY ΔT
- CONSTRAINT $\int \sigma_x \, dA = 0$
- CLT IN-PLANE STRESSES NOT RECOVERED WHEN INTERLAMINAR STRESSES $\rightarrow 0$
- $\varepsilon_x \neq \varepsilon_x \Big|_{CLT}$

- IN-PLANE STRESSES AND STRAINS FROM LAMINATE THEORY VERY ACCURATE EXCEPT FOR VERY NARROW LAMINATES
    - GREATEST DIFFERENCE IS FOR ε_x DUE TO ΔT

AD P001251

MECHANICS OF DELAMINATION UNDER COMPRESSIVE LOADS

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OBJECTIVES:

1. TO STUDY THE DELAMINATION MECHANISMS IN GRAPHITE-EPOXY LAMINATES SUBJECTED TO COMPRESSIVE STATIC AND FATIGUE LOADS;
2. TO FORMULATE ANALYTICAL MODELS FOR THE INITIATION AND THE GROWTH BEHAVIOR OF DELAMINATION CRACKING UNDER BOTH STATIC AND FATIGUE (COMPRESSIVE) LOADS.

CONCLUSIONS:

1. WHEN STRUCTURAL BUCKLING IS PREVENTED, A LAMINATE SUBJECTED TO UNIAXIAL COMPRESSION MAY DEVELOP FREE EDGE DELAMINATION IN THE SAME MANNER AS OBSERVED IN TENSION;

2. THE DRIVING FORCE FOR DELAMINATION IS DERIVED FROM THE CONCENTRATION OF FREE EDGE INTERLAMINAR STRESSES;

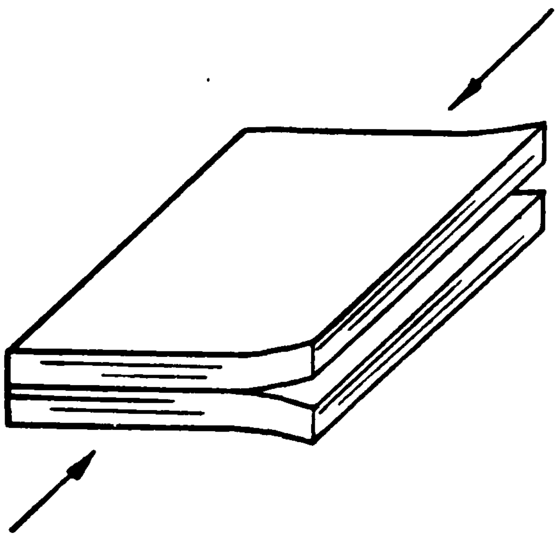
3. A MODEL BASED ON FRACTURE MECHANICS (STRAIN ENERGY RELEASE RATE) IS SHOWN TO PREDICT THE INITIATION OF DELAMINATION UNDER STATIC (COMPRESSIVE) LOAD;

4. EXPERIMENTS SUGGEST THAT THE QUANTITY OF STRAIN ENERGY RELEASE RATE ALSO PLAYS A DOMINANT ROLE IN THE GROWTH OF DELAMINATION UNDER FATIGUE (COMPRESSIVE) LOAD;

5. POST-DELAMINATION LAMINATE FAILURE IS GENERALLY ASSOCIATED WITH LOCAL LAMINA BUCKLING; HENCE, DELAMINATION UNDER COMPRESSION IS OFTEN CATASTROPHIC;

6. THE DELAMINATION FRONT IS GENERALLY A 2-DIMENSIONAL CONTOUR IN THE DELAMINATION PLANE; THE EXPANSION OF THE CONTOUR IS DIRECTIONALLY DEPENDENT. A 3-DIMENSIONAL (FINITE ELEMENT) SIMULATION IS REQUIRED TO CALCULATE THE QUANTITY OF THE STRAIN ENERGY RELEASE RATE.

SCOPE OF EXPERIMENT



LOADING -

1. STATIC; CROSSHEAD DISPLACEMENT RATE 0.012 CM/MIN
2. FATIGUE; R = 0.1, F = 5HZ

LAMINATES - AS-3501-6

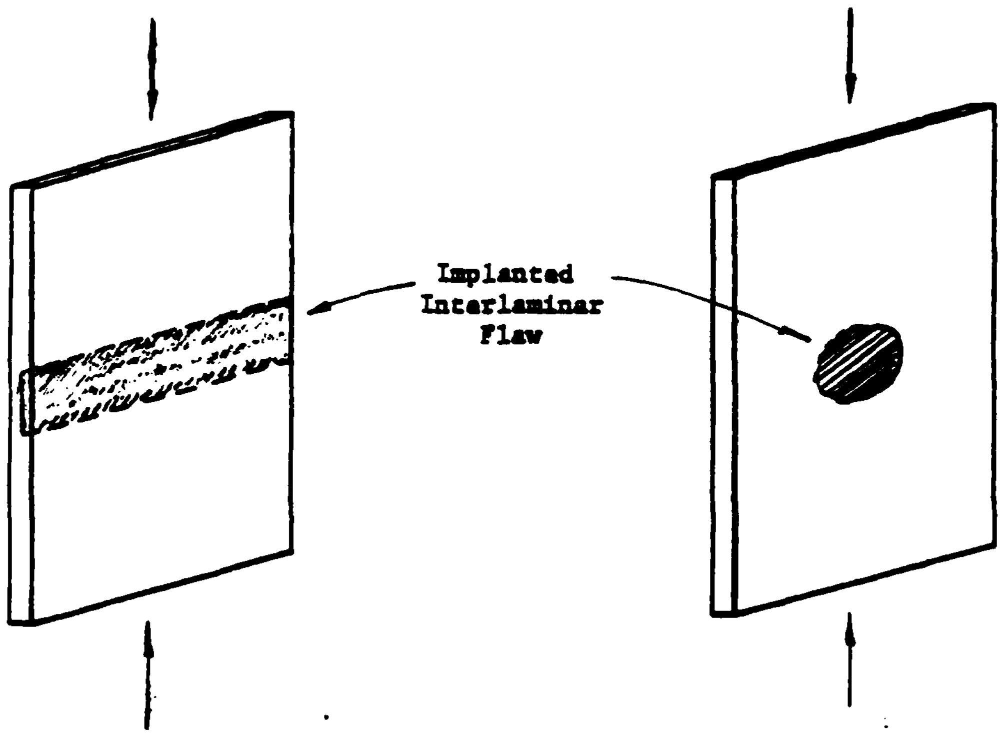
1. $(0_2/90_2/45_2/-45_2)_s$, $(90_2/0_2/-45_2/45_2)_s$
2. $(0/90/0/90/45/-45/45/-45)_s$

   $(90/0/90/0/-45/45/-45/45)_s$
3. THE ABOVE WITH INTERLAMINAR IMPLANTS:

   TYPE A IMPLANT

   TYPE C IMPLANT

TYPE A AND TYPE C INTERLAMINAR INPLANTS



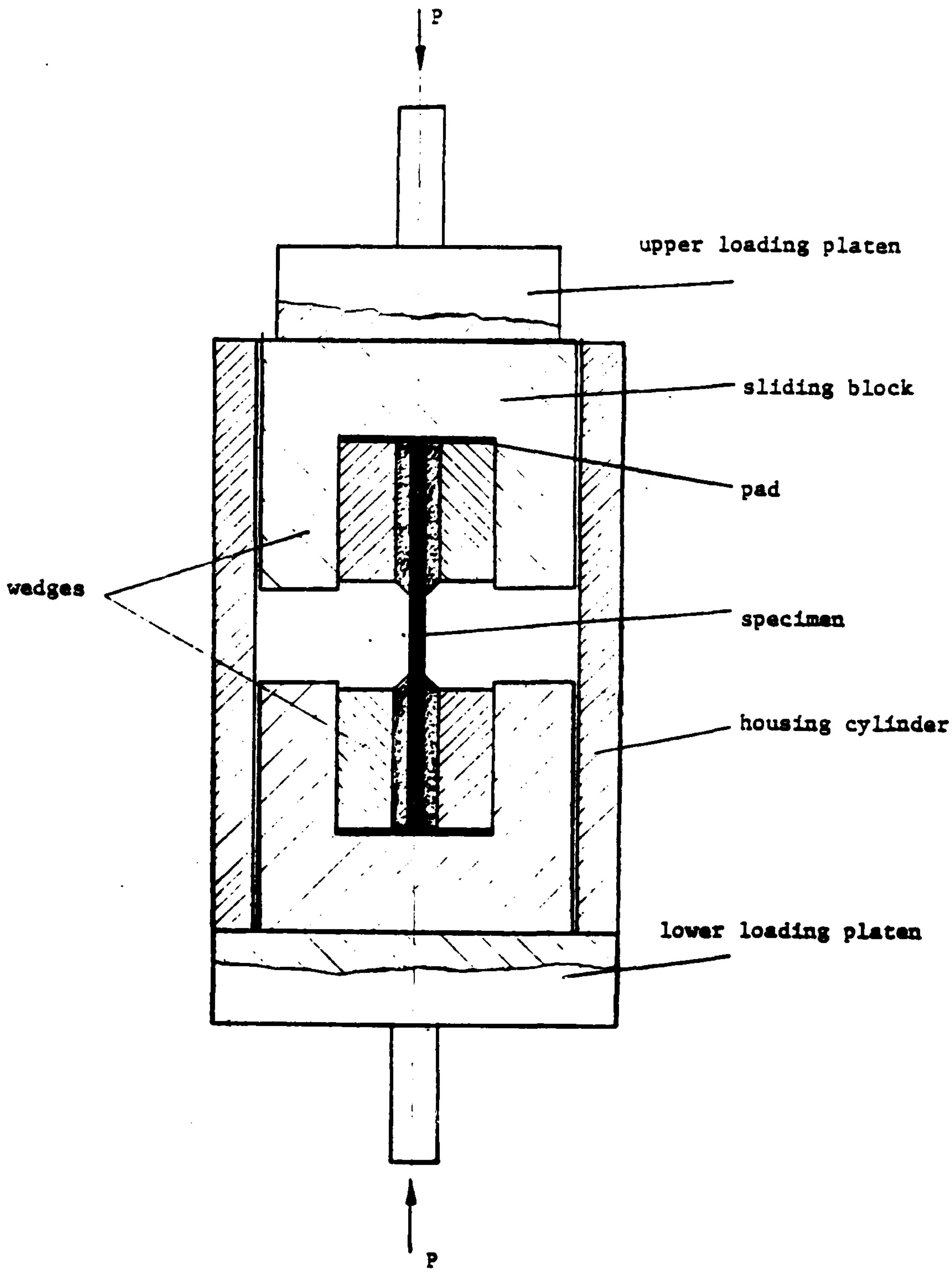
TYPE A

TYPE C

THE INPLANTS ARE INTRODUCED BY INBEDDING TEFLON FILM (-50μ THICKNESS) AT THE -45/45 INTERFACE.

TYPE A : 6.3MM X 25.4MM STRIP FILM

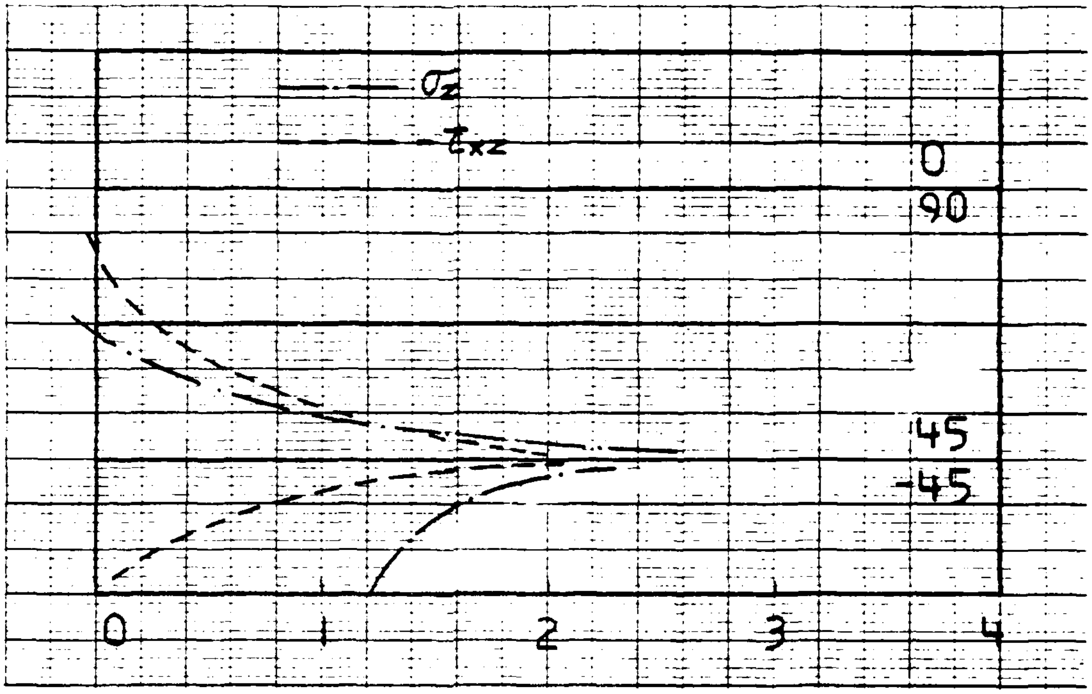
TYPE C : 6.3MM DIAMETER ROUND FILM

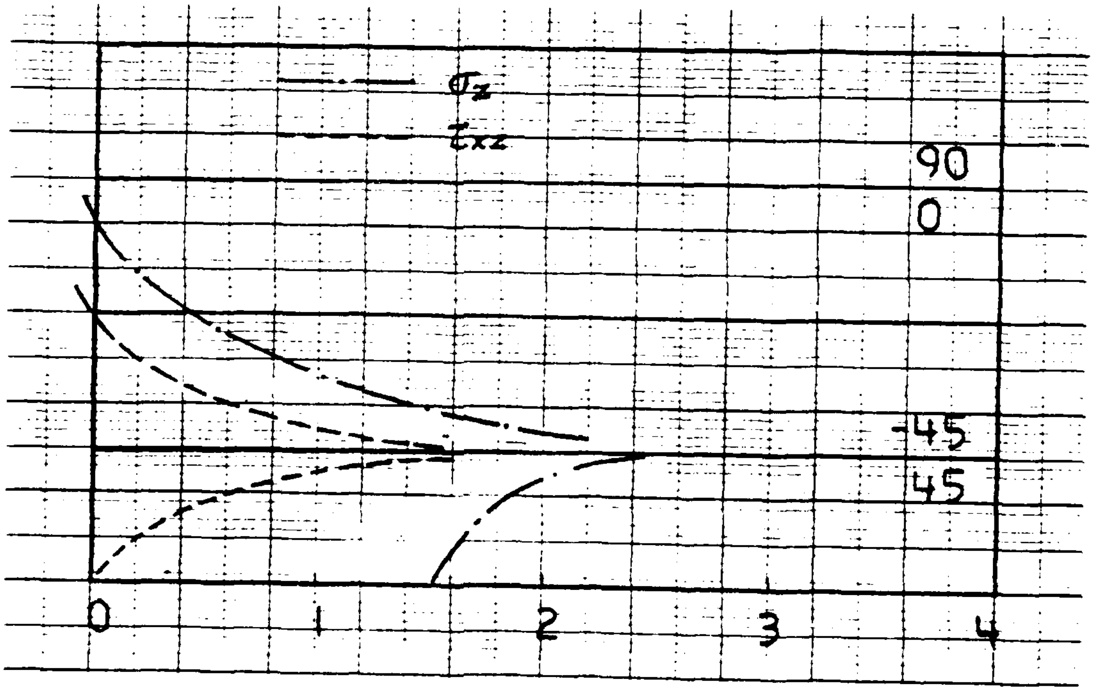


COMPRESSION TEST APPARATUS

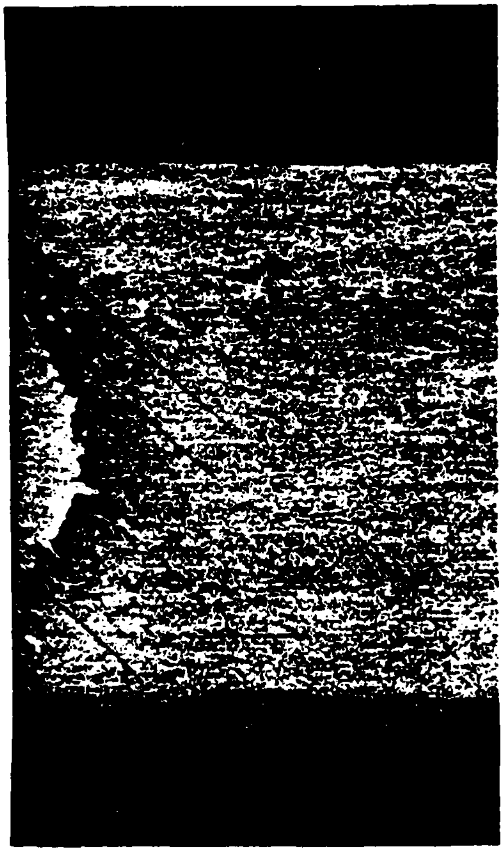
FREE EDGE INTERLAMINAR STRESS DISTRIBUTIONS

(PSI PER 10^{-6} COMPRESSIVE STRAIN)





CRITERION FOR INITIATION



$$G(\bar{e}_X) = (C_e\,\bar{e}_X^{\,2})2T$$

$$G(\bar{e}_X) \to G_C$$

$$C_e = C_e^{\,I} + C_e^{\,II} + C_e^{\,III}$$

$$G_e = G_{(I,II,III)c}$$

	$C_e^{\,I}$	$C_e^{\,III}$	$C_e^{\,III}/C_e^{\,I}$
$(0_2/90_2/45_2/-45_2)_s$	0.86×10^6	1.03	1.2
$(90_2/0_2/-45_2/45_2)_s$	1.46	0.69	0.5

DELAMINATION INITIATION

(STATIC COMPRESSIVE LOADING)

	PREDICTED	EXPERIMENT
$(0_2/90_2/45_2/-45_2)_s$	51.4 KSI	49.2 KSI
$(90_2/0_2/-45_2/45_2)_s$	49.0 KSI	45.7
$(0/90/0/90/45/-45---)_s$	72.4	62.0*
$(90/0/90/0/-45/45---)_s$	69.0	62.5*

CALCULATION BASED ON $G_{(I,III)c}$ = 1.0 IN-LB/IN^2

* DELAMINATION OBSERVED ONLY AT LAMINATE FAILURE.

EFFECT OF INTERLAMINAR INPLANTS

(STATIC COMPRESSIVE LOADING)

LAMINATE	ONSET DELAM	FAILURE
$(0_2/90_2/45_2/-45_2)_s$	49.2 KSI	53.6
$(0_2/90_2/45_2/-45_2)_s$-A	35.0	41.5
$(0_2/90_2/45_2/-45_2)_s$-C	36.4(*)	48.0
$(0/90/0/90/45/-45---)_s$	62.0(**)	62.0
$(0/90/0/90/45/-45---)_s$-A	45.0	55.5
$(0/90/0/90/45/-45---)_s$-C	38.5(***)	53.0

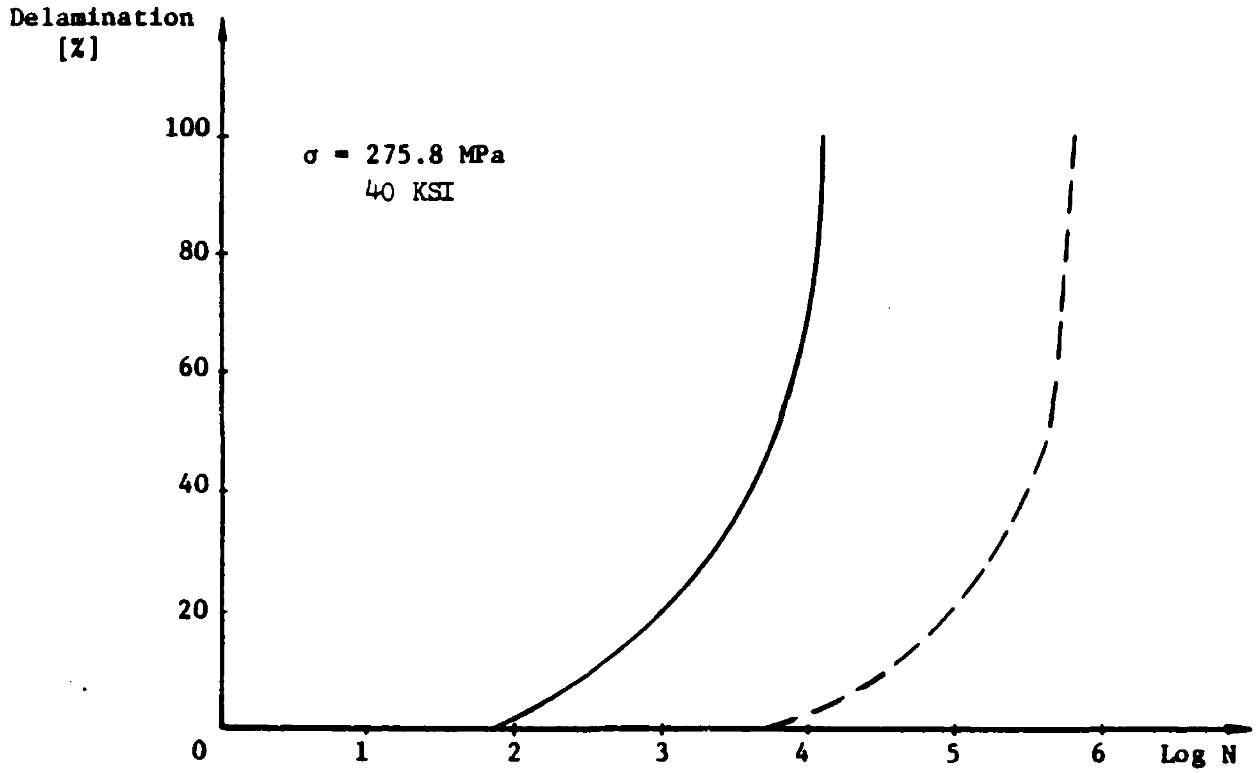
* FIRST DELAMINATION AT FREE EDGE

** SIMULTANEOUS DELAMINATION/FAILURE

*** FIRST DELAMINATION AROUND CIRCULAR FILM

EFFECT OF INTERLAMINAR INPLANTS

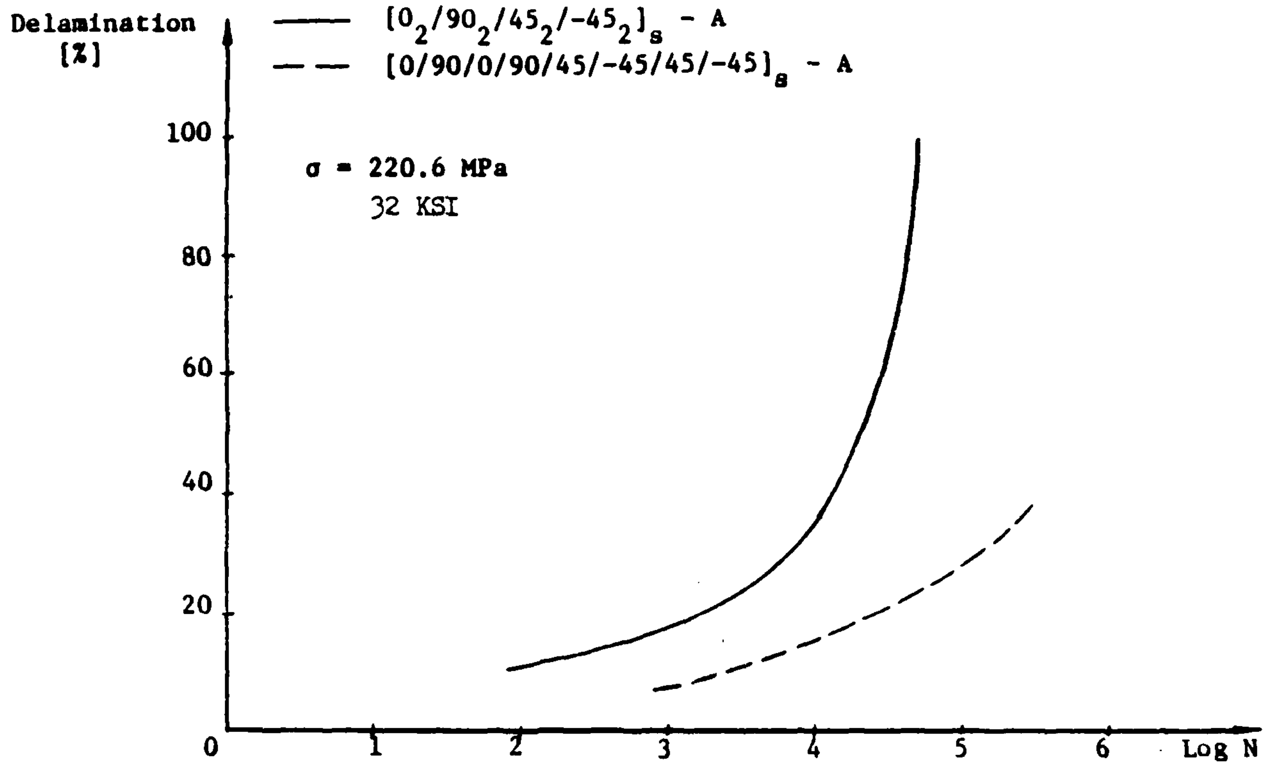
(COMPRESSIVE FATIGUE LOADING)

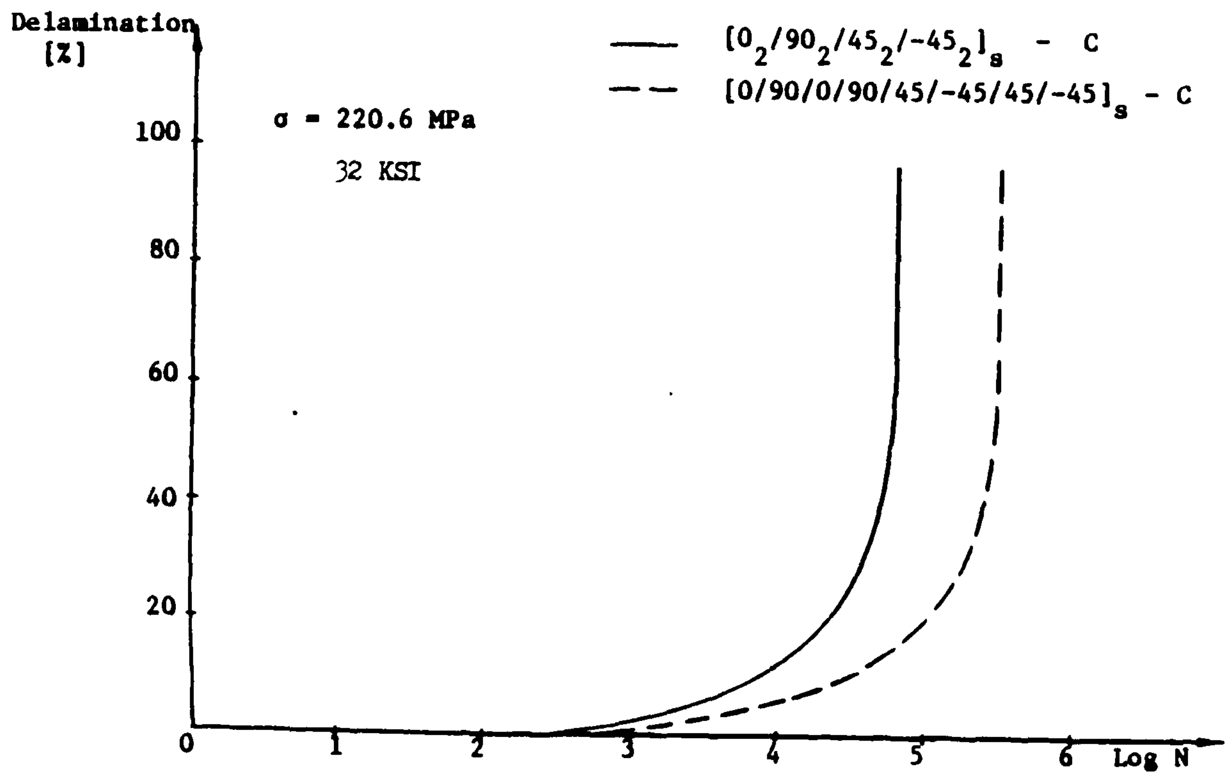


—— $[0_2/90_2/45_2/-45_2]_s$ laminate

—·— $[0/90/0/90/45/-45/45/-45]_s$ laminate

EFFECT OF INPLANTS - CONTINUED





A SUGGESTED DELAMINATION GROWTH MODEL

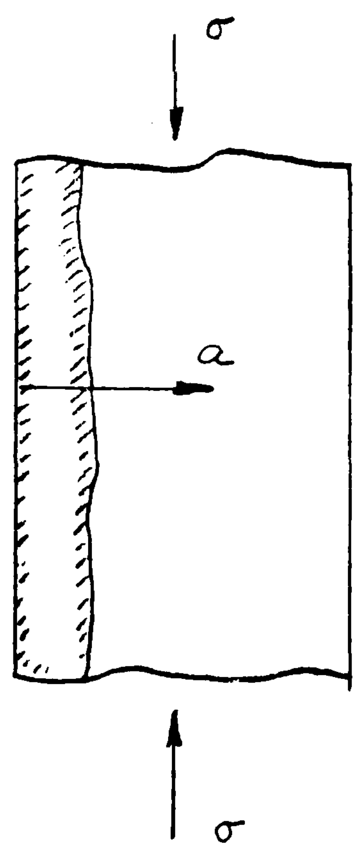
SINCE $G = G(a, \sigma$, LAMINATION GEOMETRY), WE MAY PROPOSE A GROWTH RATE LAW IN THE FORM

$$\frac{Da}{DN} = F(G/G_C)$$

A SIMPLE FORM OF THE ABOVE MAY BE

$$\frac{Da}{DN} = C_I (G/G_C)^{C_2}$$

WHERE C_I AND C_2 ARE EMPIRICAL CONSTANTS.



A CUMULATIVE DAMAGE MODEL FOR ADVANCED COMPOSITE MATERIALS

A Presentation at the
Eighth Annual Mechanics of Composites Review
Dayton, Ohio - October 1982

by

Pei Chi Chou

PROJECT

SPONSORED BY: AFWAL/MATERIALS LABORATORY

MONITOR: Mr. MARVIN K. KNIGHT

CONTRACTOR: DYNA EAST CORP. (DREXEL UNIVERSITY)

INVESTIGATORS: Dr. P.C. Chou, Dr. A.S.D. Wang, Dr. H. Miller

OBJECTIVES

DEVELOPMENT OF A CUMULATIVE DAMAGE MODEL TO PREDICT THE FAILURE MODES, STRENGTH, AND FATIGUE LIFE OF COMPOSITE MATERIALS. THE MODEL WILL UTILIZE FRACTURE MECHANICS, RANDOM VARIABLES, AND FINITE-ELEMENT CODES, AS NEEDED. THE RESULTS MUST BE VERIFIED BY EXPERIMENTS.

STEPS OF DEVELOPMENT

1. BASIC FRACTURE MECHANICS APPROACH TO EDGE DELAMINATION AND TRANSVERSE CRACK, (STATIC, ONSET OF FIRST CRACK).

2. MODEL FOR MULTIPLE-TRANSVERSE CRACKS, STATIC.

3. MODEL FOR FATIGUE, TRANSVERSE CRACKS AND DELAMINATION.

4. CUMULATIVE DAMAGE MODEL.

5. EXTENSION TO FINAL LAMINATE FAILURE.

CONCLUSIONS

1. A MODEL FOR MULTIPLE TRANSVERSE CRACKING IN THE 90° PLIES WAS DEVELOPED. IT IS BASED ON THE ASSUMPTION THAT THE LENGTH OF THE INHERENT MICROCRACKS IN THE MATERIAL IS DISTRIBUTED RANDOMLY. USING A TWO-PARAMETER WEIBULL DISTRIBUTION AND THE CRITERION FOR CRACK PROPAGATION BASED FRACTURE MECHANICS, RESULTS FROM THE MODEL AGREE WITH EXPERIMENTAL DATA.

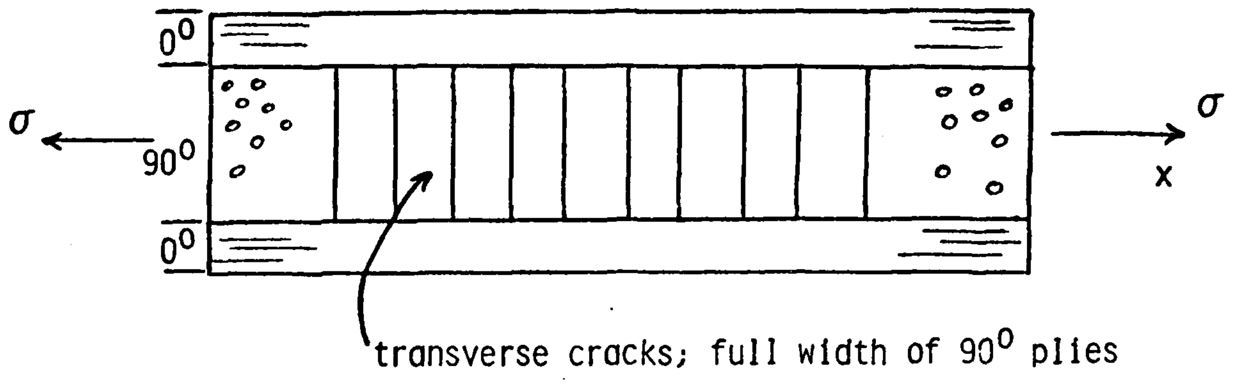
2. UNDER FATIGUE LOADING, THE MODEL ASSUMES THAT THE LENGTH OF A CRACK EXTENDS UNDER EACH CYCLE ACCORDING TO AN EMPIRICAL RULE. TOGETHER WITH THE RANDOM DISTRIBUTION OF THE INITIAL CRACK LENGTH, THE MODEL PREDICTS THE CORRECT CRACK DENSITY AS A FUNCTION OF FATIGUE CYCLES.

3. A CUMULATIVE DAMAGE MODEL IS ALSO PROPOSED. IT IS BASED ON THE CONCEPT OF "DAMAGED STATE." IT IS SIMILAR TO THE MINER'S RULE, EXCEPT HERE IT IS THE "DAMAGE" (CRACK DENSITY, OR DELAMINATION AREA) THAT IS ACCUMULATING.

4. PLANS FOR EXTENDING THE PRESENT APPROACH TO THE FINAL FAILURE OF OTHER TYPES OF LAMINATES ARE MADE.

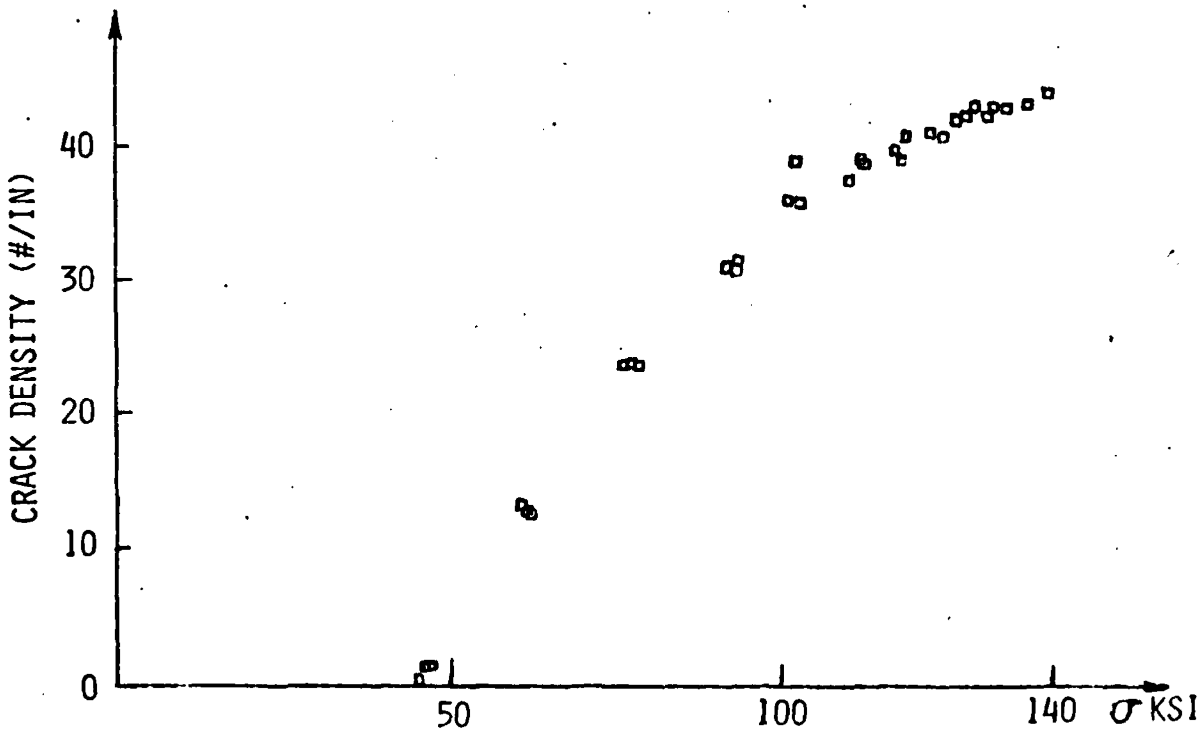
STATIC CASE

EXPERIMENTAL RESULT



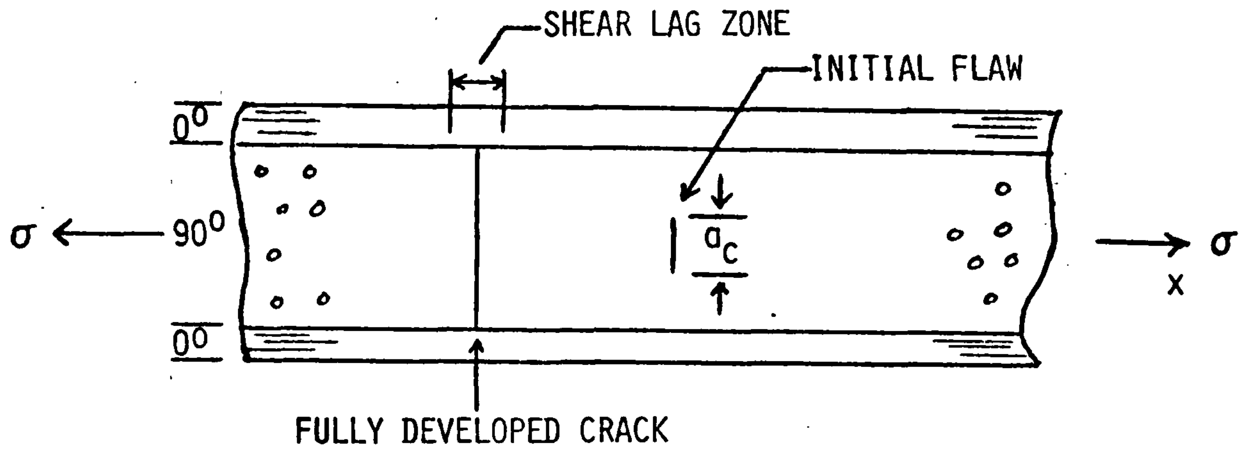
n = crack density

= number of cracks per unit length



EXPERIMENTAL LOAD-TRANSVERSE CRACK DENSITY DATA FOR $[0_2/90_2]_s$

FRACTURE MECHANICS RESULTS



1. INITIAL FLAW a_c

   IN THE 90^0 PLIES THERE ARE INITIAL FLAWS, OR CRACKS OF LENGTH a_c.

2. LOAD FOR a_c TO PROPAGATE

   FROM FRACTURE MECHANICS PRINCIPLE AND A FINITE ELEMENT PROGRAM, FOR EACH FLAW LENGTH a_c, THERE IS A LOAD (STRESS) THAT WILL CAUSE THE FLAW TO PROPAGATE TO FULL WIDTH OF 90^0 PLIES.

3. SHEAR LAG ZONE

   A "SHEAR LAG" ZONE SURROUNDS A FULLY DEVELOPED CRACK. OUTSIDE OF THIS ZONE, THE STATE OF STRESS AND STRAIN IS NOT AFFECTED BY THIS CRACK.

4. CRACK INDEPENDENCE

   OUTSIDE OF THE SHEAR-LAG ZONES OF FULLY DEVELOPED CRACKS THE CRACK FORMATION IS INDEPENDENT.

RANDOM VARIABLE APPROACH

1. RANDOM INITIAL FLAW

   FOR A GIVEN SPECIMEN, ASSUME THERE IS A RANDOM DISTRIBUTION OF INITIAL FLAWS OF LENGTH a_c, OR EQUIVALENT LENGTH a_c.

2. CRACK DEVELOPMENT

   UNDER A LOAD σ_1 ALL INITIAL CRACKS THAT ARE NOT IN THE SHEAR LAG ZONE OF OTHER DEVELOPED CRACKS AND ARE OF A CERTAIN LENGTH a_{c_1} OR LARGER, ARE FULLY DEVELOPED.

MATHEMATICAL DEVELOPMENT

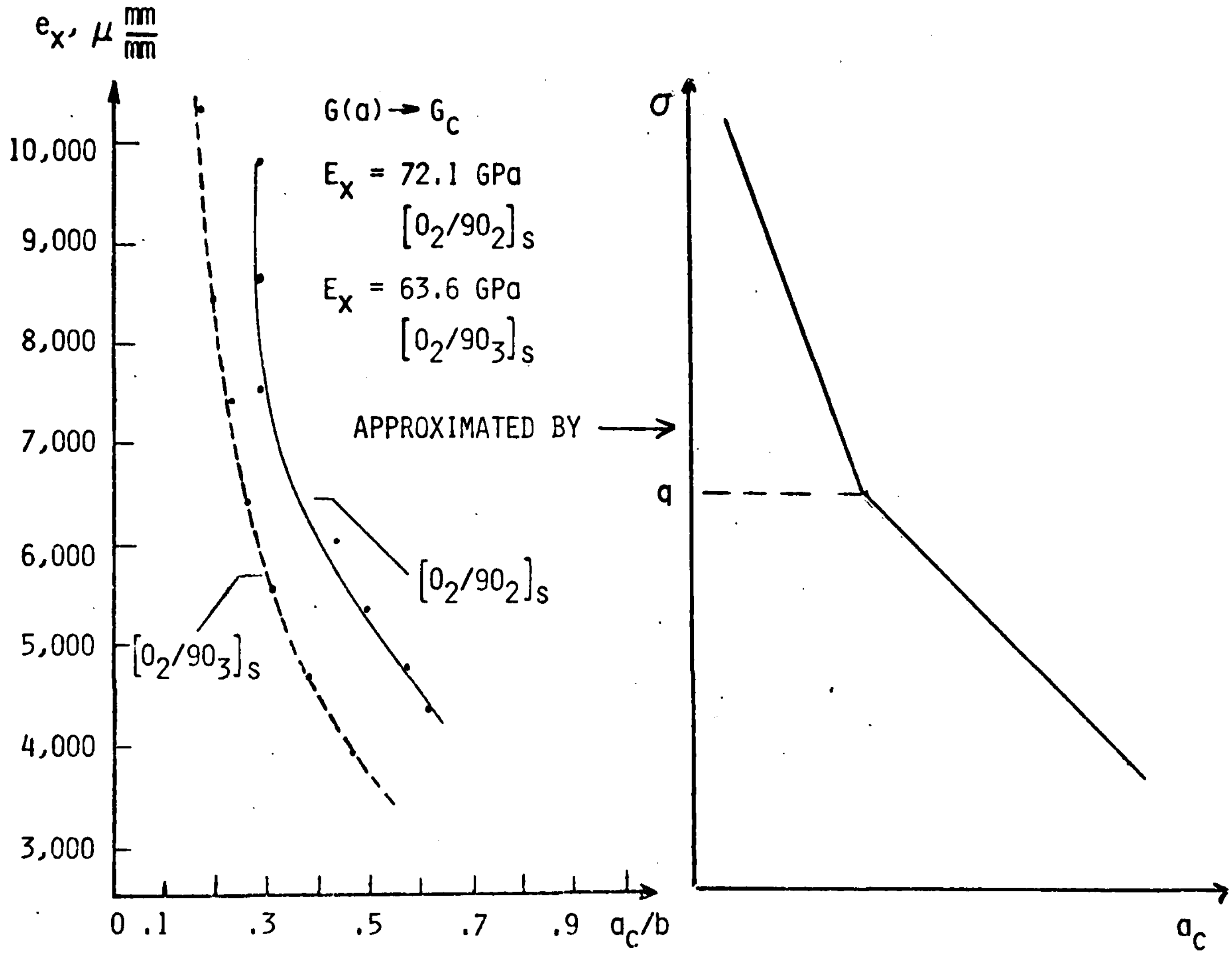
1. DISTRIBUTION AND NUMBER OF a_c

   ASSUME a_c HAS A TWO-PARAMETER WEIBULL DISTRIBUTION

$$F(a_c) = 1 - \exp\left[-\left(\frac{a_c}{\beta}\right)^{\alpha}\right] \qquad (1)$$

$F(a_c)$ = PERCENT OF FLAWS THAT HAVE LENGTH a_c OR SMALLER OR SMALLER.

2. LOAD VS. FLAW SIZE RELATION $\sigma = \sigma(a_c)$ FROM FRACTURE MECHANICS AND FINITE-ELEMENT



TRANSVERSE CRACK INITIATION STRAIN VS. RELATIVE CRACK LENGTH; $(0_2/90_2)_s$ AND $(0_2/90_3)_s$.

$$\sigma = B - C\ a_c, \quad \text{FOR } \sigma < a$$

$$\sigma = B' - c'\ a_c \quad \text{FOR } \sigma > a \qquad (2)$$

3. COMBINING (1) AND (2) GIVE

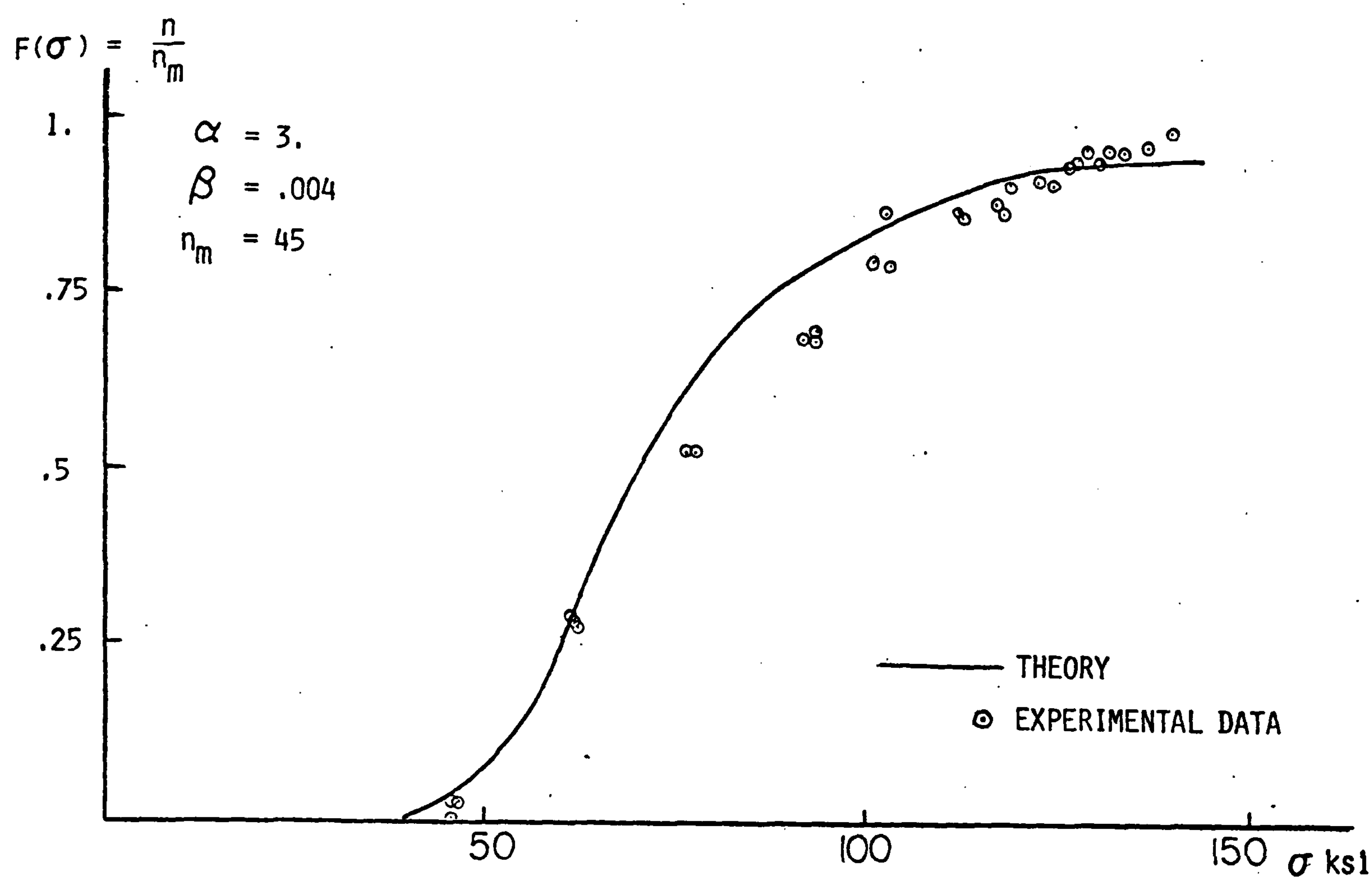
$$\left.\begin{aligned} F(\sigma) &= \exp\left[-\left(\frac{B - \sigma}{\beta c}\right)^{\alpha}\right] \text{ FOR } \sigma < q \\ F(\sigma) &= \text{constant} + \exp\left[-\left(\frac{B' - \sigma}{\beta_{c}'}\right)^{\alpha}\right] \text{ FOR } \sigma > q \end{aligned}\right\} \quad (3)$$

$F(\sigma)$ = PERCENT OF FULLY DEVELOPED FLAWS UNDER LOAD σ.

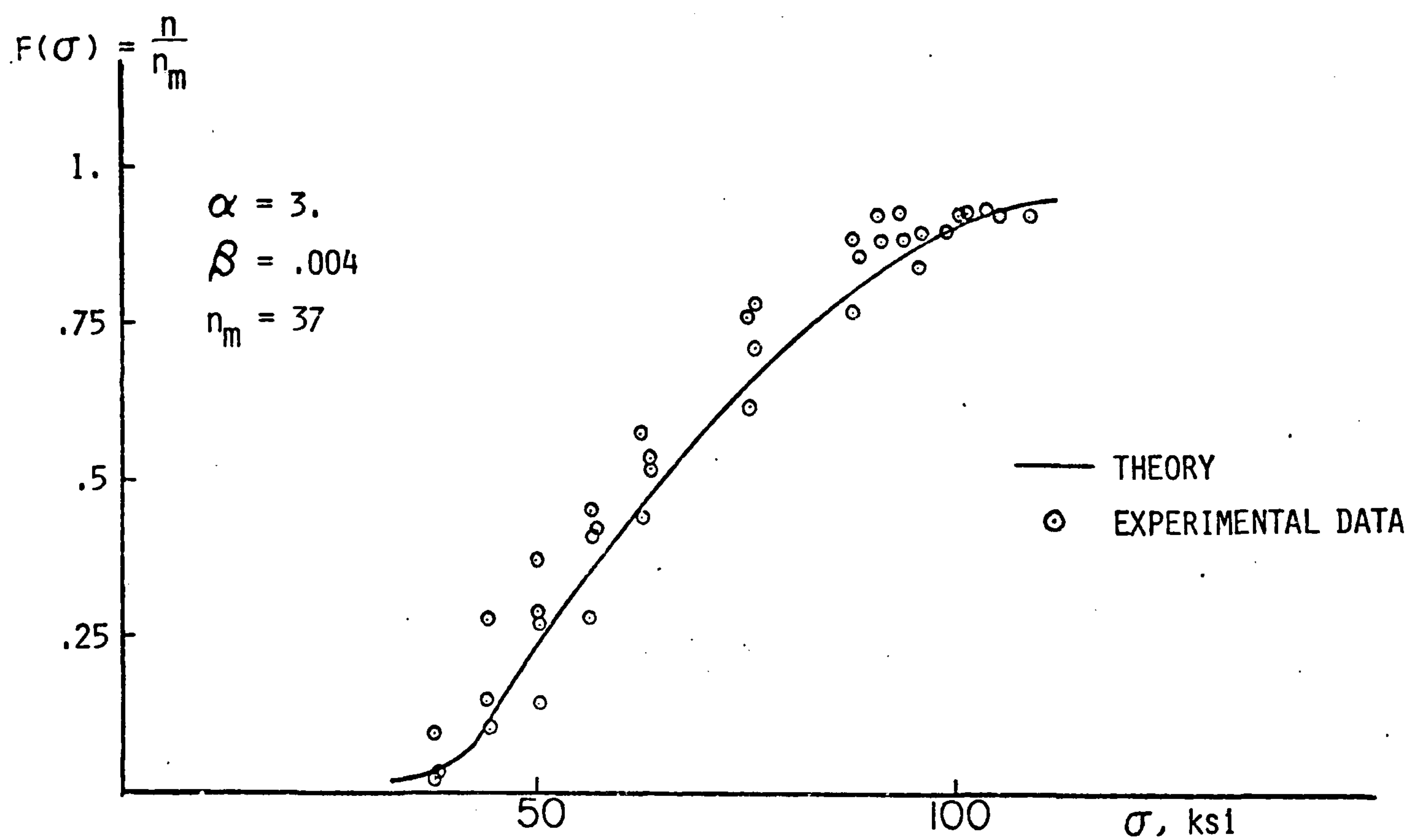
4. CRACK DENSITY VS. LOAD

$$n = n_m F(\sigma)$$

n_m = SATURATION CRACK DENSITY WHICH DEPENDS ON BOTH THE NUMBER OF INITIAL CRACKS AND THE SHEAR LAG ZONE.



COMPARISON OF THE CUMULATIVE DISTRIBUTION OF σ WITH NORMALIZED EXPERIMENTAL TRANSVERSE CRACK DENSITY DATA FOR $[0_2/90_2]_s$ LAMINATE.

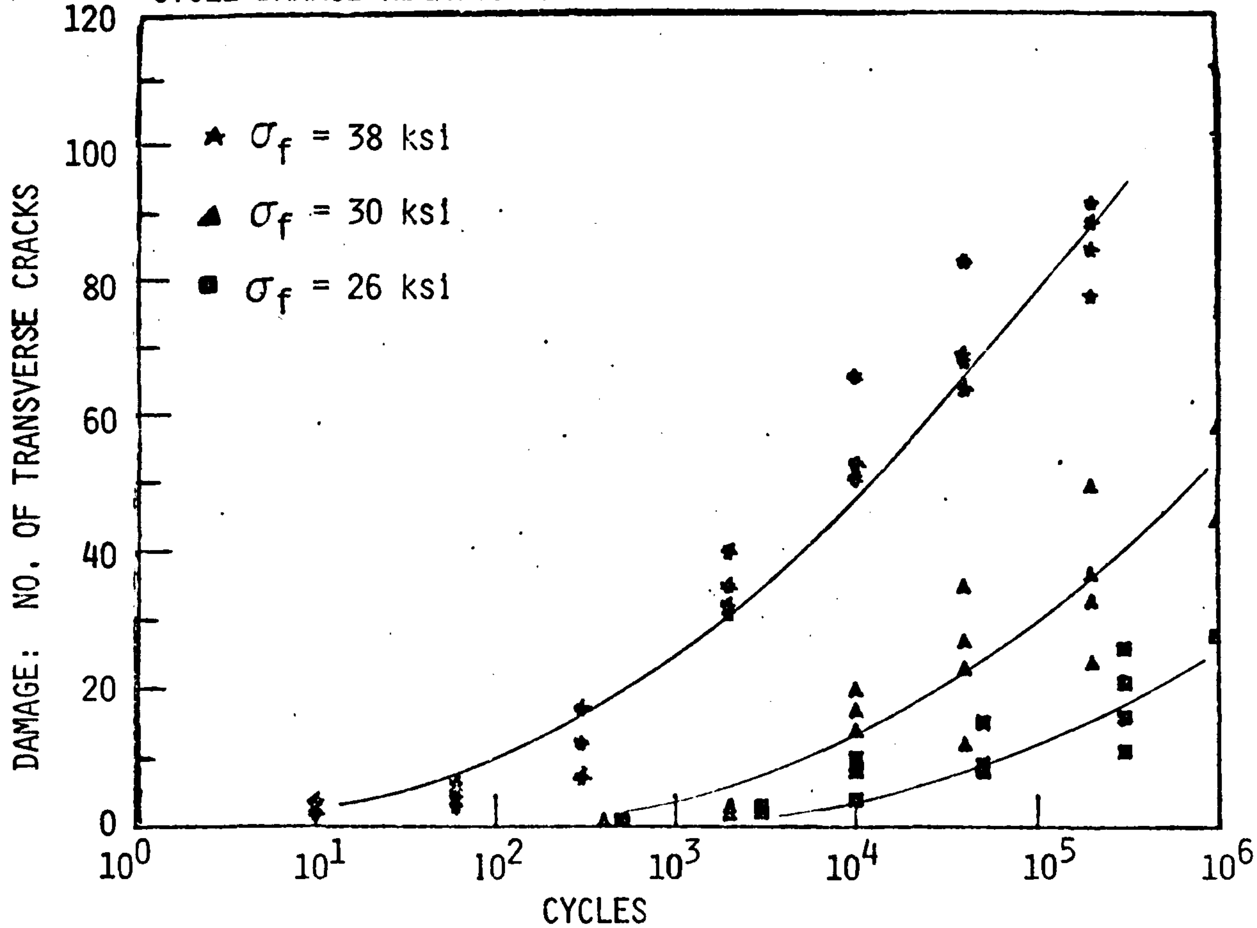


COMPARISON OF THE CUMULATIVE DISTRIBUTION OF σ WITH NORMALIZED EXPERIMENTAL TRANSVERSE CRACK DENSITY DATA FOR $[0_2/90_3]_s$ LAMINATE.

FATIGUE CASE

EXPERIMENTAL RESULTS

CYCLE-DAMAGE RELATION FOR $0_2/90_3$

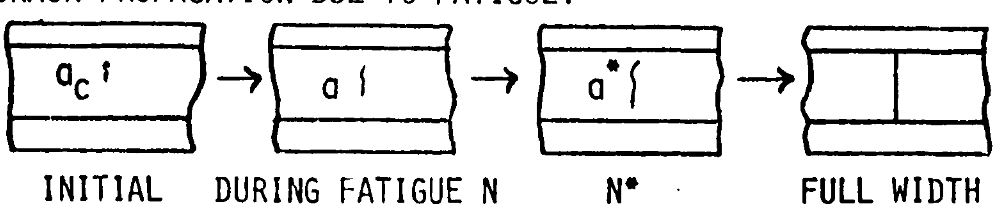


CRACK DENSITY n VS. FATIGUE CYCLE N - CURVES UNDER CONSTANT AMPLITUDE LOAD

FATIGUE MODEL

1. INITIAL FLAW DISTRIBUTION AND NUMBER DETERMINED FROM STATIC TESTS; α, β, n_m DETERMINED.

2. CRACK PROPAGATION DUE TO FATIGUE.



GROWTH DUE TO FATIGUE OF ONE CRACK.

$$\frac{da}{dN} = f(N, a, \sigma, \bar{G} \ldots) \quad \text{-- GENERAL}$$

$$\frac{da}{dN} = k \frac{a}{N} \quad \text{-- A SPECIFIC CASE}$$

$$k = k(\sigma, G_c)$$

A CRACK OF INITIAL LENGTH a_c, WILL INCREASE IN LENGTH PER CYCLE OF LOADING.

AFTER INTEGRATION

$$a = a_c N^k \qquad \text{OR} \qquad \ln \frac{a}{a_c} = k \ln N$$

3. DETERMINE a^* FROM STATIC ANALYSIS.

a^* = THE CRACK LENGTH THAT WILL PROPAGATE TO FULL WIDTH UNDER LOAD σ.

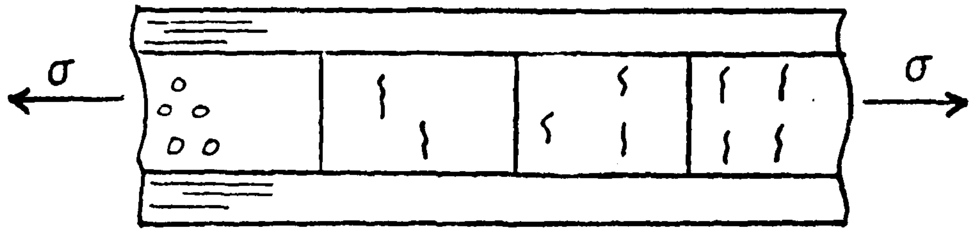
OR

a^* = WIDTH OF 90^o PLIES

4. DETERMINE N^*

N^* = THE FATIGUE CYCLE THAT WILL MAKE a_c GROW TO a^*, THUS FULL WIDTH.

5. DETERMINATION OF F(N*)



GROWTH OF CRACKS OF A TEST SPECIMEN

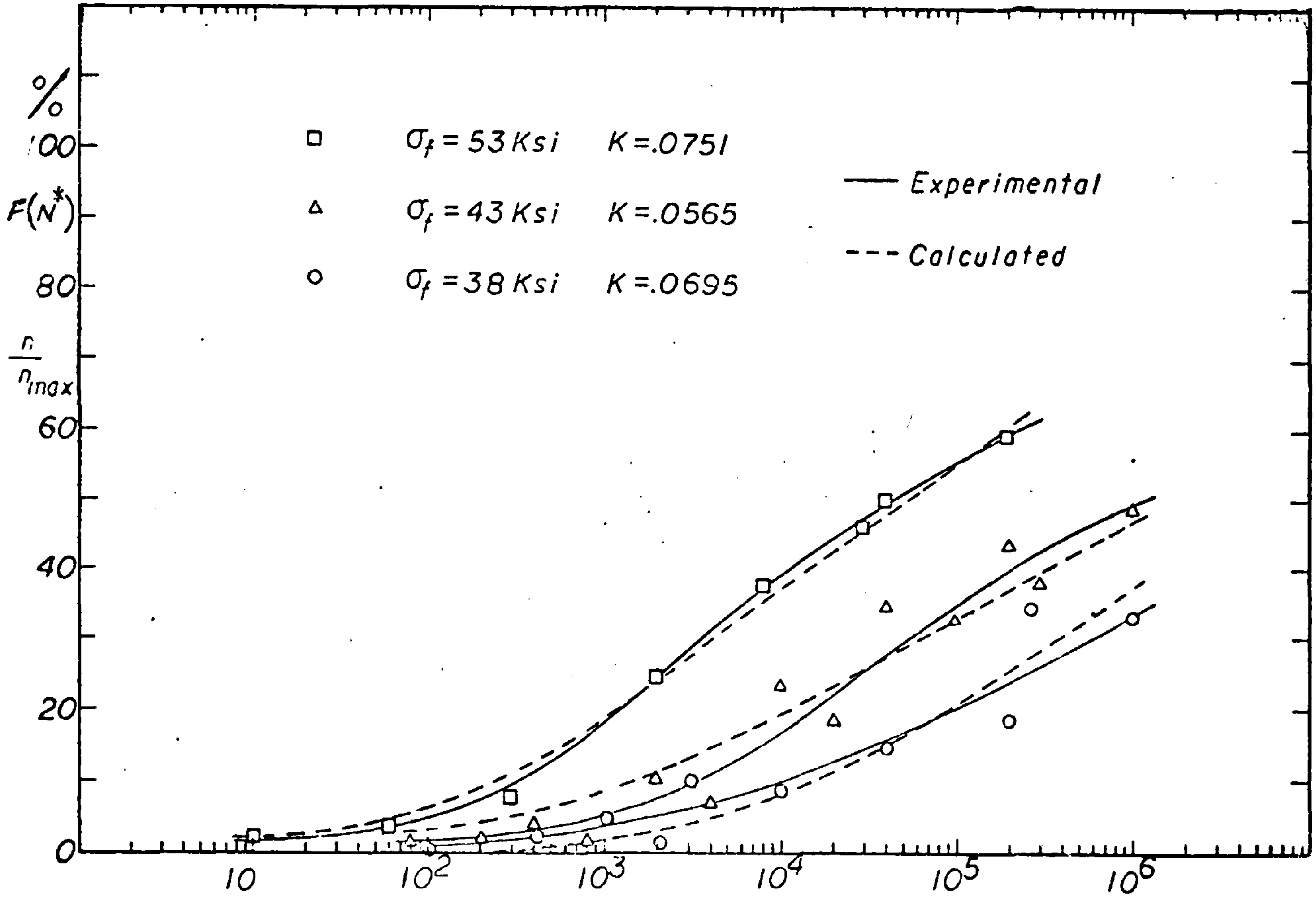
$$a^* = a_c (N^*)^k \quad \text{OR} \quad a_c = a^* \frac{1}{(N^*)^k}$$

SINCE a_c IS RANDOM, N* IS ALSO RANDOM,

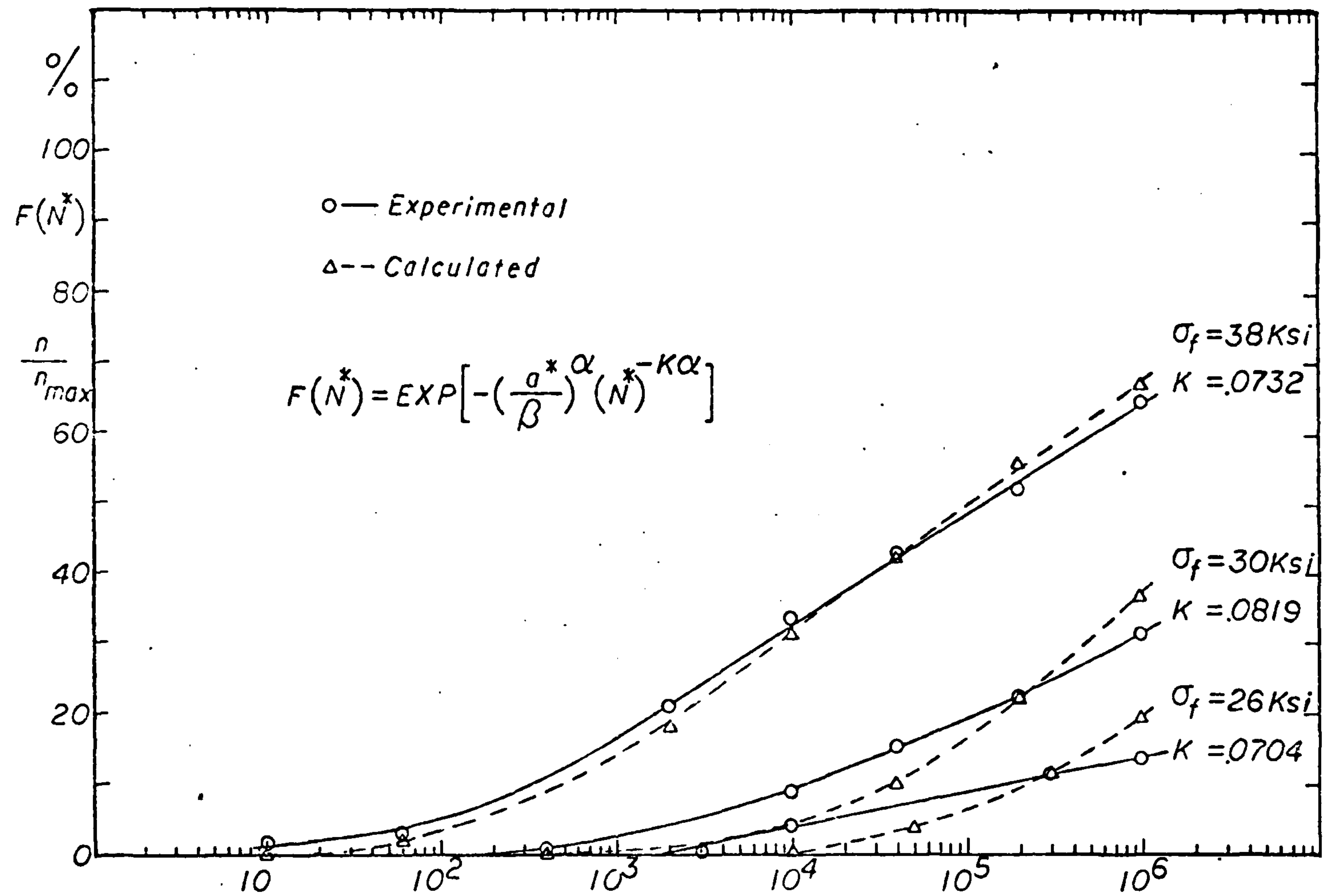
$$f(N^*) = f(a_c)\left|\frac{da_c}{dN^*}\right|$$

$$F(N^*) = \exp\left[-\frac{a}{\beta}(N^*)^{-k\alpha}\right]$$

AS IN STATIC CASE, $n = n_m F(N^*)$



DISTRIBUTION OF N* FOR $[0_2/90_2]_s$ LAMINATE FOR THREE MAXIMUM FATIGUE LOADS UNDER CONSTANT AMPLITUDE FATIGUE.

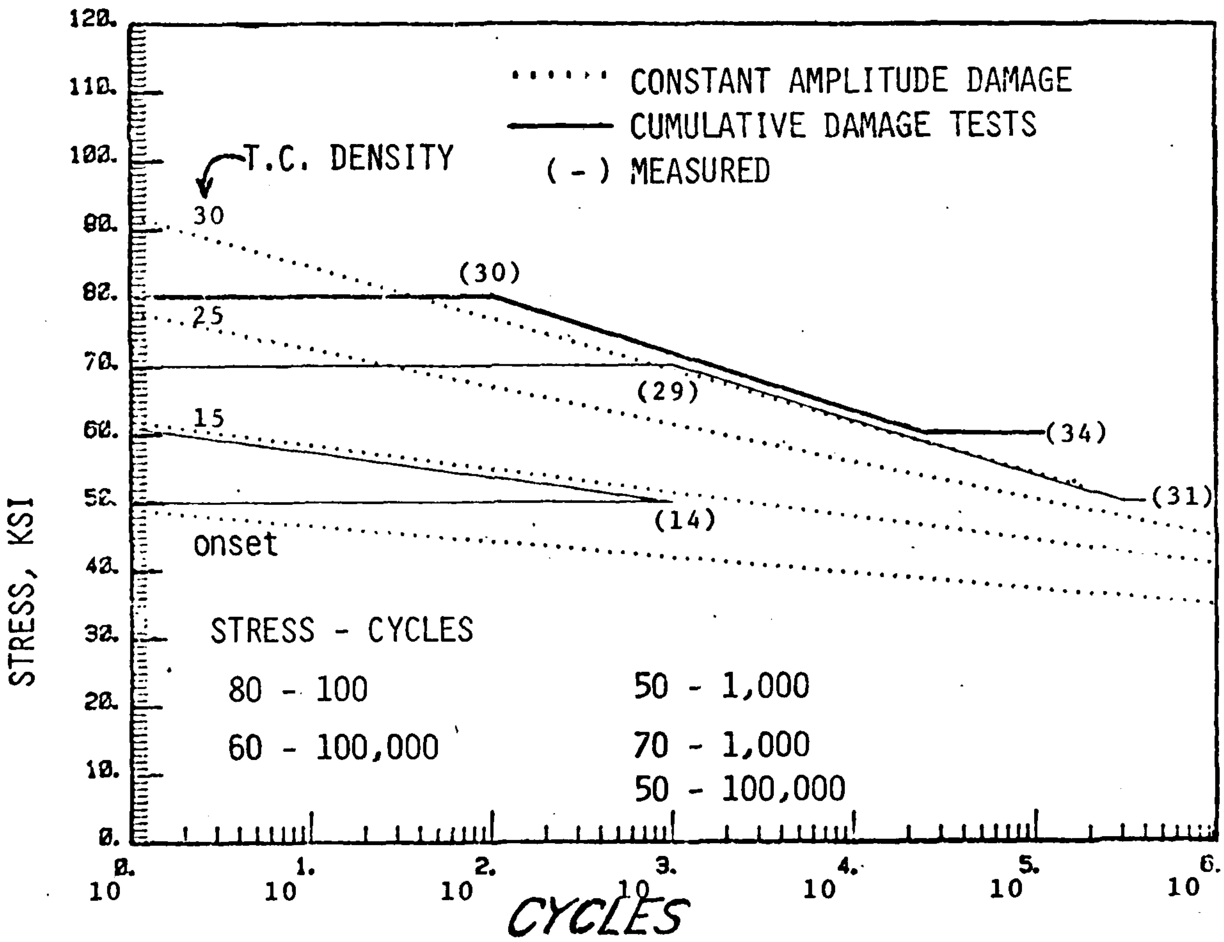


DISTRIBUTIONS OF N* FOR $[0_2/90_3]_s$ LAMINATE FOR THREE MAXIMUM FATIGUE LOADS UNDER CONSTANT AMPLITUDE FATIGUE.

CUMULATIVE DAMAGE

1. CONSTANT DAMAGE STATE CONCEPT.
2. USE OF CONSTANT AMPLITUDE FATIGUE MODEL TO GENERATE CONSTANT DAMAGE CURVES IN S - N PLANE.
3. TRACE SPECTRUM LOADING USING DAMAGE CURVES.

CUMULATIVE DAMAGE TESTS $(0_2/90_2)_s$



AD P001253

PROPERTY DEGRADATION APPROACH TO CUMULATIVE DAMAGE MODELING OF ADVANCED COMPOSITES

GENERAL DYNAMICS
FORT WORTH DIVISION

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CONTRACT NUMBER: F33615-81-C-5049
SPONSER : AFWAL/MLBM
PROJECT ENGINEER: M. KNIGHT

PROGRAM OBJECTIVE:

- O TO DEVELOP A CUMULATIVE DAMAGE MODEL FOR ADVANCED COMPOSITE MATERIALS THAT IS CAPABLE OF PREDICTING THE STRENGTH, STIFFNESS, AND LIFE OF AN ARBITRARY COMPOSITE LAMINATE WHEN SUBJECTED TO A VARIETY OF COMPLEX LOAD HISTORIES.

THREE PHASE PROGRAM

- O PHASE I : MODEL DEVELOPMENT
- O PHASE II : MODEL REFINEMENT
- O PHASE III: MODEL VERIFICATION

CONCLUSIONS TO DATE

EXPERIMENTAL

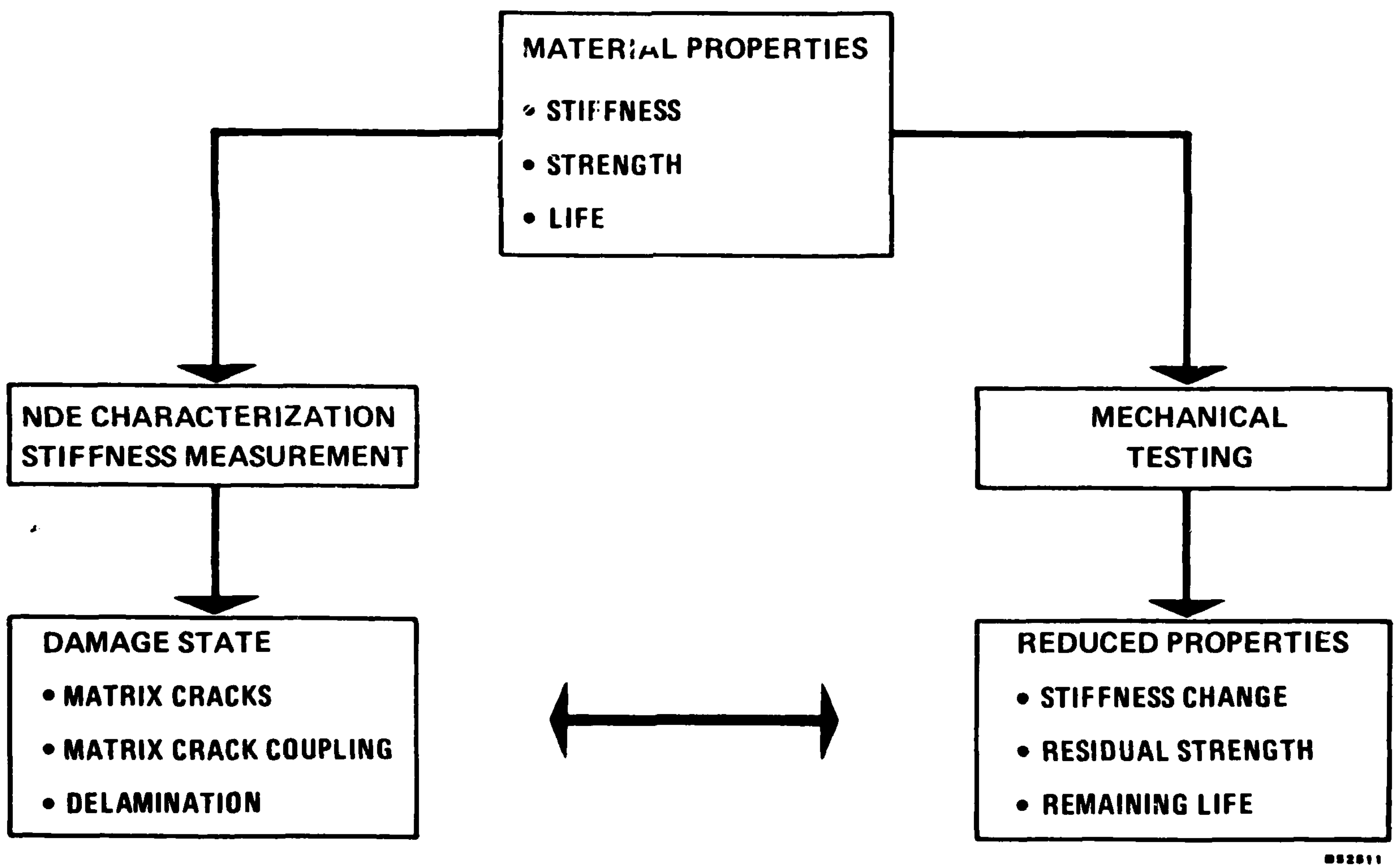
- THERE IS AN INTERACTION BETWEEN THE DAMAGE MODES AND MECHANISMS OBTAINED IN TENSION (TRANSVERSE CRACKING) AND COMPRESSION (DELAMINATION).
- THE EXISTENCE OF TRANSVERSE CRACKS OBTAINED UNDER TENSILE LOADING SUBSTANTIALLY REDUCES THE DELAMINATION RESISTANCE IN COMPRESSION.
- DAMAGE OCCURS IN A CONSISTANT AND PROGRESSIVE MANNER.

MODEL

- BASIC MODEL HAS BEEN CONCEIVED AND DEVELOPED.
- MODEL BASED UPON THE PRINCIPAL OF MECHANICS.
- MODEL BASED UPON EXPERIMENTALLY OBSERVED DAMAGE MECHANISMS.
- RESIDUAL STRENGTH IS THE BASIS FOR EQUIVALENCY OF DAMAGE STATES.
- RESIDUAL STRENGTH INFLUENCED BY LOCAL STRESS REDISTRIBUTIONS.
- MODEL DEVELOPED AT ENGINEERING (PLY) LEVEL.
- REASONABLE AGREEMENT BETWEEN EXPERIMENTAL RESULTS AND MODEL PREDICTIONS OF RESIDUAL STRENGTH.
- CURRENTLY MODIFYING AND REFINING MODEL FOR COMPLEX LOAD HISTORIES.

CUMULATIVE DAMAGE MODEL FOR ADVANCED COMPOSITE MATERIALS

PROPERTY DEGRADATION APPROACH

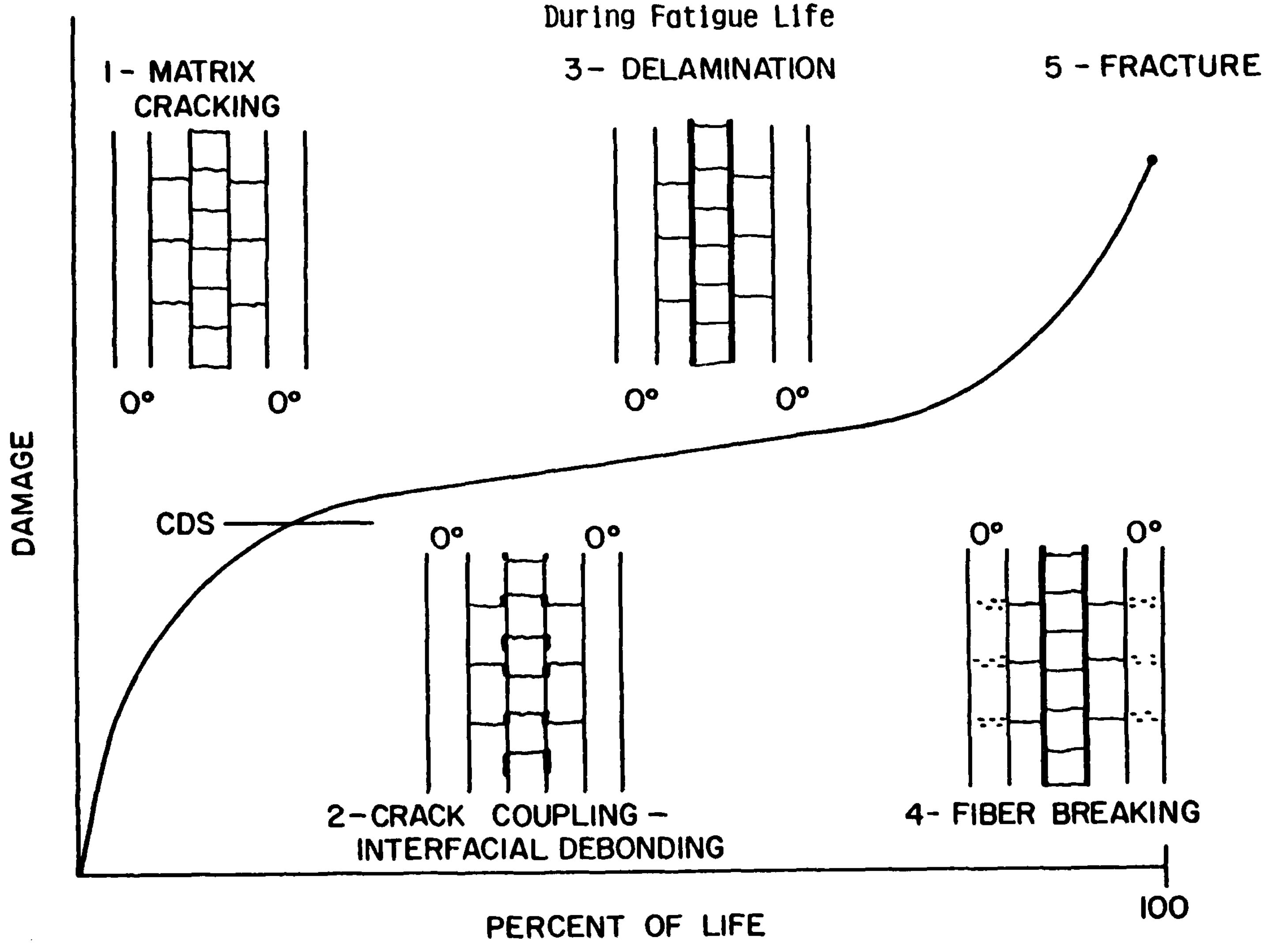


LIFE AND RESIDUAL STRENGTH DATA

QUASI-ISOTROPIC

LOAD MODE	LEVEL	TYPE A $[(0/\pm45/90)_S]_{3S}$	TYPE B $[(0/90/\pm45)_S]_{3S}$	TYPE C $[(0/+45/90/-45)_S]_{3S}$
T-T	4500 $\mu\epsilon$	NO FAILURES (90% RETENTION)	NO FAILURES (100% RETENTION)	NO FAILURES (100% RETENTION)
	6000	NO FAILURES (82% RETENTION)	NO FAILURES (85% RETENTION)	NO FAILURES (97% RETENTION)
C-C	4500 $\mu\epsilon$	NO FAILURES ±45° CRACK	NO FAILURES	
	5000		FAILURE @ 500K	
	5500		FAILURE @ 50K	
	6000	NO FAILURE ±45° CRACK	FAILURE @ 5K	NO FAILURE
	7000			NO FAILURE
	8000			FAILURE @ 360K
T-C	4000 $\mu\epsilon$			FAILURE @ 650K
	4500	FAILURE @ 50K	FAILURE @ 50K	FAILURE @ 100K
	6000		FAILURE 1K	

Controlling Damage Modes During Fatigue Life



APPROACH

Metals → Composites

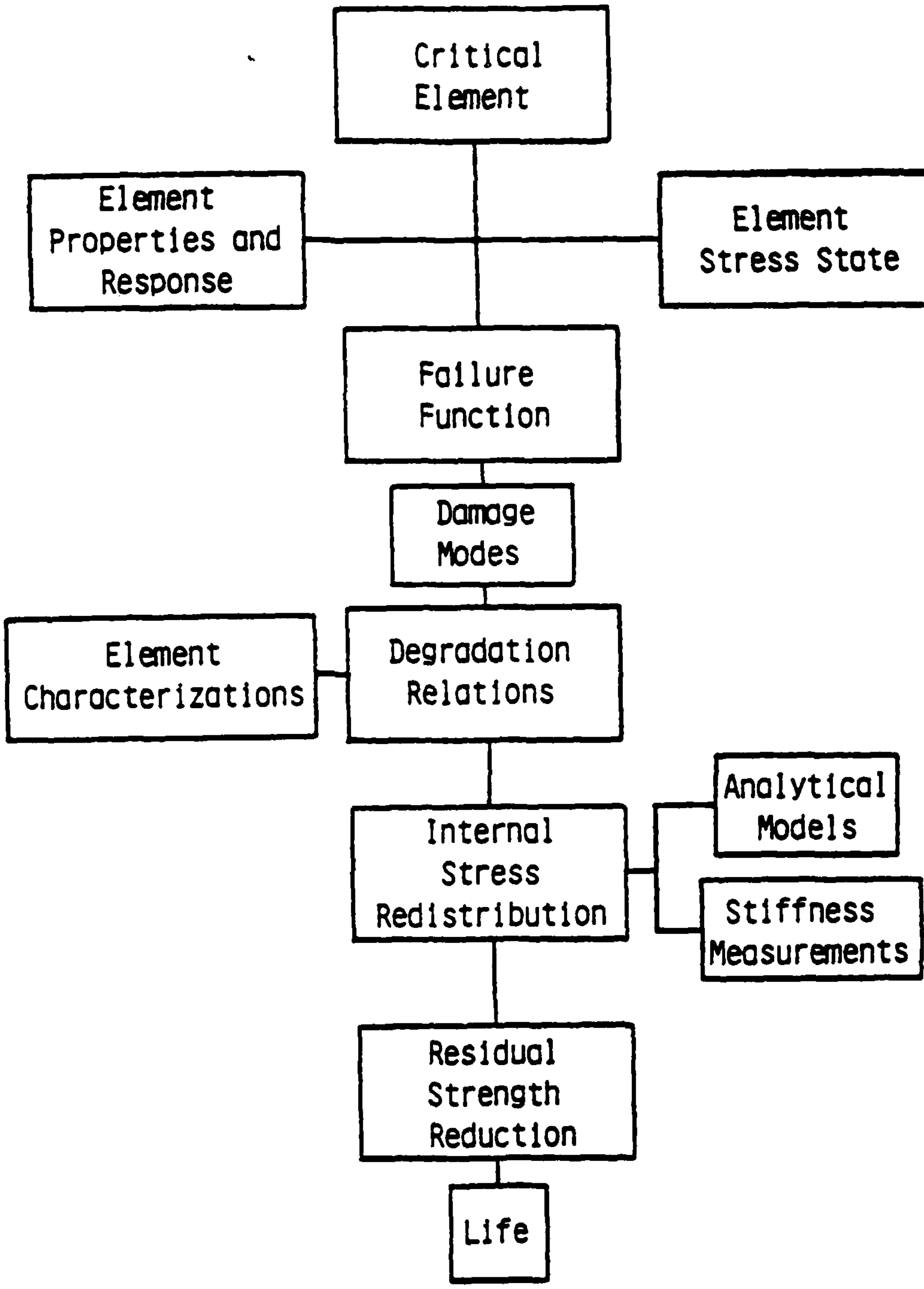
MECHANICS	PHENOMENOLOGICAL
Fracture mechanics	Crack propagation equation
→ Damage mechanics	(Paris, Foreman, etc.)
	→ Degradation equation
	(Broutman, Reifsnider)

↓

Residual Strength

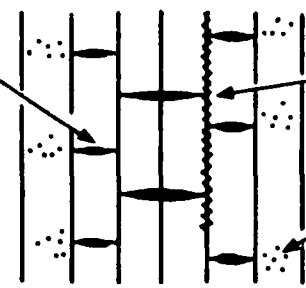
↓

LIFE


Critical Element
Element Properties and Response
Element Stress State
Failure Function
Damage Modes
Element Characterizations
Degradation Relations
Analytical Models
Internal Stress Redistribution
Stiffness Measurements
Residual Strength Reduction
Life

MECHANICS

1. Stress Redistribution



2. Failure Criterion

tensile fracture $\frac{\sigma_1^2}{X^2} + \frac{\sigma_2^2}{Y^2} - \frac{\sigma_1 \sigma_2}{X^2} + \frac{\tau^2}{S^2} = F$ etc.

buckling failure $\frac{N_x}{N_{xc}} = F$, $N_{xc} = N_{xc}(D_{ij}, a, b, m, n)$ etc.

stress concentration $\frac{K}{K_c} = F$, $K_c = \sigma_0 \sqrt{\pi a_0 \left(\frac{c}{2c + a_0}\right)}$ etc.

delamination induced tensile failure $\frac{\epsilon}{\epsilon_c} = \frac{E_c}{E} = F$ etc.

Phenomenological

1. Critical element degradation rules $f\left(\frac{n}{N}\right)$
   i. fatigue behavior of 0 degree plies

$$N = \log^{-1}\left[\frac{B}{A - S/S_u}\right]^{1/x} \qquad S = S(n)$$

   ii. flaw growth $\frac{da}{dn} = C_1 \mathcal{G}^b \qquad \mathcal{G} = \mathcal{G}(n)$

2. Form of residual strength equation:

   change in residual strength =

$$(1 - F^r) = \int (1 - F^L)\, f\left(\frac{n}{N}\right) d\left(\frac{n}{N}\right)$$

F^r = normalized residual strength

F^L = local value of failure function for redistributed stress state

SCENARIO:

TENSION - TENSION, NO SIGNIFICANT DELAMINATION EFFECT:

DAMAGE AND ΔE $\Rightarrow$ LOCAL STRESS STATE*

ANTICIPATED FAILURE MODE

NORMALIZED CHANGE IN RESIDUAL STRENGTH

CRITICAL ELEMENT DEGRADATION EQUATION

LOCAL STRESS STATE FROM ΔE MEASUREMENT*

$$\left(1 - F^r(n)\right) = \int_0^1 \left(1 - F^L(n)\right) i \left(\frac{n}{N}\right)^{i-1} d\left(\frac{n}{N}\right)$$

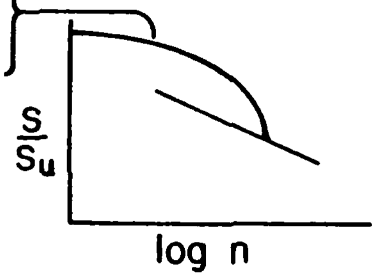
FAILURE FUNCTION FOR CRITICAL ELEMENT O DEGREE PLIES

LIFE RELATIONSHIP FOR O DEGREE PLIES

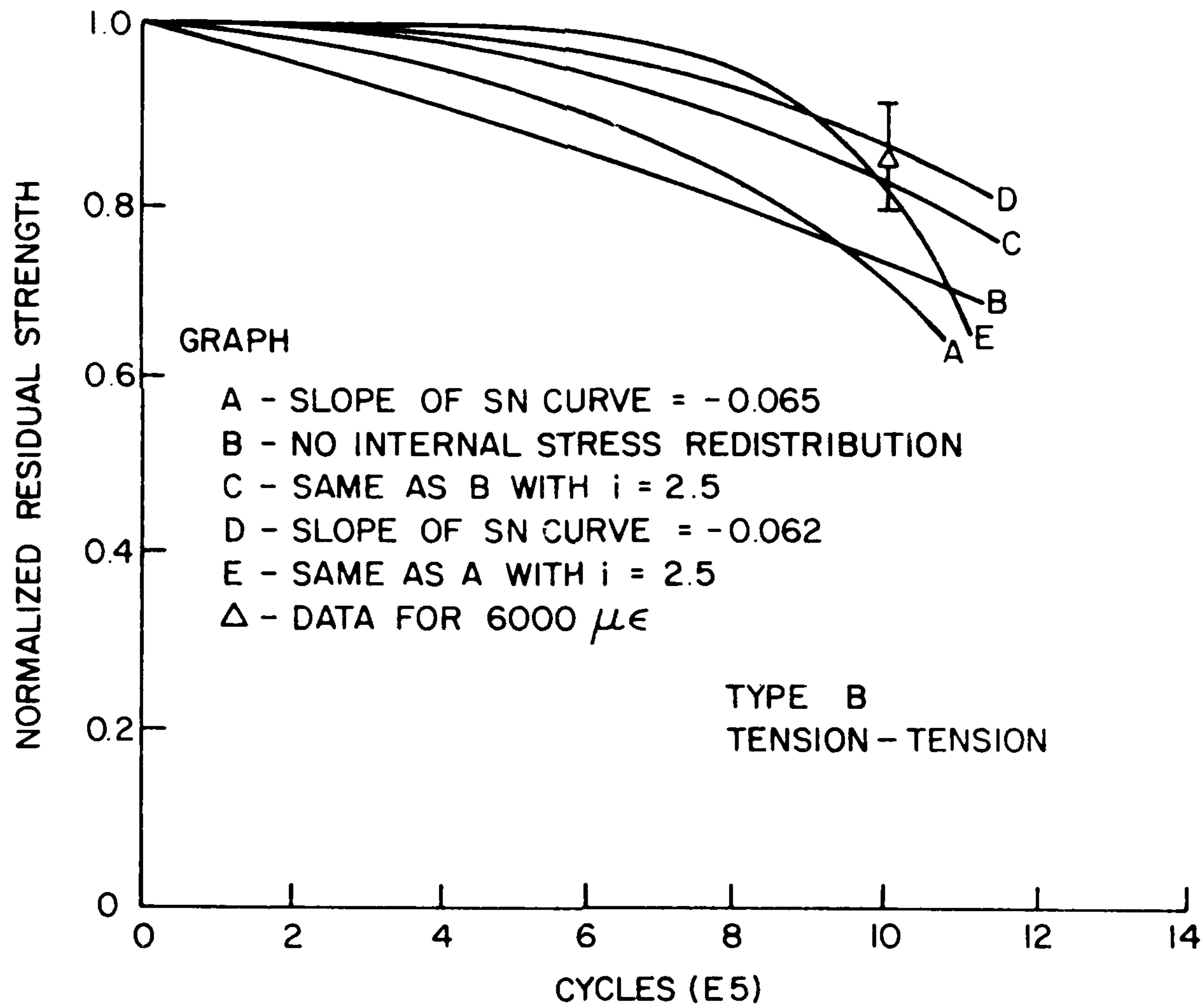
$$N = \log^{-1}\left[\frac{B}{\frac{S(n)}{S_u} - A}\right]^{1/x}$$

$$F(n) = \frac{\sigma_1^2(n)}{X^2} + \frac{\sigma_2^2(n)}{Y^2} - \frac{\sigma_1(n)\sigma_2(n)}{X^2} + \frac{\tau^2(n)}{S^2}$$

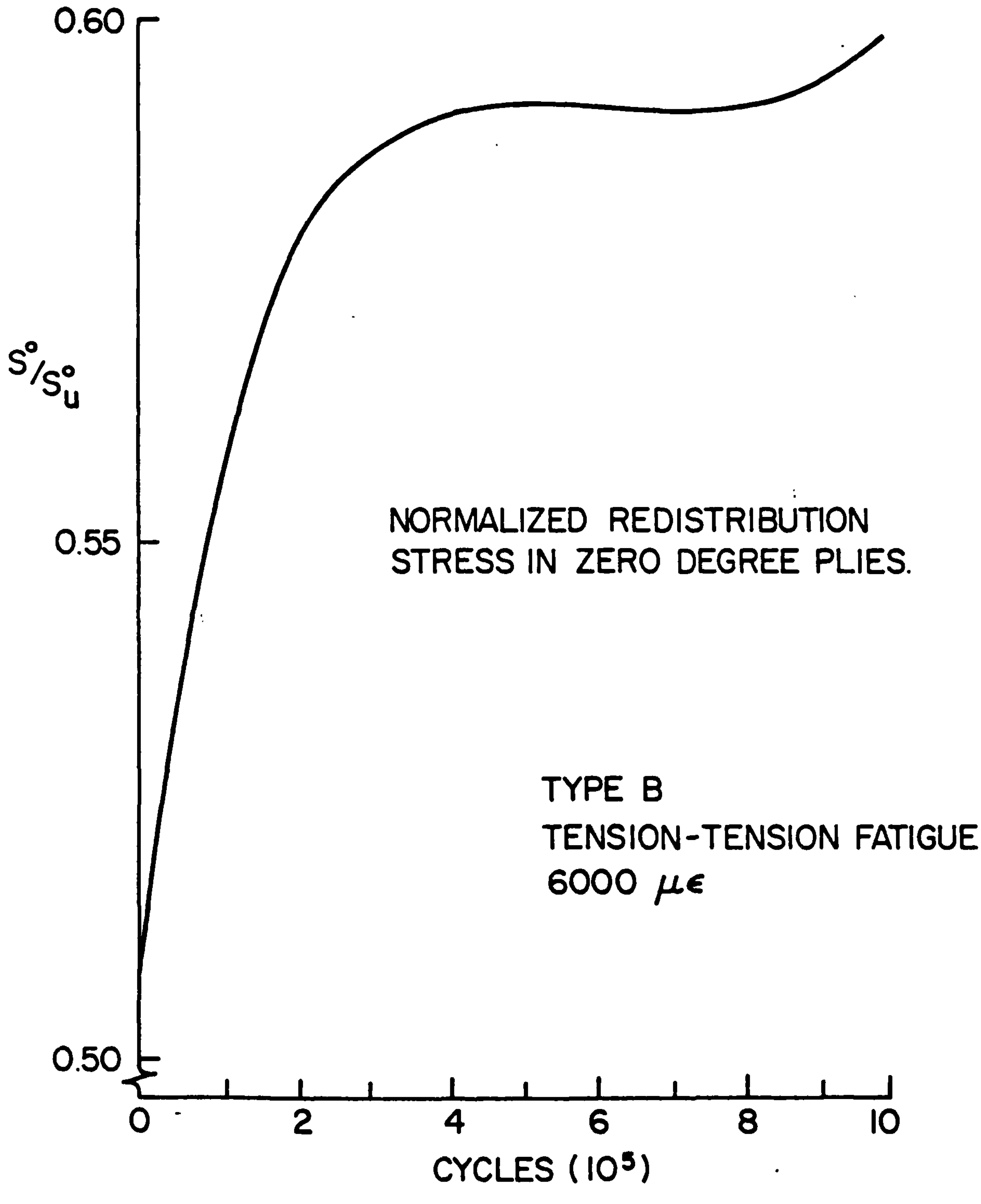
SHAPE OF RESIDUAL STRENGTH CURVE FOR O DEGREE PLIES

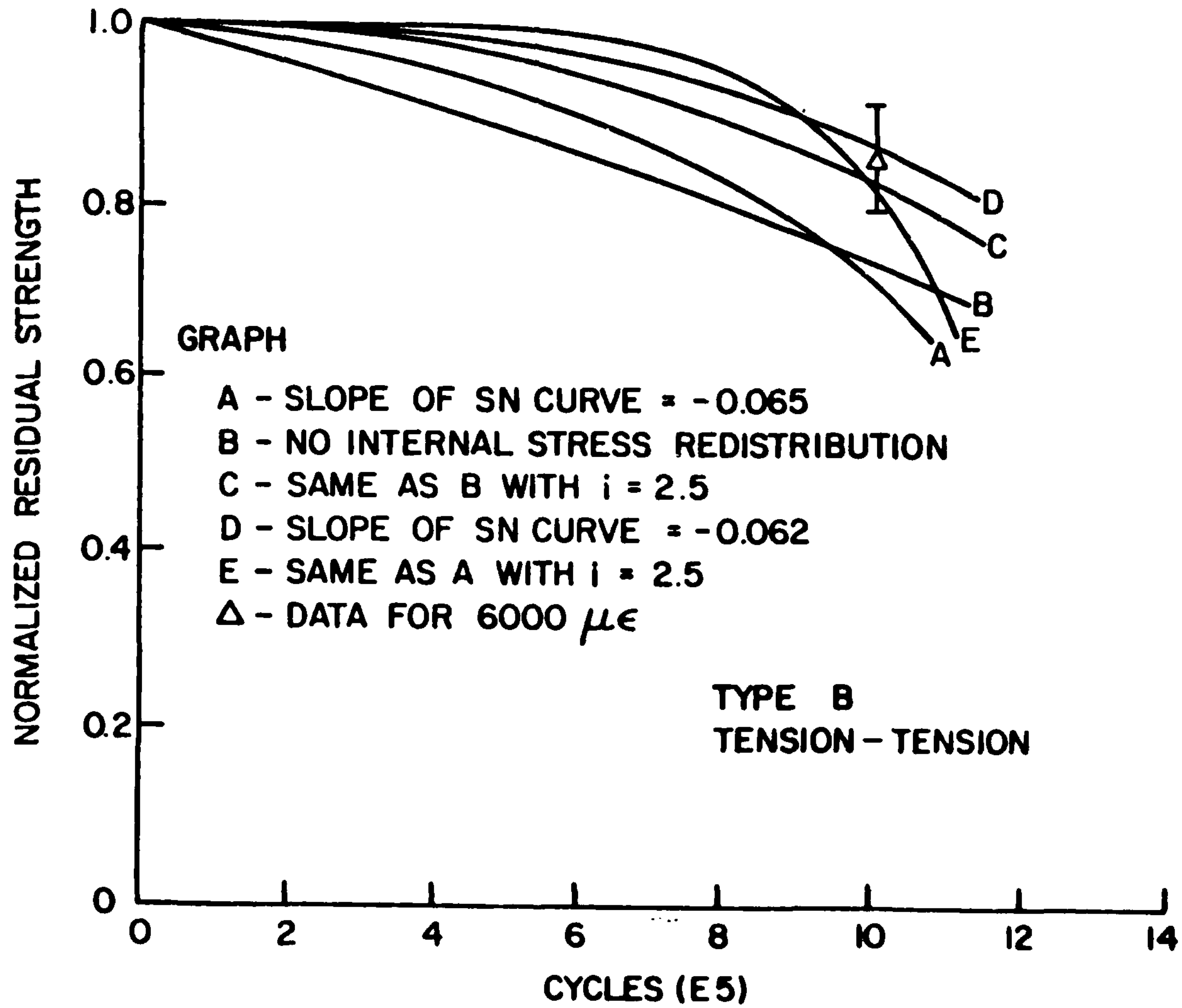


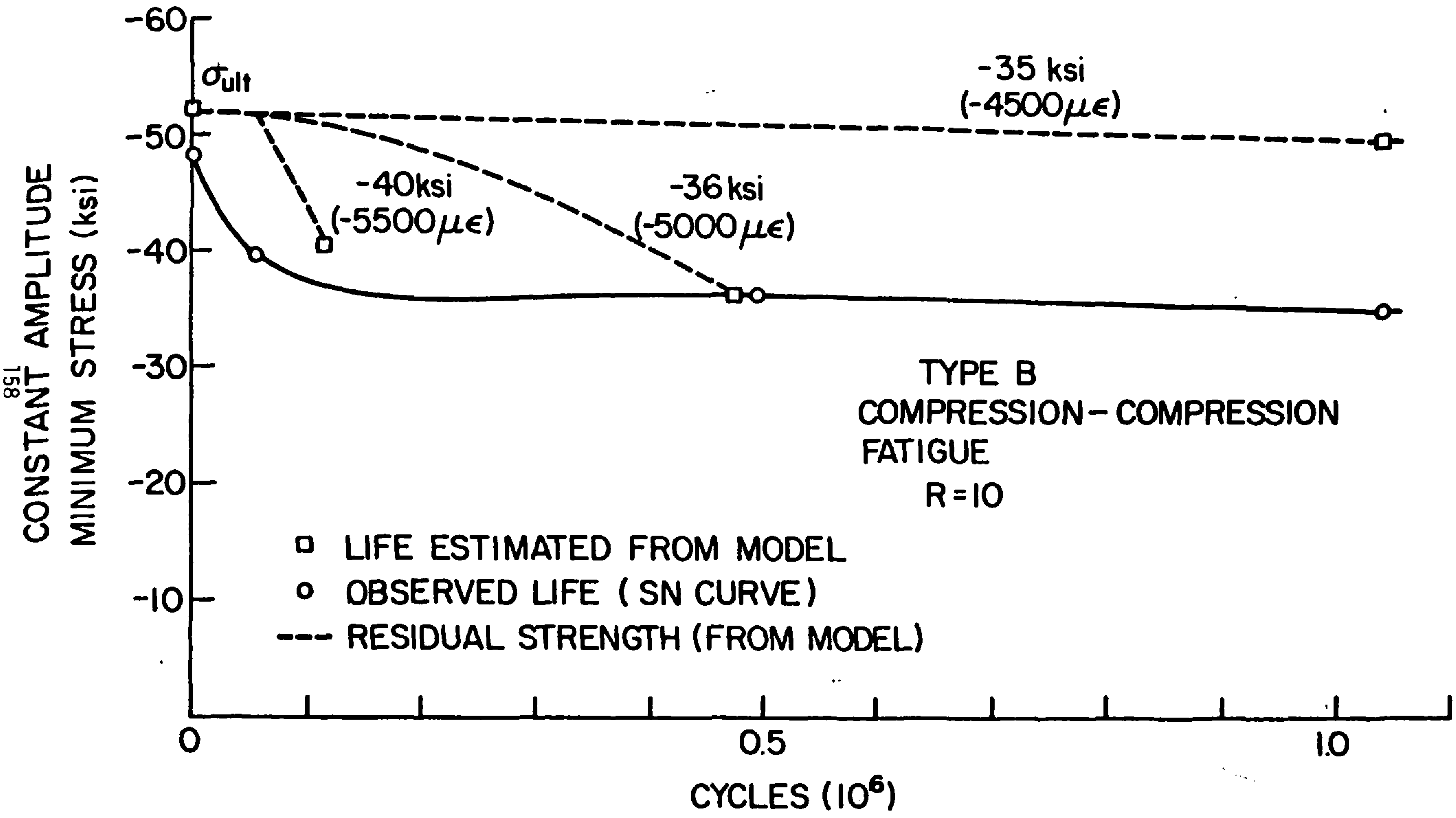
FROM <u>MECHANICS</u> MODELS OF DAMAGE



NORMALIZED RESIDUAL STRENGTH
1.0
0.8
0.6
0.4
0.2
0
0
2
4
6
8
10
12
14
CYCLES (E5)
D
C
B
E
A
GRAPH
A - SLOPE OF SN CURVE = -0.065
B - NO INTERNAL STRESS REDISTRIBUTION
C - SAME AS B WITH i = 2.5
D - SLOPE OF SN CURVE = -0.062
E - SAME AS A WITH i = 2.5
Δ - DATA FOR 6000 με
TYPE B
TENSION - TENSION


0.60
0.55
0.50
S^{o}/S_{u}^{o}
NORMALIZED REDISTRIBUTION
STRESS IN ZERO DEGREE PLIES.
TYPE B
TENSION-TENSION FATIGUE
6000 $\mu\epsilon$
0
2
4
6
8
10
CYCLES (10^5)


1.0
0.8
0.6
0.4
0.2
0
NORMALIZED RESIDUAL STRENGTH
0
2
4
6
8
10
12
14
CYCLES (E5)
D
C
B
E
A
GRAPH
A - SLOPE OF SN CURVE = -0.065
B - NO INTERNAL STRESS REDISTRIBUTION
C - SAME AS B WITH i = 2.5
D - SLOPE OF SN CURVE = -0.062
E - SAME AS A WITH i = 2.5
Δ - DATA FOR 6000 με
TYPE B
TENSION - TENSION


CONSTANT AMPLITUDE
MINIMUM STRESS (ksi)
-60
-50
-40
-30
-20
-10
σ_{ult}
-35 ksi
(-4500 $\mu\epsilon$)
-40ksi
(-5500 $\mu\epsilon$)
-36 ksi
(-5000 $\mu\epsilon$)
TYPE B
COMPRESSION - COMPRESSION
FATIGUE
R = 10
LIFE ESTIMATED FROM MODEL
OBSERVED LIFE (SN CURVE)
RESIDUAL STRENGTH (FROM MODEL)
0
0.5
1.0
CYCLES (10^6)

FATIGUE DAMAGE-STRENGTH RELATIONSHIPS IN COMPOSITE LAMINATES

K. L. Reifsnider, W. W. Stinchcomb, E. G. Henneke, II,
J. C. Duke, Jr. and R. D. Jamison

Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061

Objective: The objective of this program shall be to determine the nature of damage events, caused by cyclic (fatigue) loading, which are directly related to the fracture of composite laminates and to develop an understanding of how such damage events affect the strength of the laminates.

CONCLUSIONS

1. The reduction of strength and life during long-term fatigue loading occurs because of matrix cracking, delamination and fiber fractures which accumulate during cyclic loading.

2. The major effect of fatigue damage, relative to residual strength and life, is to cause internal local stress redistribution which alters the amount of load carried by the 0 degree ply in that region.

3. Fiber failure is increased by fatigue loading to levels of occurrence which may be two orders of magnitude greater than levels observed during quasi-static loading.

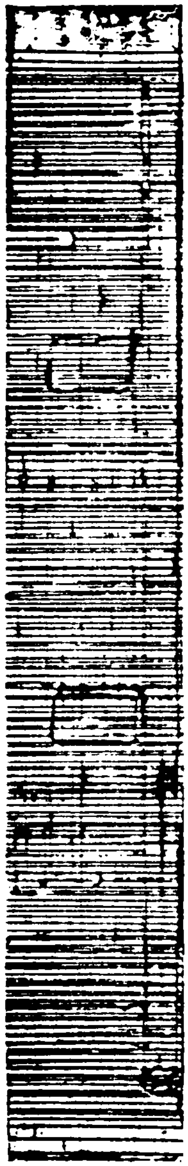
4. Matrix cracks can cause preferential fiber failures.

5. Local (internal) delamination can cause very large stress redistribution in the adjacent plies.

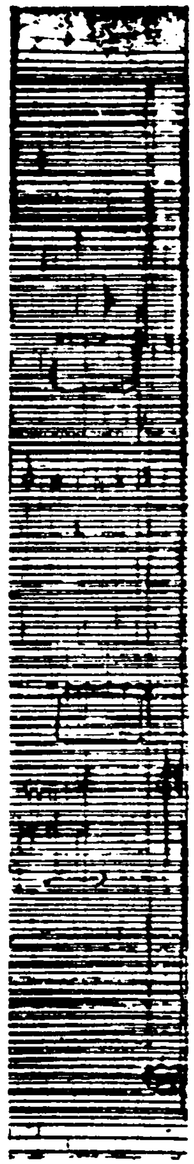
6. The collective condition defined by all extant damage and the current (local) state of stress in the region of subsequent failure defines the residual strength and life of a fatigue-loaded laminate.

7. Fatigue loading greatly influences the process of failure but does not change the basic nature of the final failure event for the fiber-dominated luminates investigated.

8. Changes in laminate stiffness can be used to monitor the development of fatigue damage and can be directly related to internal stress redistribution.

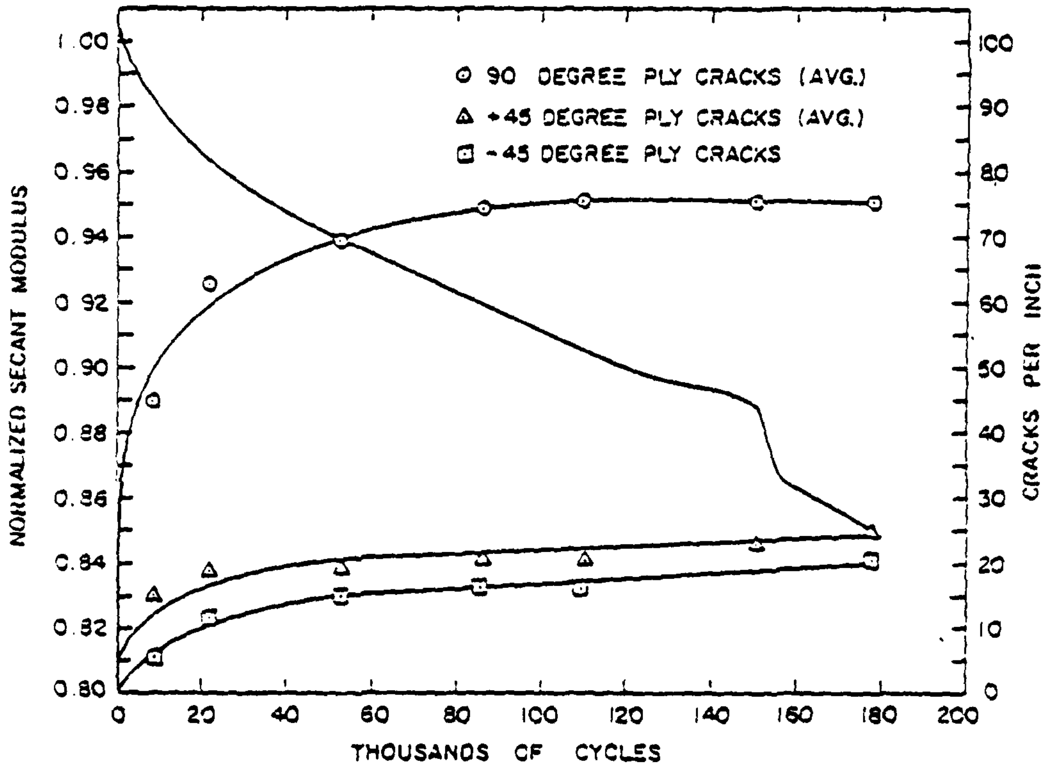


a

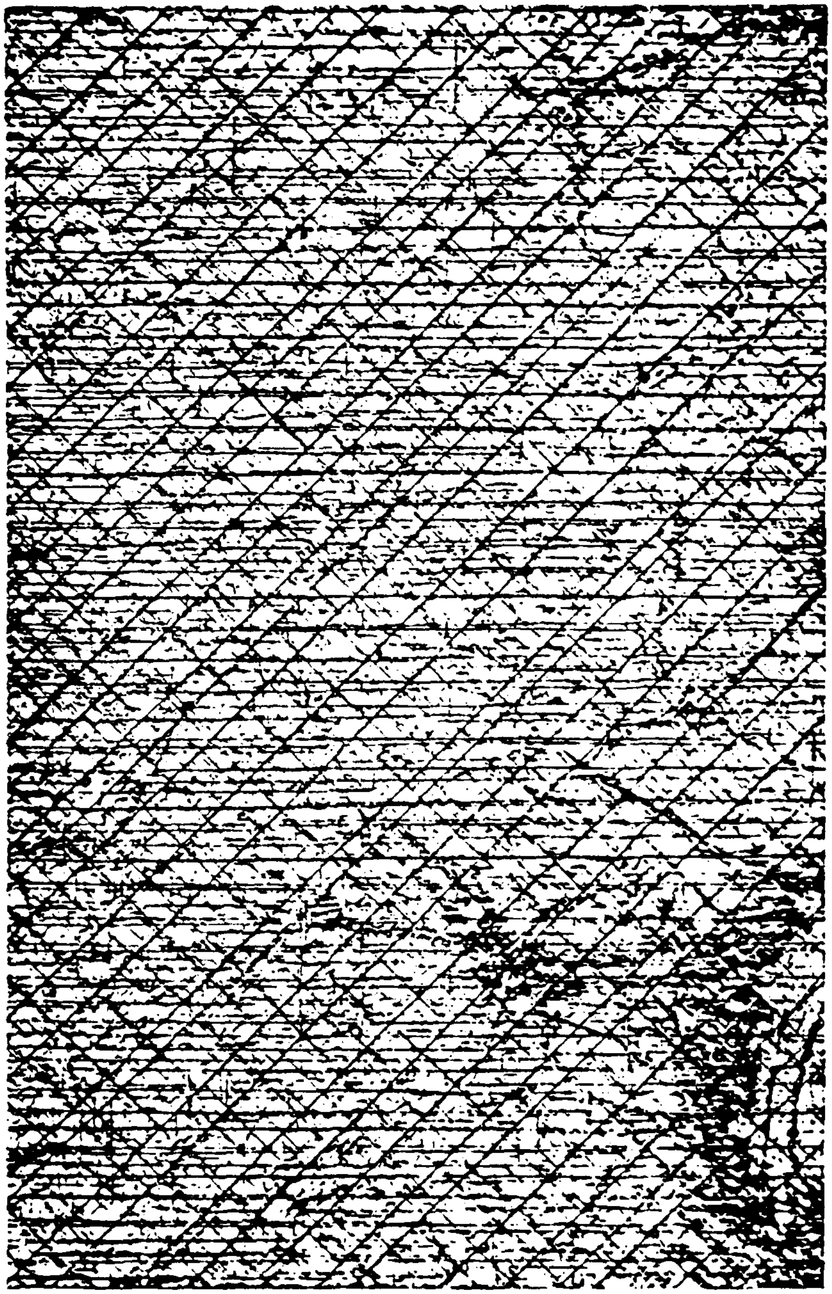


b

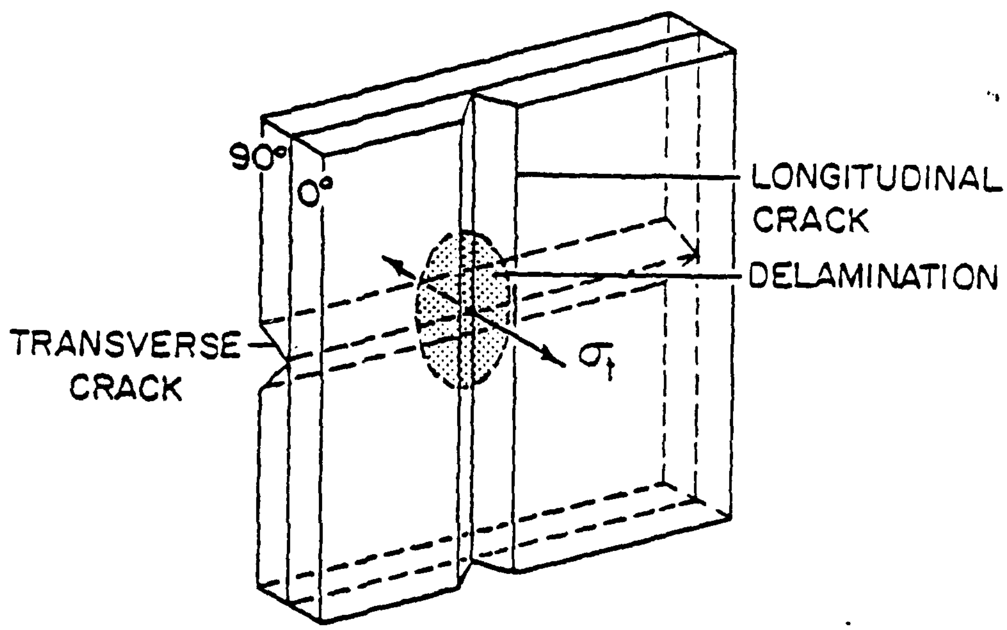
TYPICAL STEREO X-RAY IMAGE PAIR.



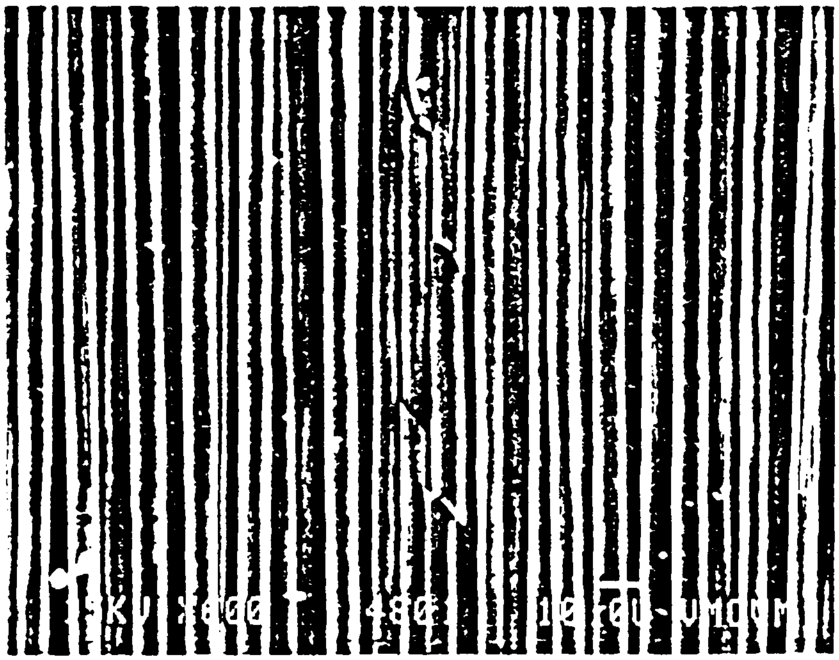
STIFFNESS REDUCTION AND CRACK DEVELOPMENT IN A $[0,90,\pm45]_s$ LAMINATE.



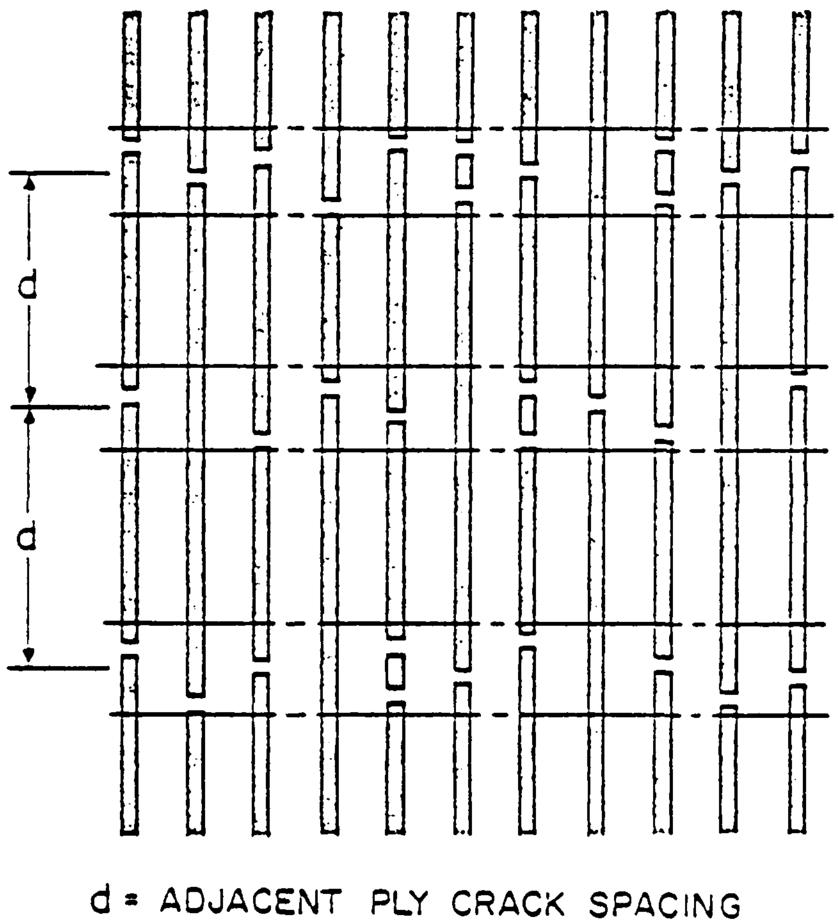
MICRO-DAMAGE IN A $[0,90,\pm45]_S$ LAMINATE.



INFLUENCE OF LONGITUDINAL CRACK ON INTERIOR DELAMINATION FORMATION.



MIXED MODE OF FIBER FRACTURE IN $[0,\pm45]_s$ LAMINATES.



TYPICAL FIBER BREAK PATTERN IN 0 DEGREE PLIES.

TABLE. CALCULATED AND MEASURED LAMINATE STIFFNESS REDUCTION DUE TO PLY CRACKING AND DELAMINATION.

Specimen	Condition	A/A^*	E (GPa)	ΔE_{DEL} (%)	ΔE_{TOTAL} (%)	ΔE_{MEAS} (%)
III-L-1-23	Stage I	0.083	50.5[1]	1.7	7.1	5.2
III-L-1-22	Stage II	0.186	45.8[2]	4.2	15.8	14.7
III-L-1-15	Stage III	0.189	45.5[2]	4.8	16.4	16.7
III-L-1-21	Stage III NEAR FAILURE	0.303	44.2[2]	7.6	18.8	19.7

1. Off-axis ply E_2 and G_{12} discounted 50 percent
2. Off-axis ply E_2 and G_{12} discounted 100 percent

Reference Values:

$E_{UNDAMAGED}$ =	54.4 GPa
$E_{FULLY\ CRACKED}$ =	47.8 GPa
$E_{50\%\ CRACKED\ AND\ DELAMINATED}$ =	40.7 GPa
$E_{FULLY\ CRACKED\ AND\ DELAMINATED}$ =	35.8 GPa

A = Delaminate Area

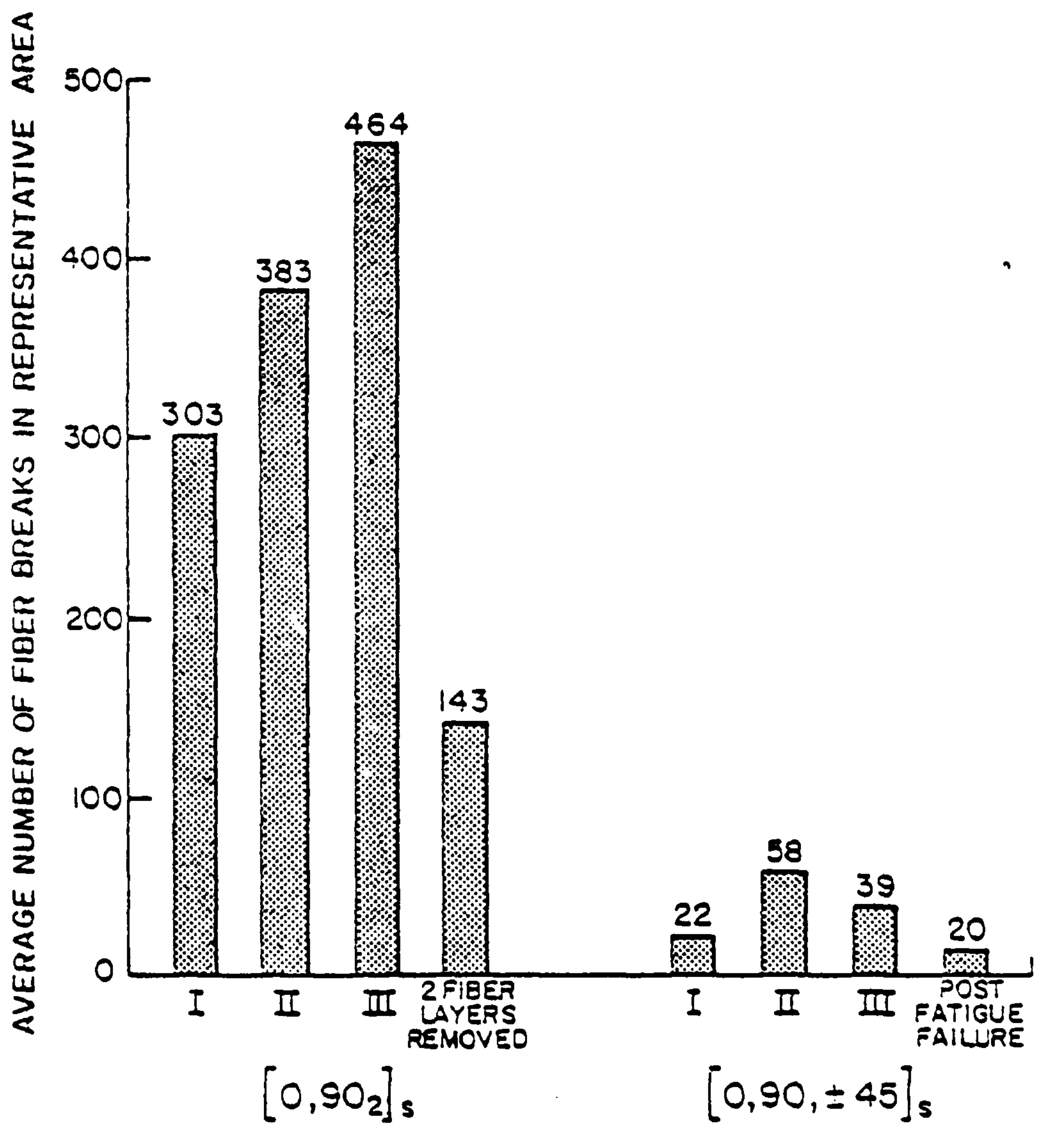
A^* = Total Interfacial Area

E = Laminate Stiffness with Partial Delamination

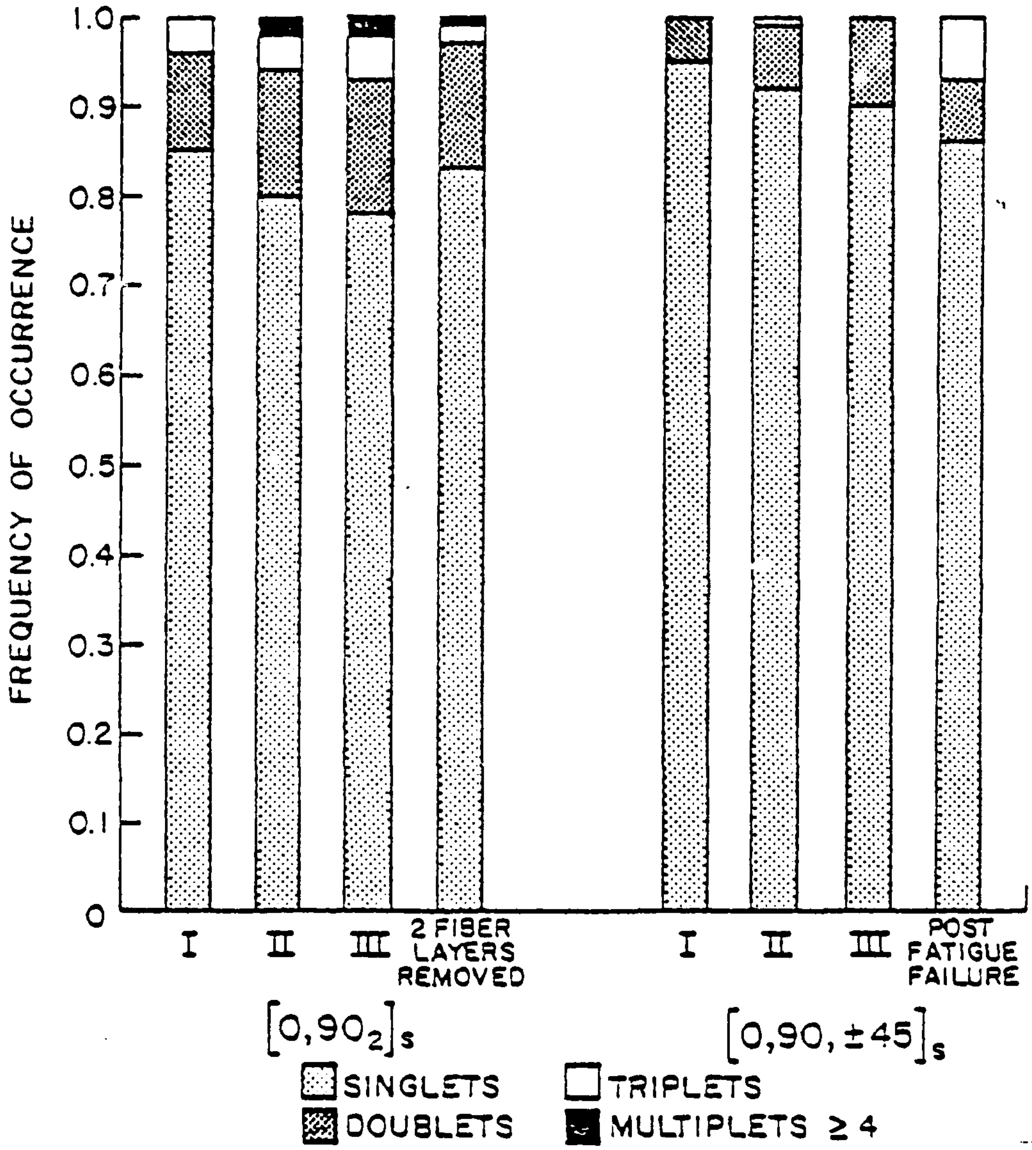
ΔE_{DEL} = Stiffness Reduction due to Dalamination Only

ΔE_{TOTAL} = Stiffness Reduction due to Cracking and Delamination

ΔE_{MEAS} = Measured Stiffness Reduction



TOTAL FIBER FRACTURES IN $[0,90,\pm45]_s$ LAMINATES.



DISTRIBUTION OF MULTIPLETS IN $[0,90,\pm45]_s$ LAMINATES.

Layup and Frequency Effects on Fatigue Life of Composites

NADC sponsor: Lee Gause

MCAIR principal investigator: Charles Saff

GP23-0763-1

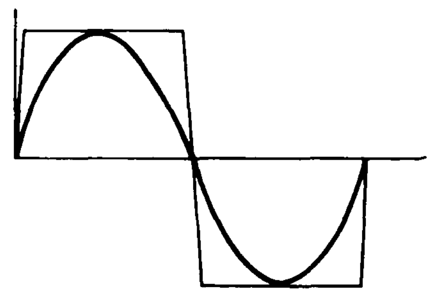
Objectives

- Determine sensitivity of life in graphite/epoxy to load frequency
- Develop an analysis technique for identifying sensitive layups

GP23-0783-2

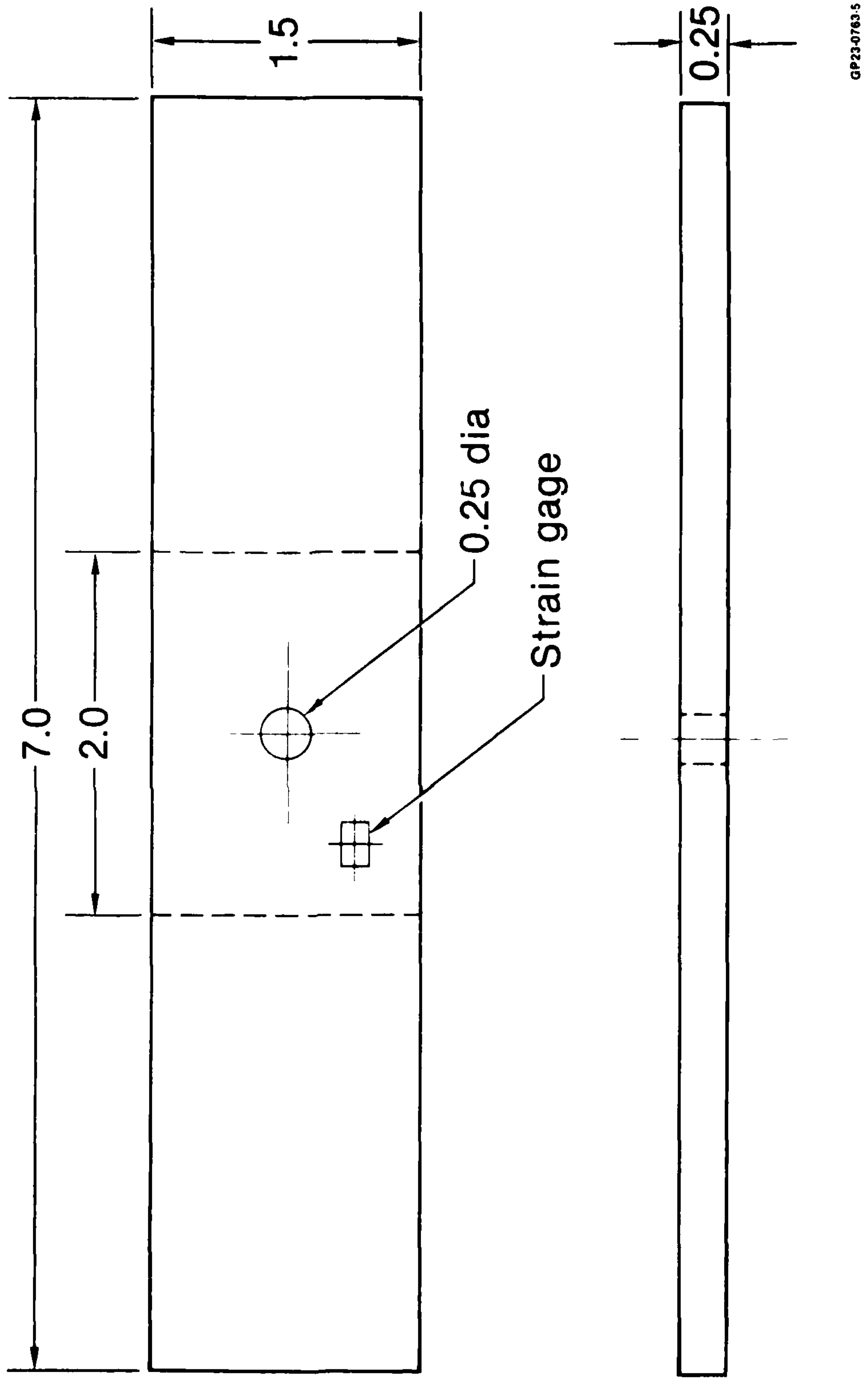
Test Program Summary

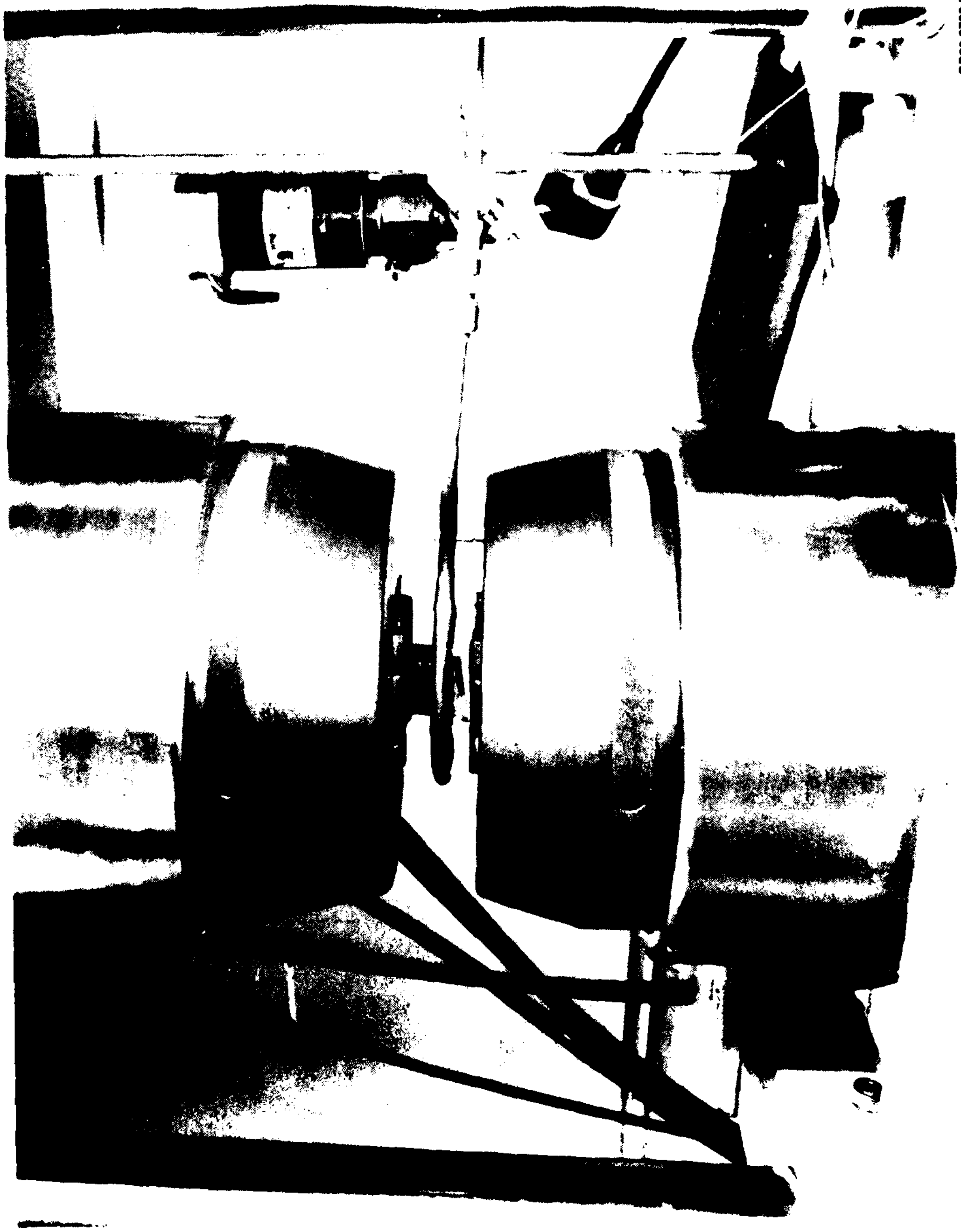
Layup %0°/% ± 45°/%90°	Trap wave	Sine wave	R > 0	Stress level
$[\pm 45°]_{2s}$	X	X	X	
0/100/0	X	X	X	X
4/48/48	X			
4/80/16	X			
16/80/4	X	X		
48/48/4	X	X		
4/32/64	X			
64/32/4	X			
48/48/4*	X			X



Load wave shapes

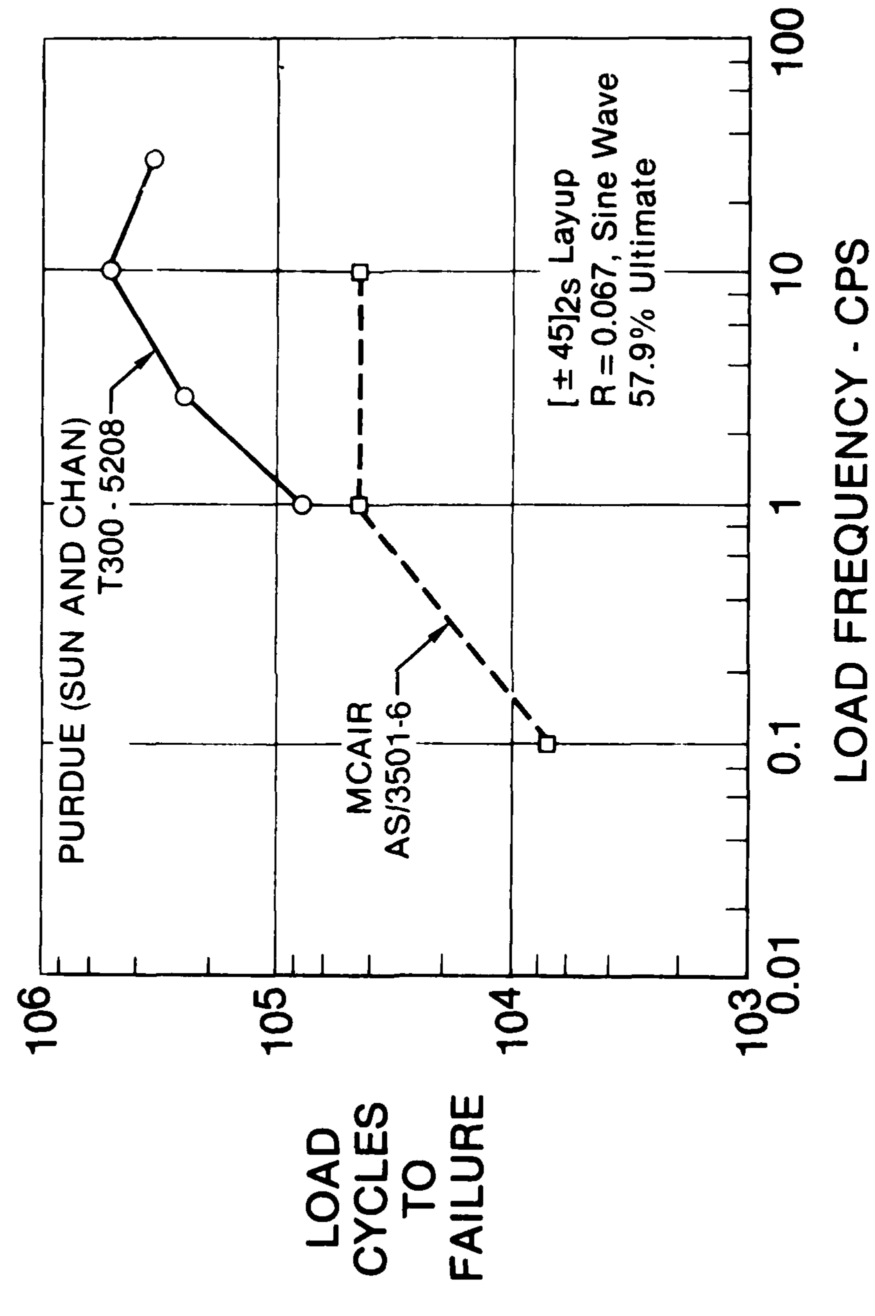
GP23-0763-4


1.5
0.25
7.0
2.0
0.25 dia
Strain gage
GP23-0763-5

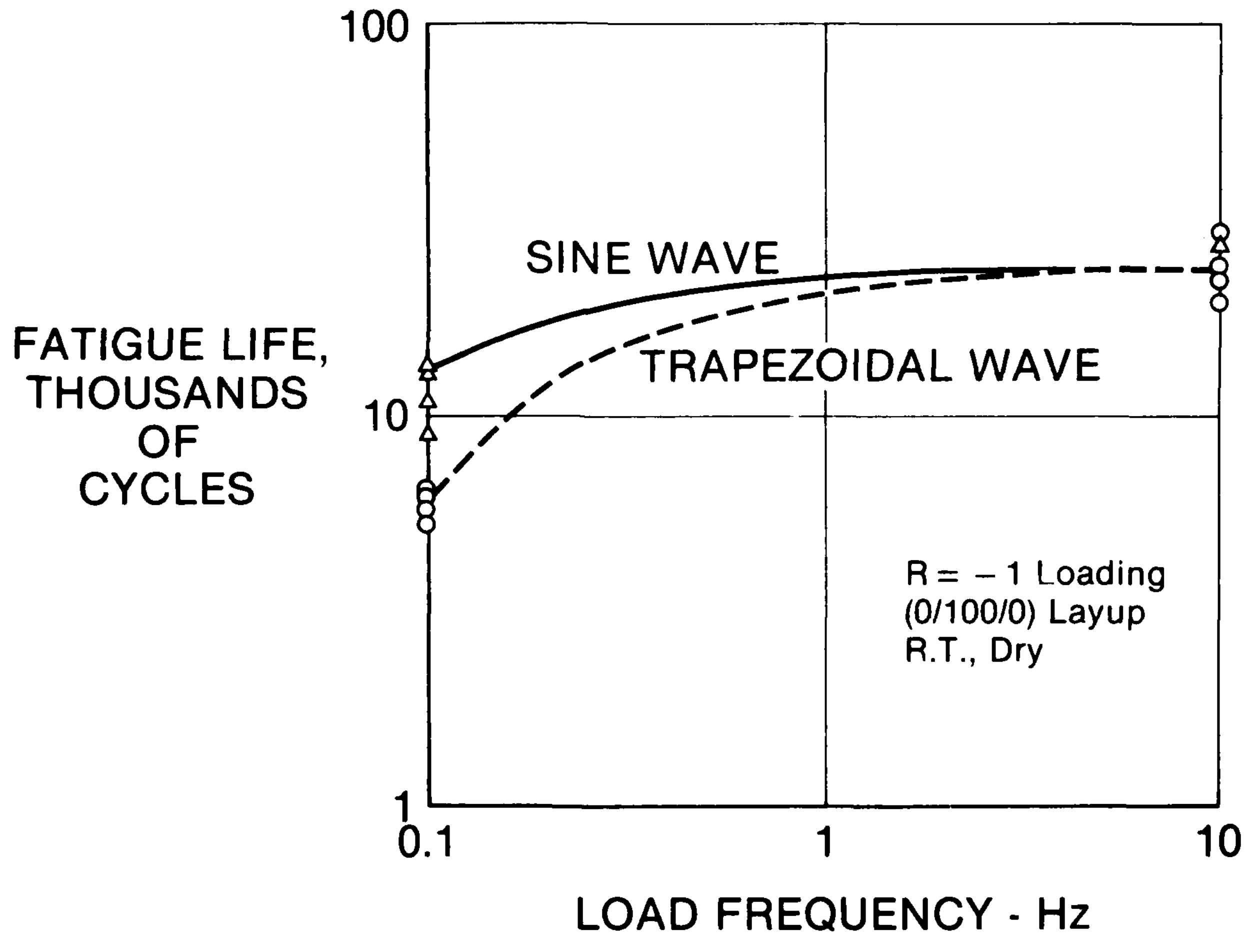


GP21-0783-8

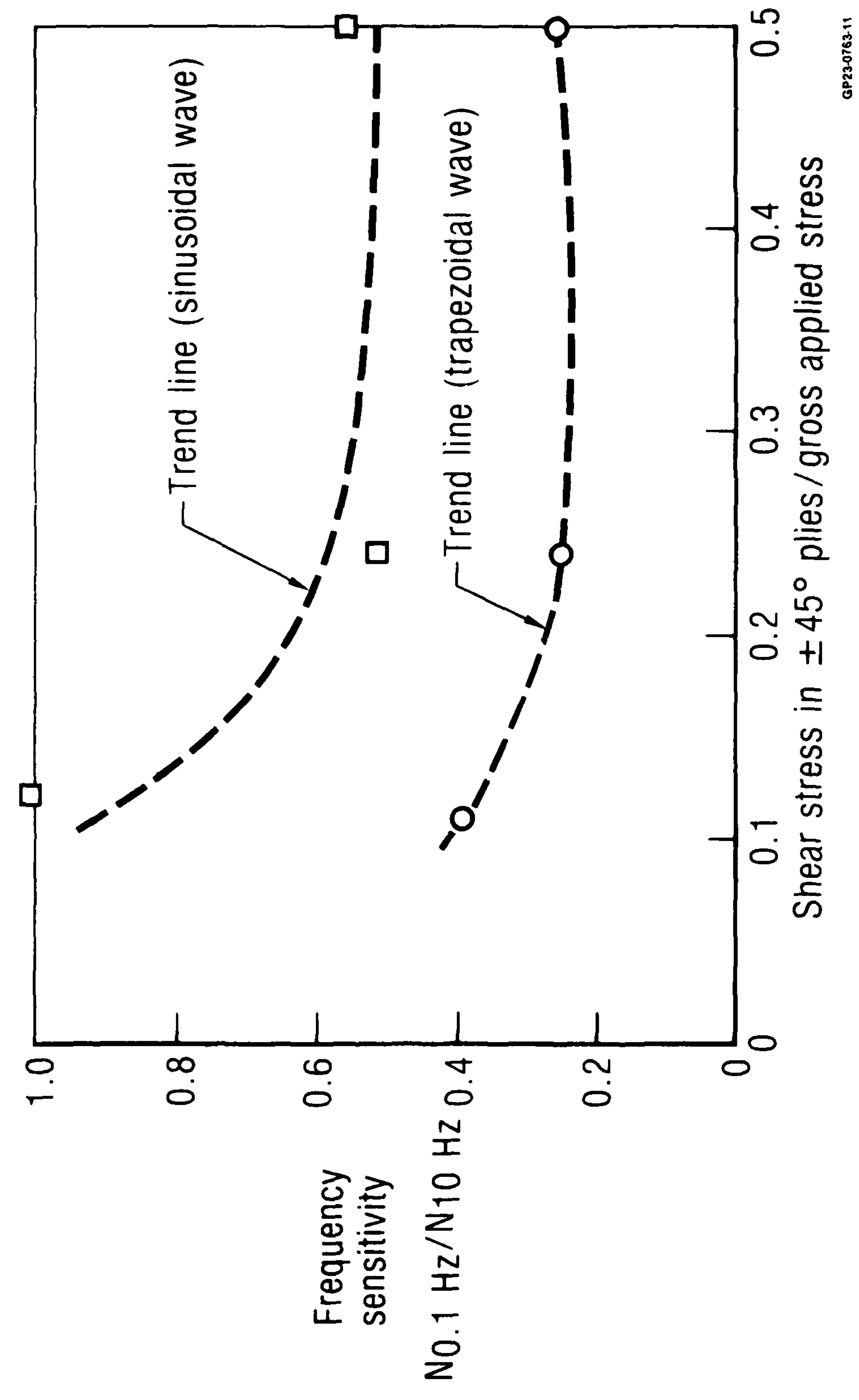
QP23-1053-3

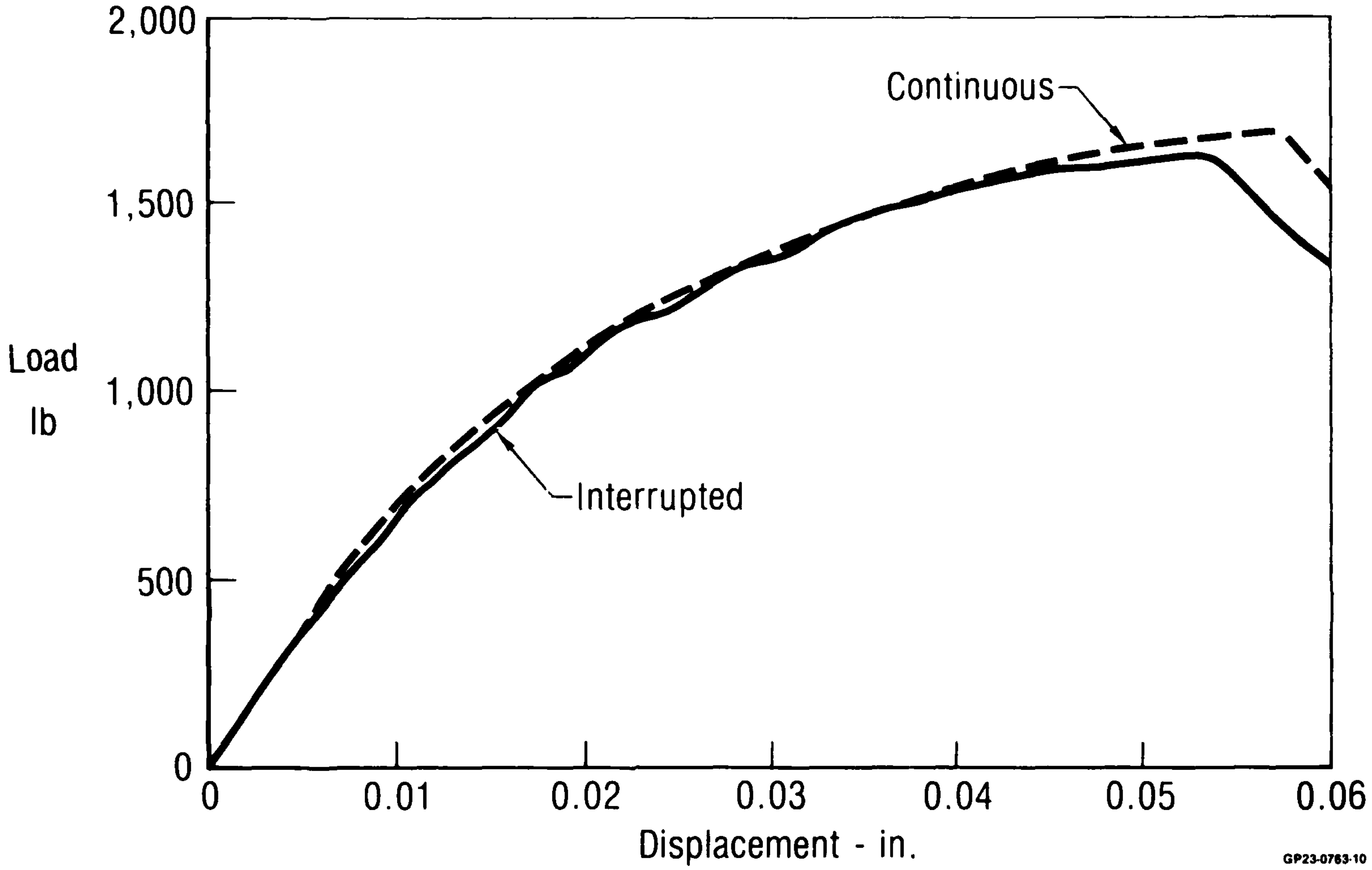


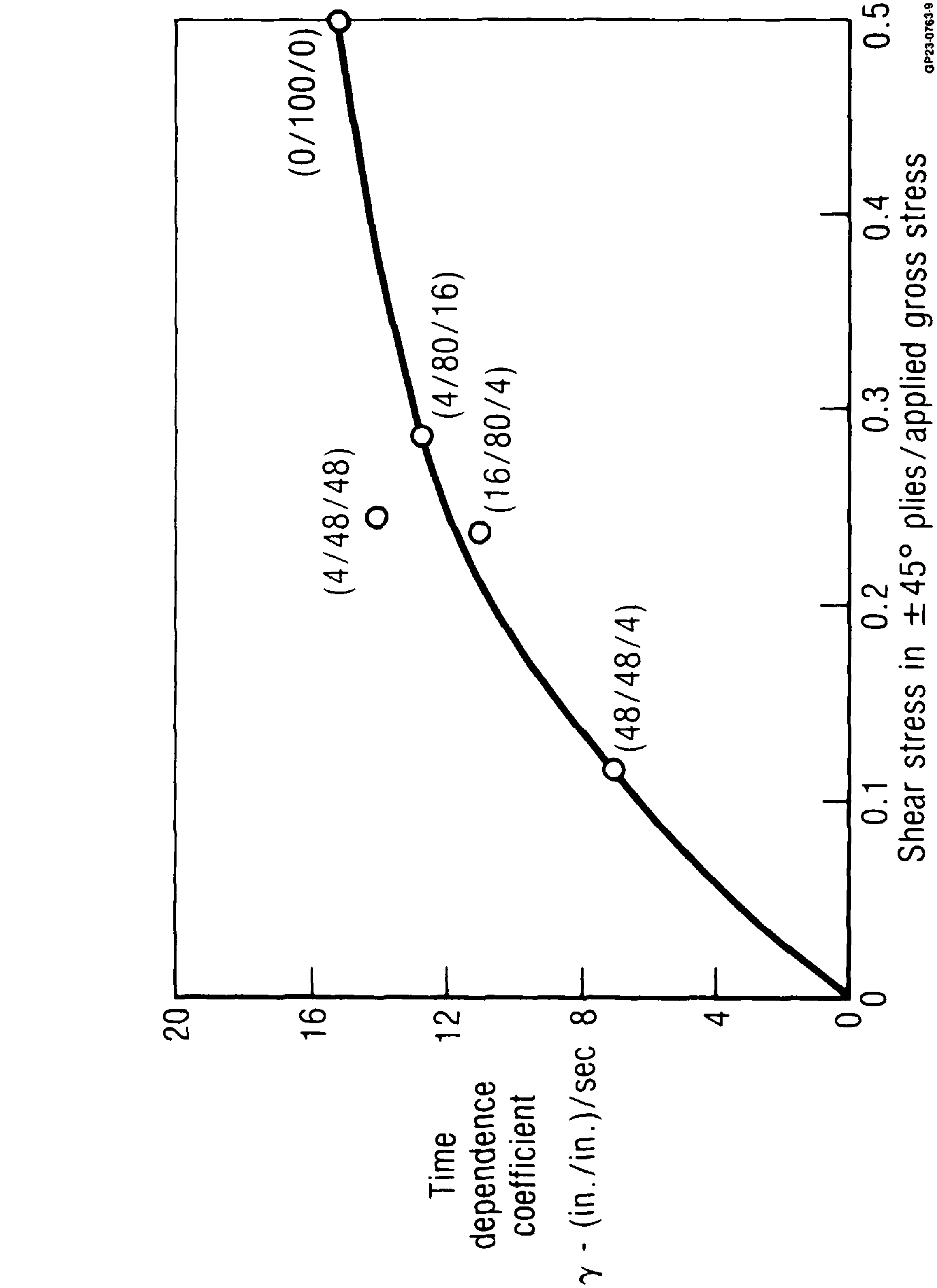
LOAD CYCLES TO FAILURE
10^6
10^5
10^4
10^3
0.01
0.1
1
10
100
LOAD FREQUENCY - CPS
PURDUE (SUN AND CHAN)
T300 - 5208
MCAIR
AS/3501-6
[± 45]2s Layup
R = 0.067, Sine Wave
57.9% Ultimate


FATIGUE LIFE, THOUSANDS OF CYCLES
100
10
1
0.1
1
10
SINE WAVE
TRAPEZOIDAL WAVE
R = −1 Loading
(0/100/0) Layup
R.T., Dry
LOAD FREQUENCY - Hz

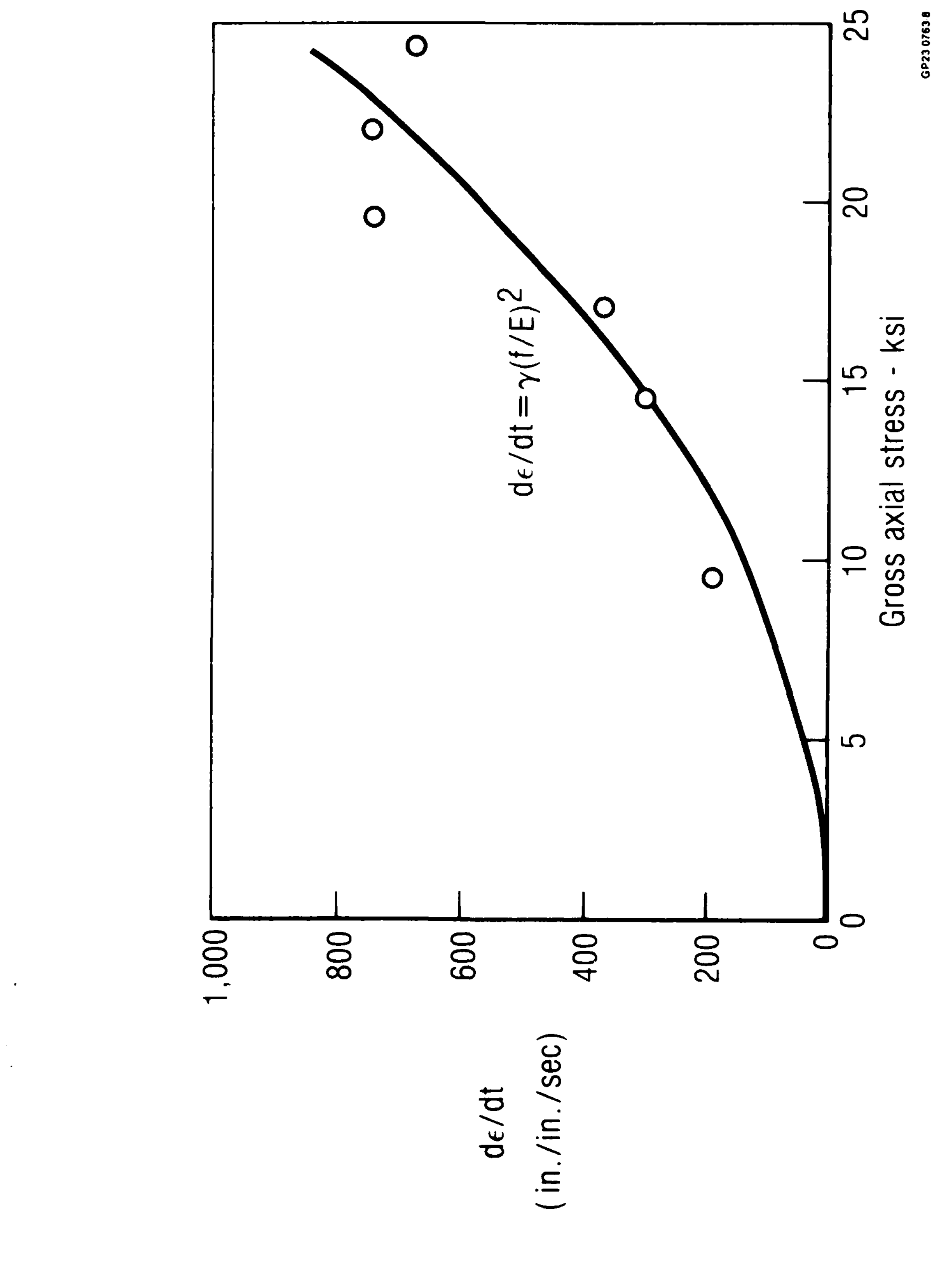
GP23-1053-2

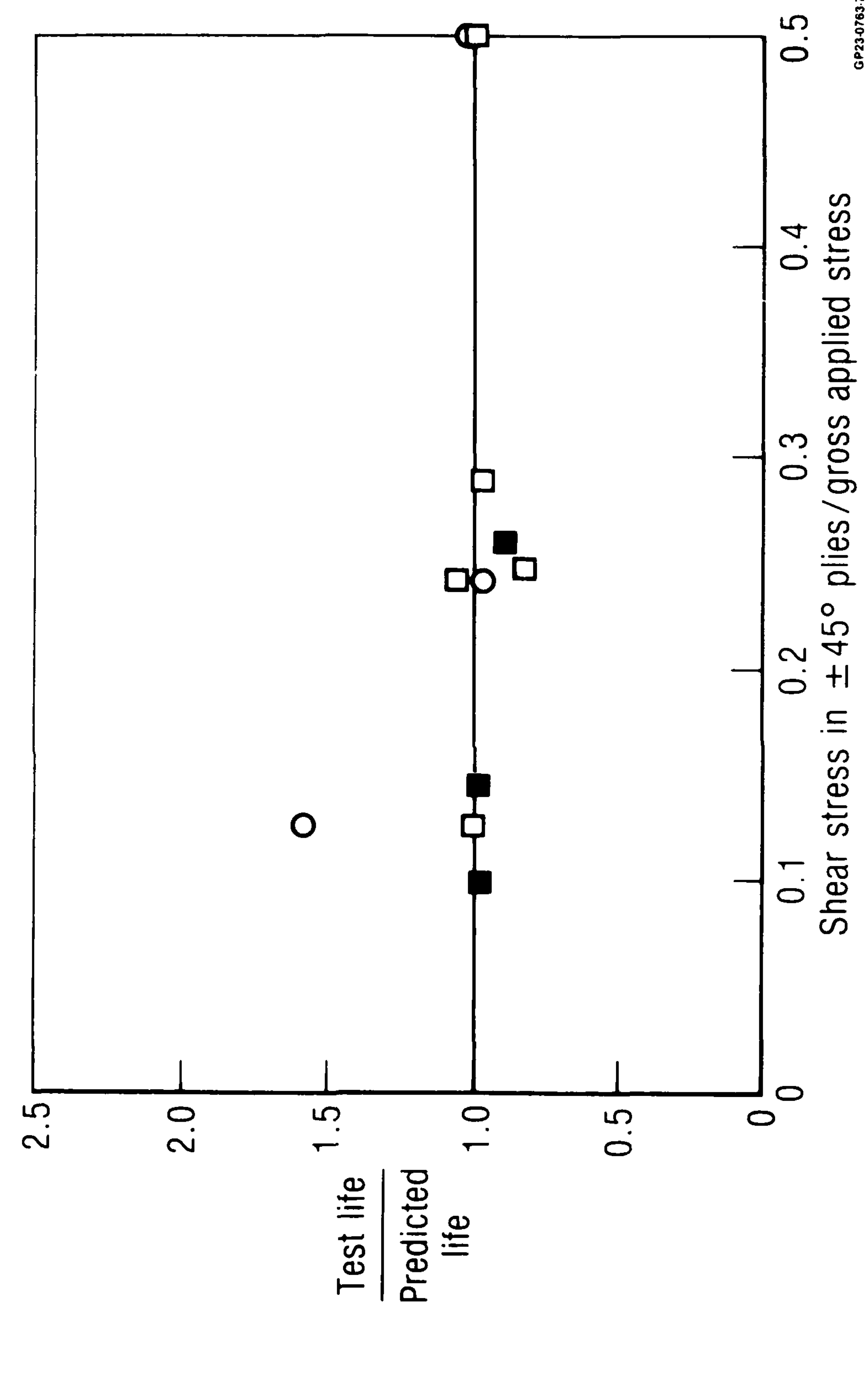

Trend line (sinusoidal wave)
Trend line (trapezoidal wave)
Frequency sensitivity
N0.1 Hz/N10 Hz
1.0
0.8
0.6
0.4
0.2
0
0.1
0.2
0.3
0.4
0.5
Shear stress in ±45° plies/gross applied stress
GP23-0763-11


2,000
1,500
1,000
500
0
Load
lb
Continuous
Interrupted
0
0.01
0.02
0.03
0.04
0.05
0.06
Displacement - in.
GP23-0763-10



Time dependence coefficient
γ - (in./in.)/sec
0
4
8
12
16
20
0
0.1
0.2
0.3
0.4
0.5
Shear stress in ±45° plies/applied gross stress
(0/100/0)
(4/80/16)
(16/80/4)
(4/48/48)
(48/48/4)
GP23-0763-9


dϵ/dt
(in./in./sec)
1,000
800
600
400
200
0
0
5
10
15
20
25
Gross axial stress - ksi
dϵ/dt = γ(f/E)^2
GP23 0763 8

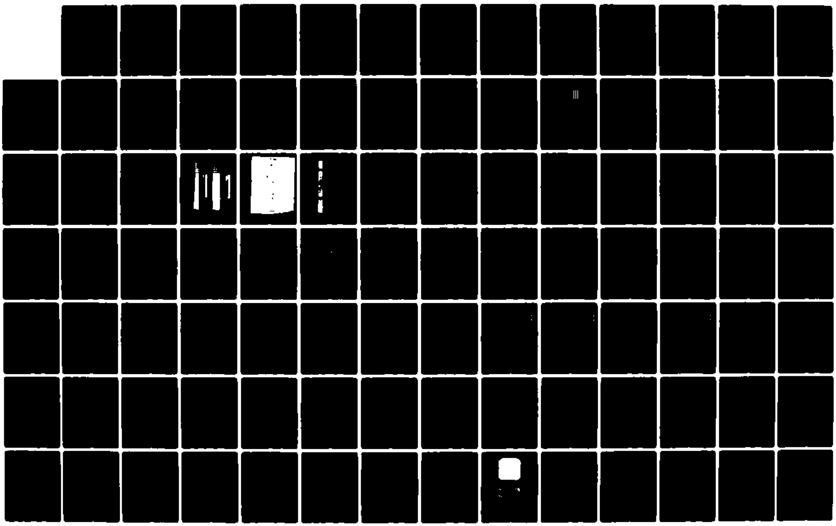

Test life / Predicted life
0
0.5
1.0
1.5
2.0
2.5
0
0.1
0.2
0.3
0.4
0.5
Shear stress in ±45° plies/gross applied stress
GP23-0763-7

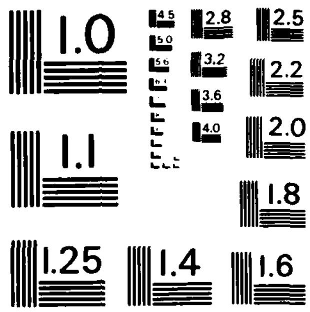
Conclusions

- Lives shorter at 0.1 Hz than at 10 Hz
- Matrix dominated layups more affected than fiber dominated layups
- At $R = -1$, sustained loads reduce life
- Statically, displacements increase with sustained load
- Displacement increases correlated with sensitivity

GP23-0763-3

AD-A130 750 PROCEEDINGS OF THE ANNUAL MECHANICS OF COMPOSITES 3/5
REVIEW (8TH) HELD AT WR..(U) AIR FORCE WRIGHT
AERONAUTICAL LABS WRIGHT-PATTERSON AFB OH L A WILSON
UNCLASSIFIED APR 83 AFWAL-TR-83-4005 F/G 11/4 NL





MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS - 1963 - A

EFFECT OF STRESS RATIO ON FATIGUE LIFE OF COMPOSITES

BY

G. P. SENDECKYJ
STRUCTURES & DYNAMICS DIVISION
FLIGHT DYNAMICS LABORATORY
AF WRIGHT AERONAUTICAL LABORATORIES

OBJECTIVES:

- DEVELOP UNDERSTANDING OF EFFECT OF STRESS RATIO ON FATIGUE LIFE OF RESIN-MATRIX COMPOSITES
- DEVELOP FATIGUE MODEL THAT PROPERLY ACCOUNTS FOR STRESS RATIO DEPENDENCE

FATIGUE MODEL

DAMAGE METRIC

$$\frac{d\sigma}{dn} = -f \cdot (1/S)\, \sigma_a^{1/S}\, \sigma^{1-1/S}$$

DETERMINISTIC FATIGUE EQUATION

$$\sigma_e = \sigma_a \left\{ (\sigma_r/\sigma_a)^{1/S} + f(n-1) \right\}^S$$

STATISTICAL INFORMATION

$$R(\sigma_e) = \exp\left[-(\sigma_e/\beta)^{\alpha} \right]$$

FATIGUE LIFE EQUATION

$$\sigma_e = \sigma_a \left[1 + f(N-1) \right]^S$$

MODEL	S	ξ
W1	S_0	1
W2	S_0	C
W3	S_0	$C(1 - R)^G$
W3A	$S_0\ (1-R)^G$	"
W4	$S_0 + D\ (1 - R)$	"
W4A	$S_0\ (1 - R)^G$	"

WHERE

S_0, C, D, G ARE CONSTANTS

MODEL PARAMETER ESTIMATION PROCEDURE

1. ASSUME MODEL PARAMETERS
2. CONVERT FATIGUE DATA TO STATIC DATA
3. FIT WEIBULL DISTRIBUTION TO STATIC DATA
4. REPEAT STEPS 1 THROUGH 4 UNTIL SHAPE PARAMATER ESTIMATE IS MAXIMIZED

REF: SENDECKYJ, ASTM STP 734, 1981, PP. 245-260

MODEL SELECTION CRITERIA

UNIQUE PARAMETER ESTIMATES

STRESS RATIO DEPENDENCE

MINIMUM NUMBER OF PARAMETERS

HIGHEST SHAPE PARAMETER ESTIMATE

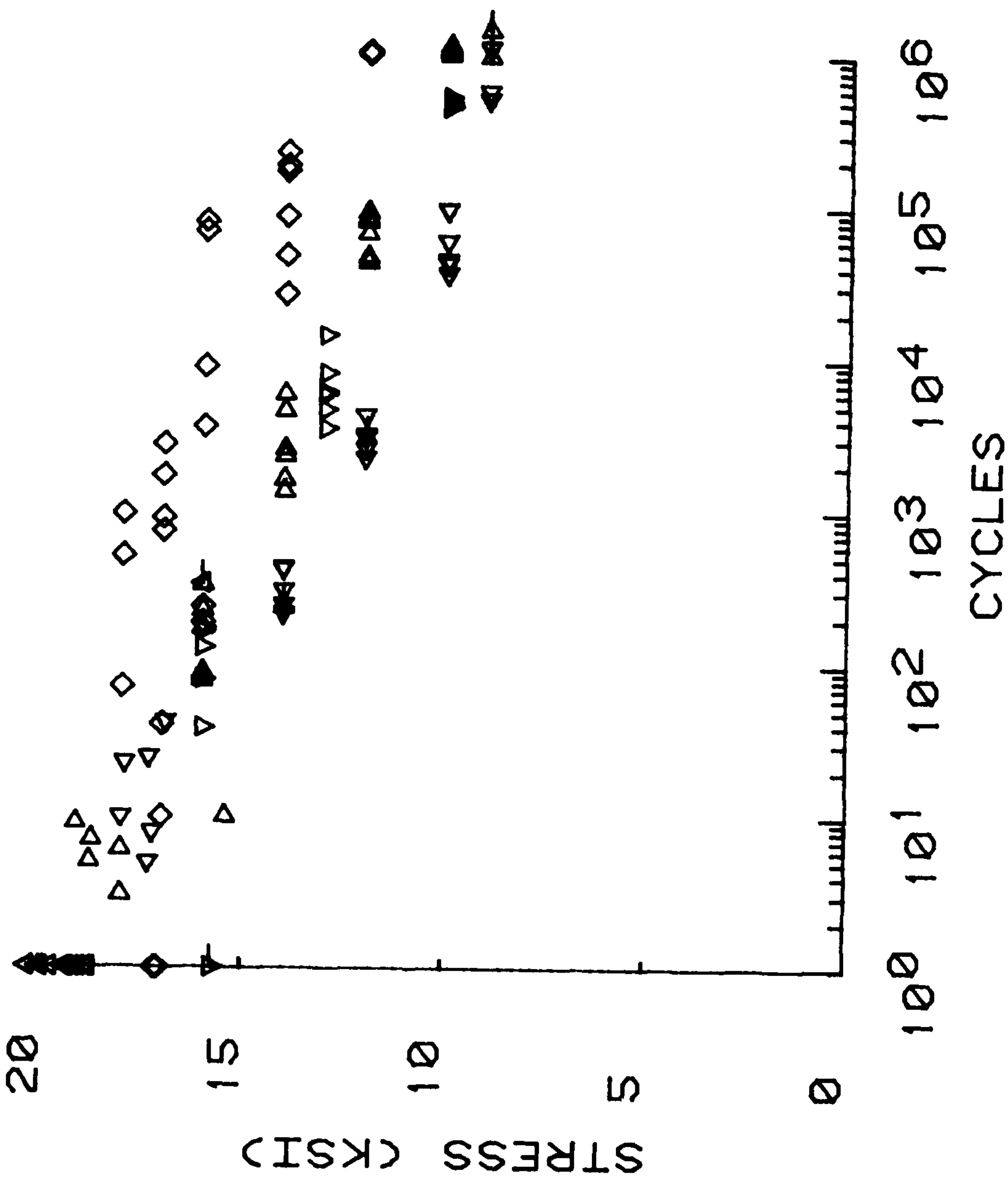
EXPERIMENTAL PROGRAM

LAMINATE	STRESS RATIOS
$(\pm 45)_{2S}$ S2/5208	0.1, 0.25, 0.4, 0.8
$(\pm 45/0_2)_S$ S2/5208	0.1, 0.294, 0.522, 0.791
$(\pm 45/0)_S$ S2/5208	0.1, 0.289, 0.438, 0.556
$(0/\pm 45/90)_S$ S2/5208	0.1, 0.4, 0.7

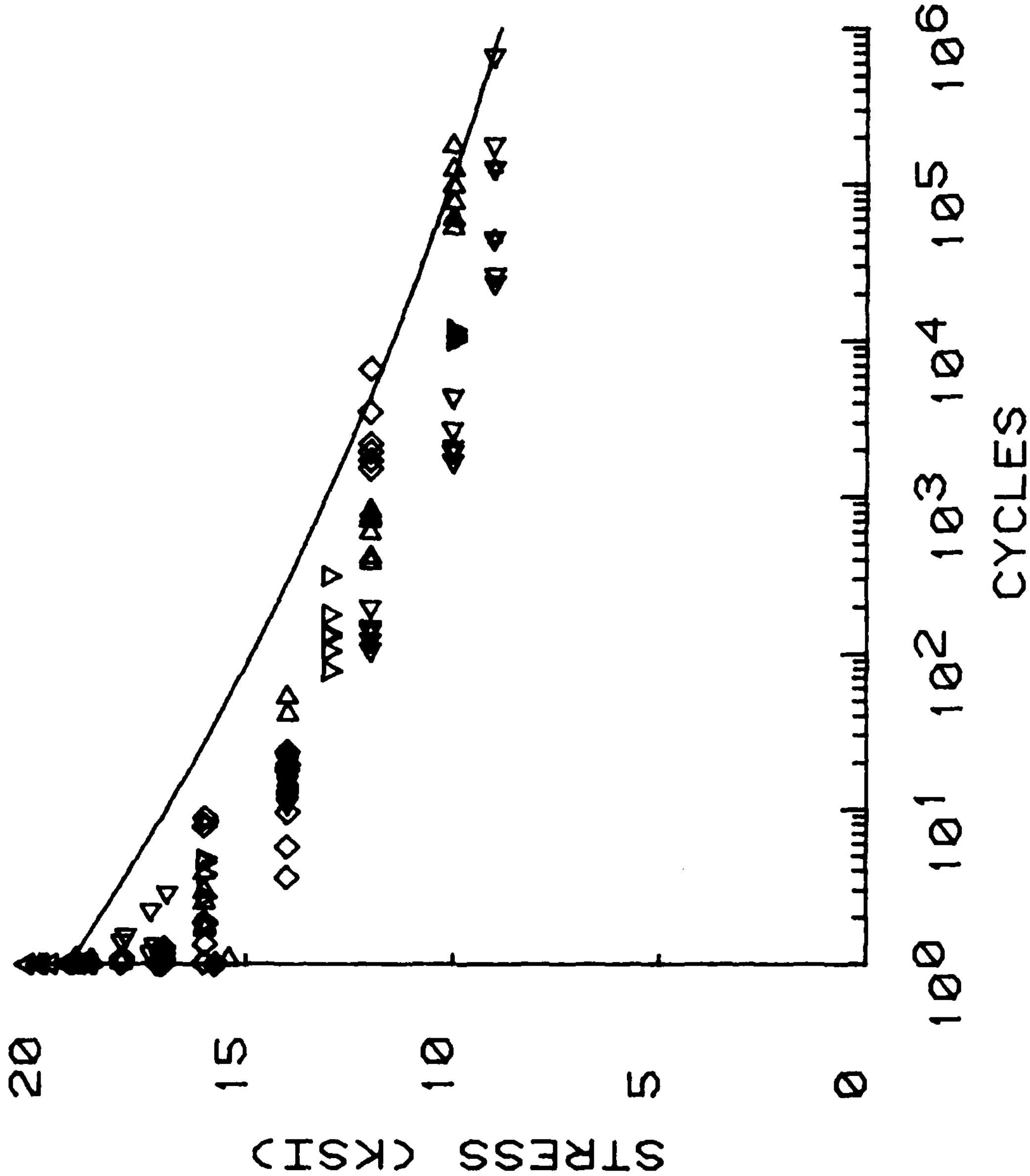
MODEL W2

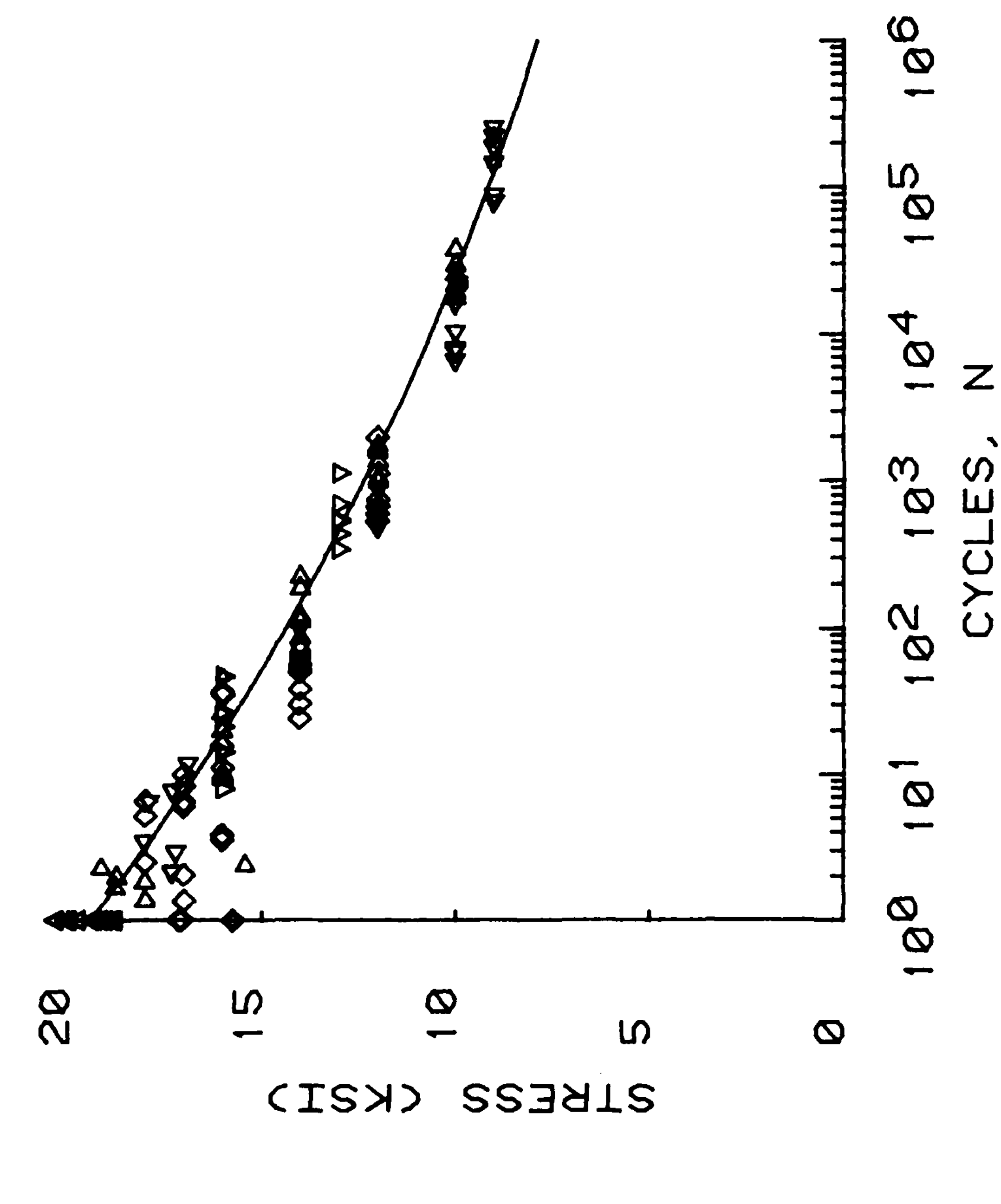
R	S_0	C	α	β
0.1	0.06051	0.4894	27.40	19.68
0.25	0.05996	0.1443	36.54	19.71
0.4	0.05012	0.2984	38.88	19.72
0.8	0.02437	0.1716	22.11	19.29

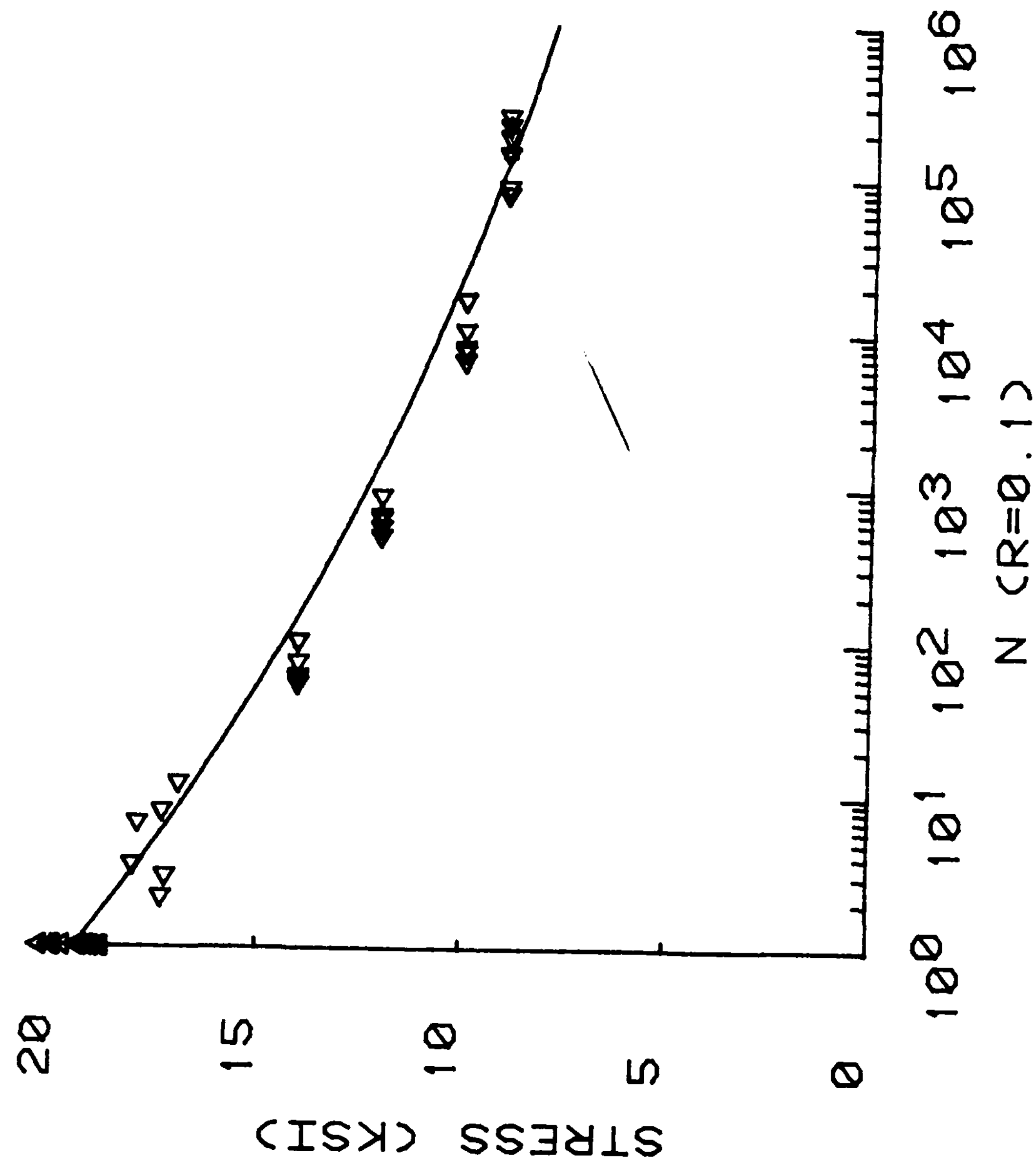
$$\sigma_e = \sigma_a \left[(\sigma_r / \sigma_a)^{1/S_0} + C(n-1) \right]^{S_0}$$

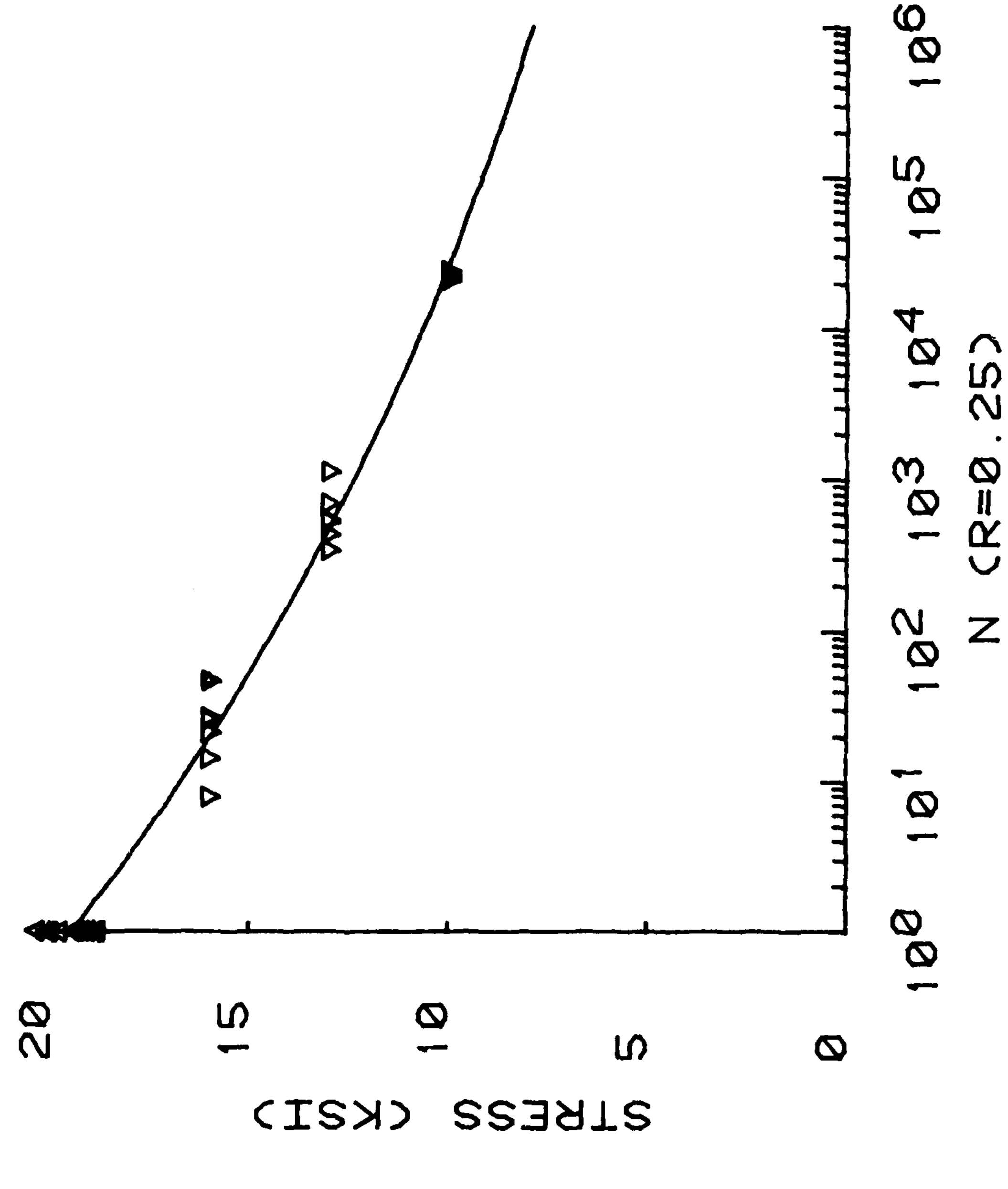

STRESS (KSI)
20
15
10
5
0
CYCLES
10^0
10^1
10^2
10^3
10^4
10^5
10^6

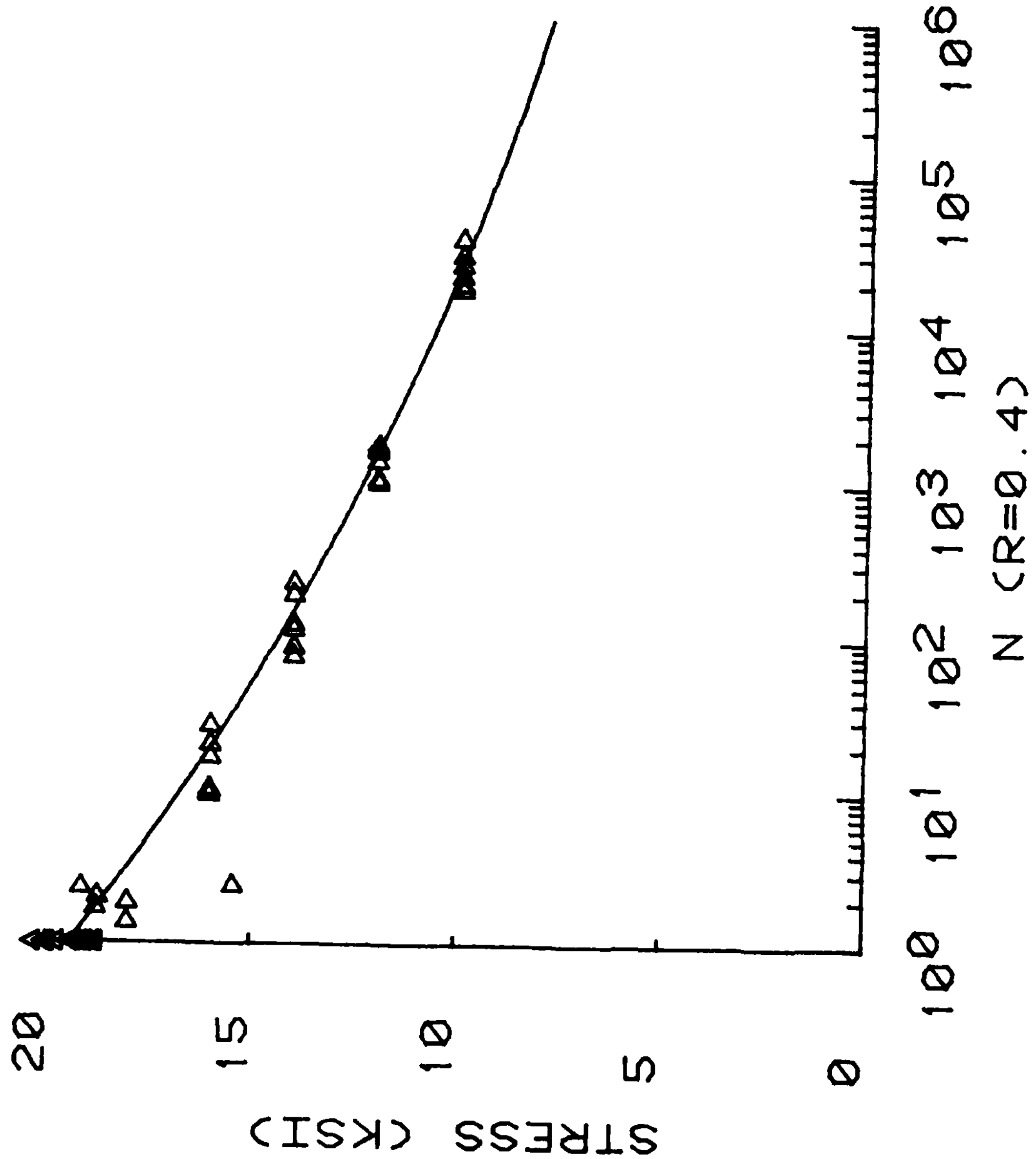
MODEL	PARAMETERS					
	S_o	D	C	D		
W3	0.05643	--------	0.07037	4.090	22.99	19.28
W3A	0.06530	--------	0.3336	0.5836	26.02	19.41
W4	0.04952	0.00936	0.6470	3.745	23.76	19.30
	0.01496	0.05176	0.3611	0.5359	25.56	19.38
W4A	0.06183	0.3419	0.4767	0.3419	25.31	19.35
	0.06530	0.5836	0.3336	0.5836	26.02	19.41

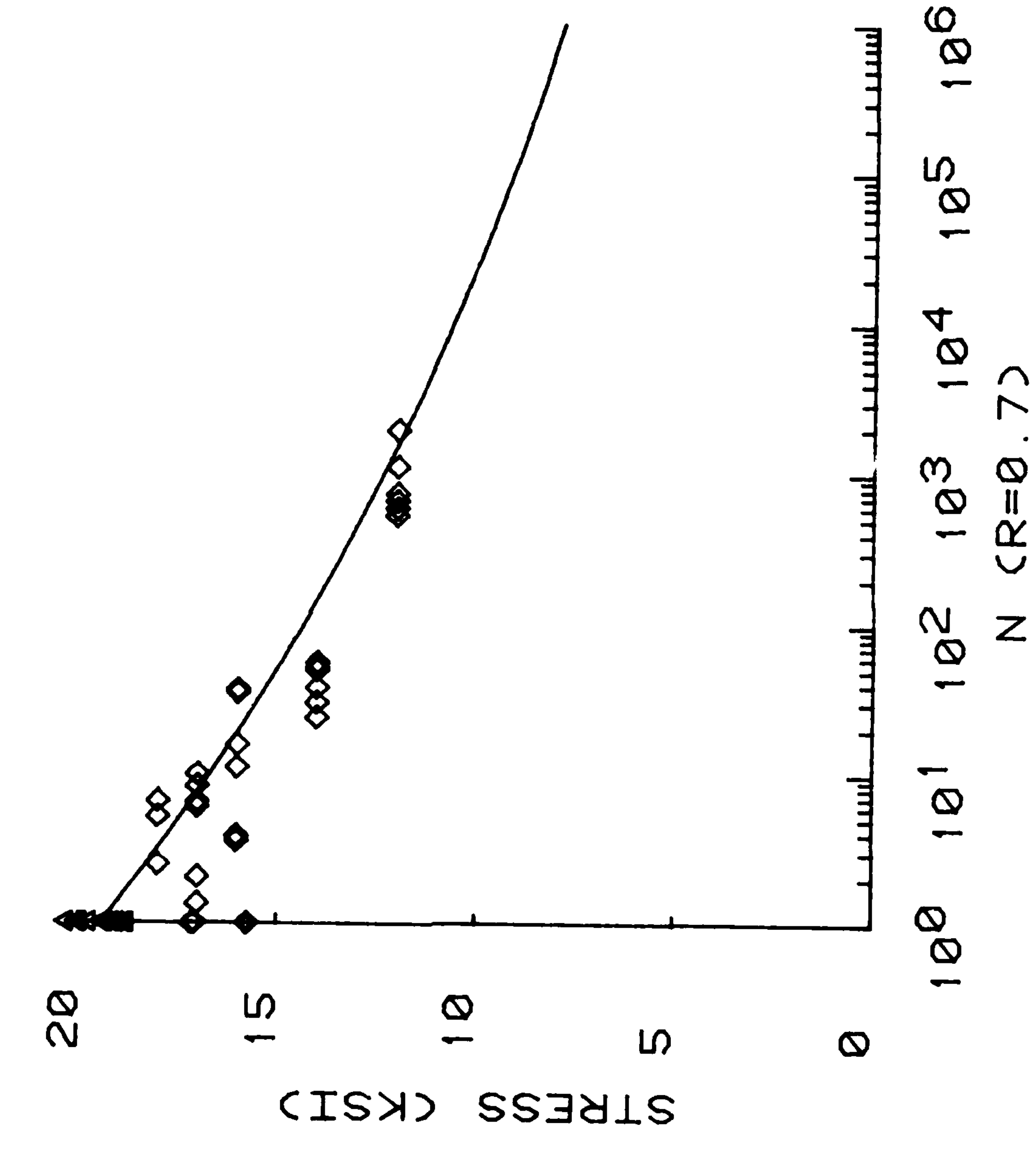

STRESS (KSI)
20
15
10
5
0
10^0
10^1
10^2
10^3
10^4
10^5
10^6
CYCLES


STRESS (KSI)
20
15
10
5
0
CYCLES, N
10^0
10^1
10^2
10^3
10^4
10^5
10^6

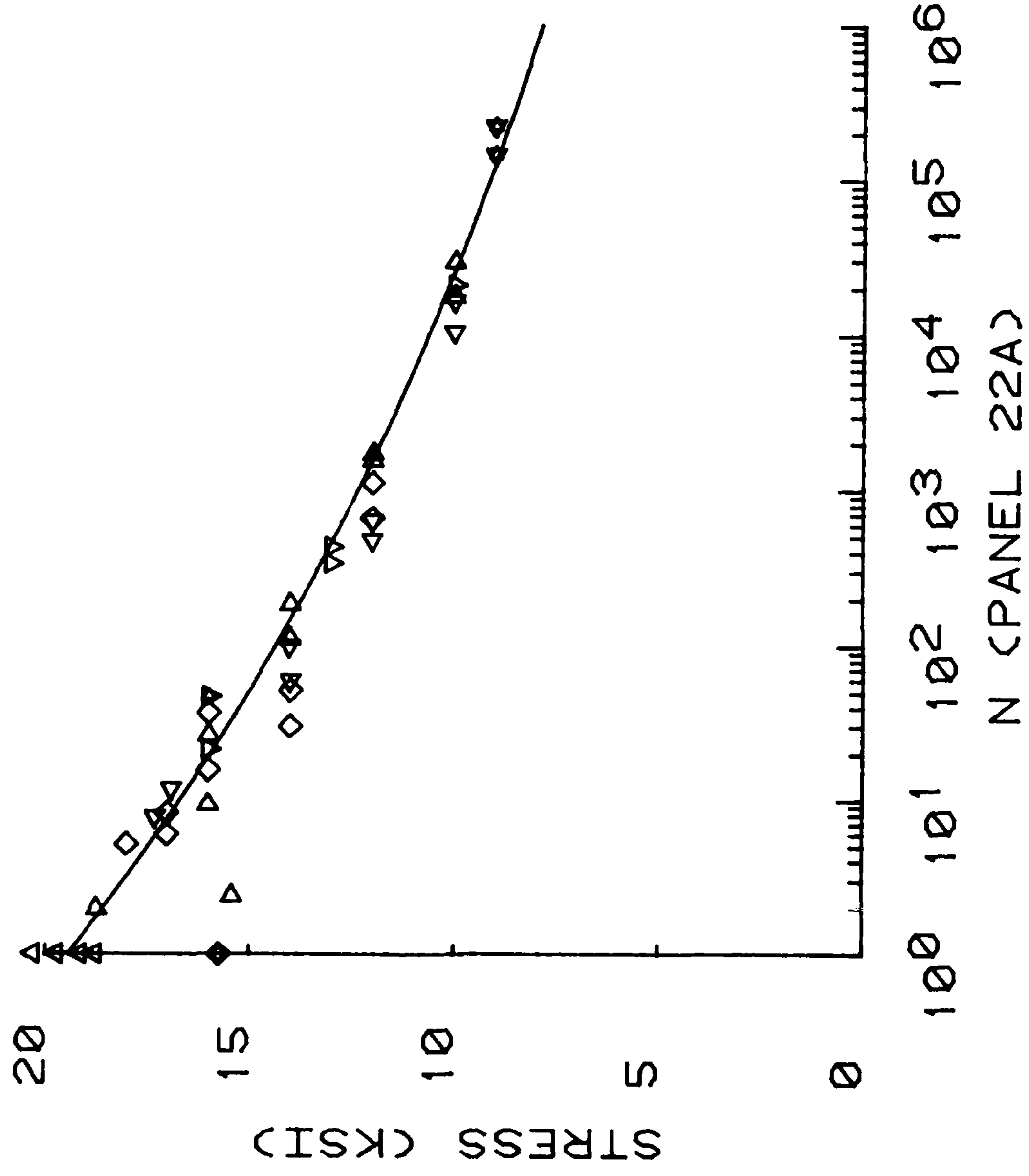

STRESS (KSI)
20
15
10
5
0
10^0
10^1
10^2
10^3
10^4
10^5
10^6
N (R=0.1)

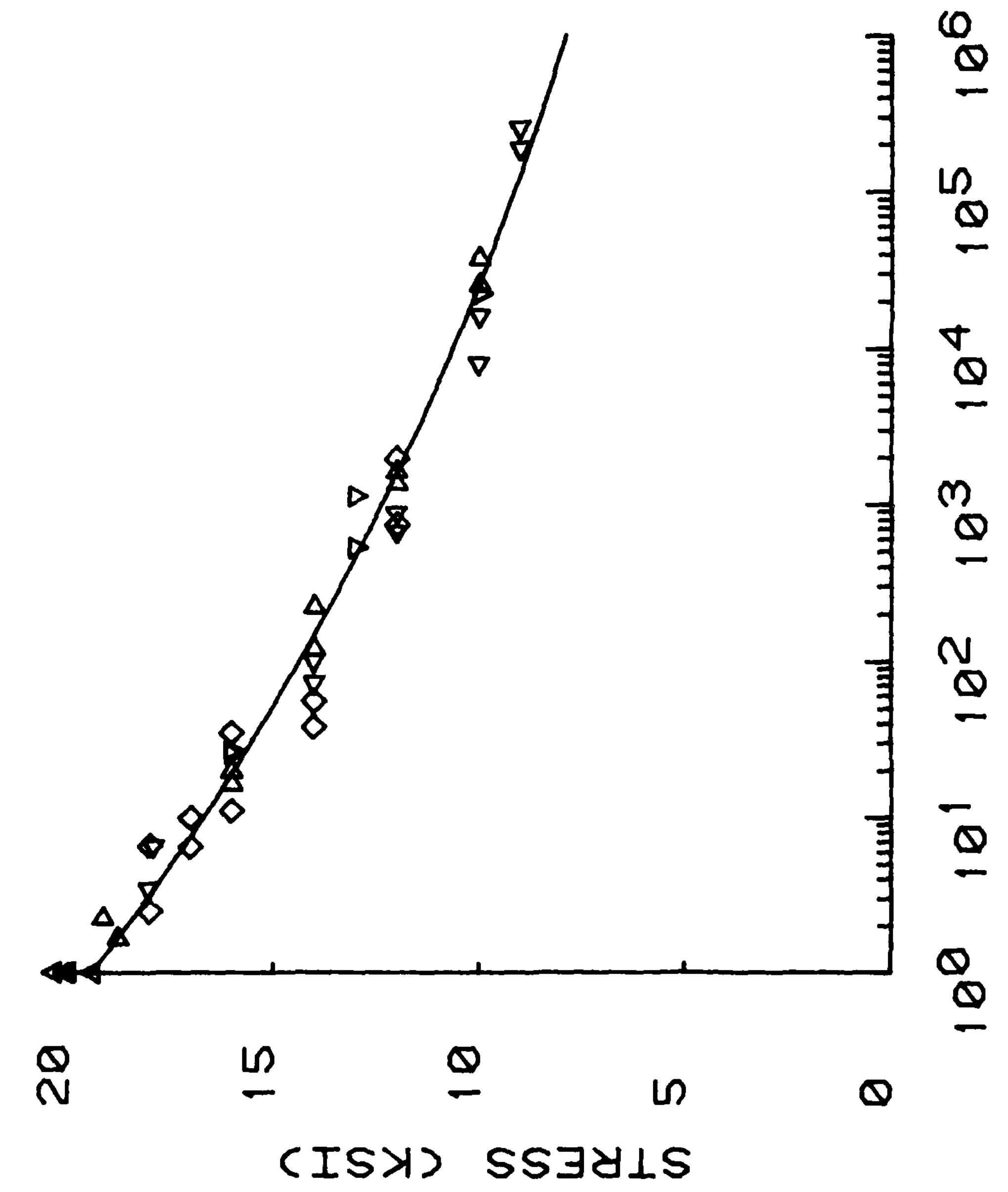

STRESS (KSI)
0
5
10
15
20
10^0
10^1
10^2
10^3
10^4
10^5
10^6
N (R=0.25)

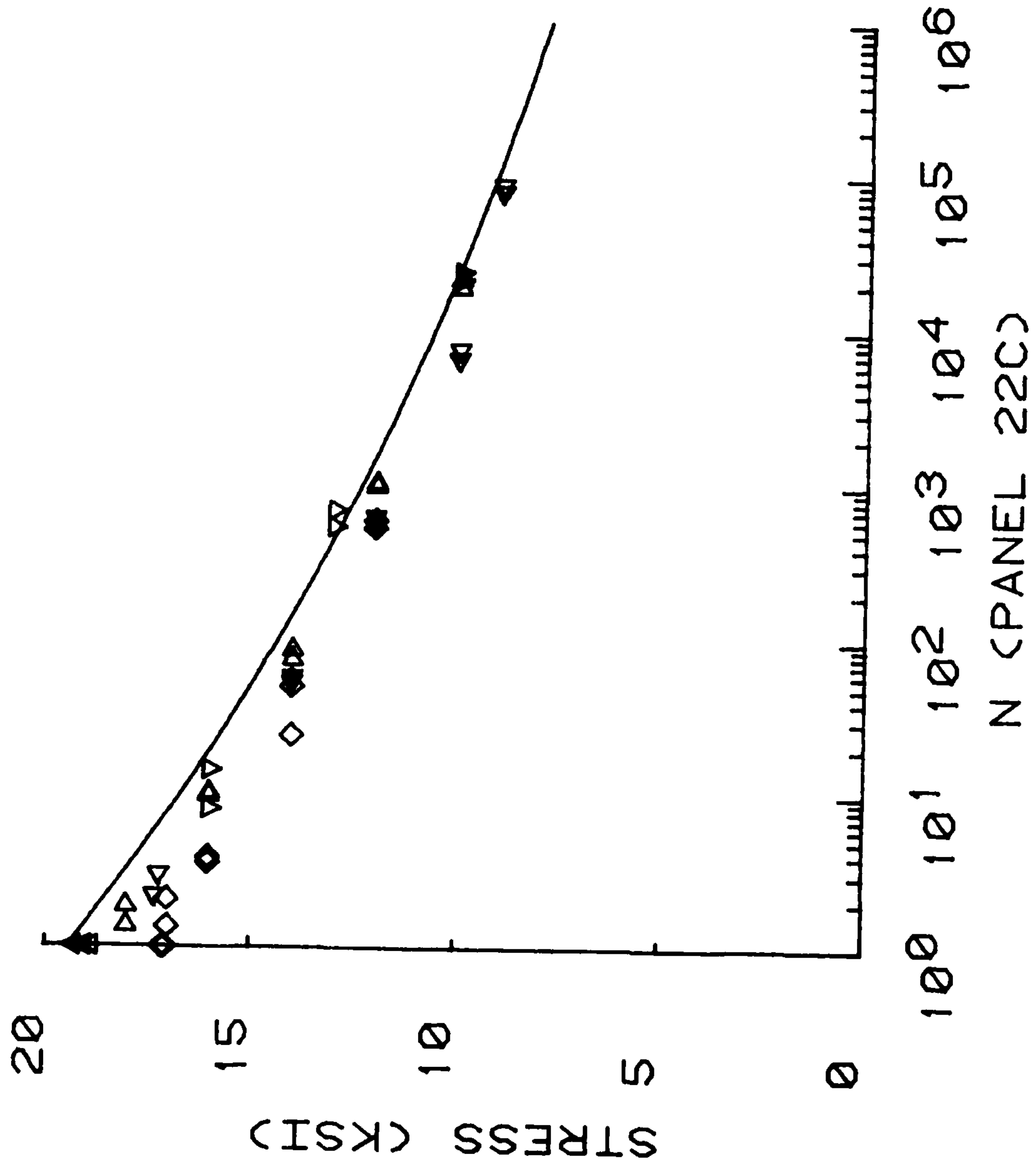

STRESS (KSI)
20
15
10
5
0
10^0
10^1
10^2
10^3
10^4
10^5
10^6
N (R=0.4)



STRESS (KSI)
20
15
10
5
0
10^0
10^1
10^2
10^3
10^4
10^5
10^6
N (R=0.7)


STRESS (KSI)
0
5
10
15
20
N (PANEL 22A)
10⁰
10¹
10²
10³
10⁴
10⁵
10⁶


STRESS (KSI)
20
15
10
5
0
N (PANEL 22B)
10^0
10^1
10^2
10^3
10^4
10^5
10^6


STRESS (KSI)
N (PANEL 22C)
0
5
10
15
20
10^0
10^1
10^2
10^3
10^4
10^5
10^6

CONCLUSIONS

PROCEDURE FOR TESTING FATIGUE MODELS DEVELOPED

MODEL W3A PROVIDES EXCELLENT FIT OF FATIGUE DATA

PROCEDURE FOR ASSESSING PANEL-TO-PANEL VARIABILITY DEVELOPED

USE OF MODEL W3A FOR LIFE PREDICTION DEMONSTRATED

HIGH-LOAD TRANSFER JOINTS FOR WING STRUCTURES

CONTRACT NUMBER. . . . . .N62269-80-C-0285 (COMPLETED)

N62269-82-C-0238 (CURRENT)

NADC PROJECT ENGINEER . . . . R. GARCIA

MCAIR PRINCIPAL INVESTIGATOR . . . . . S.P. GARBO

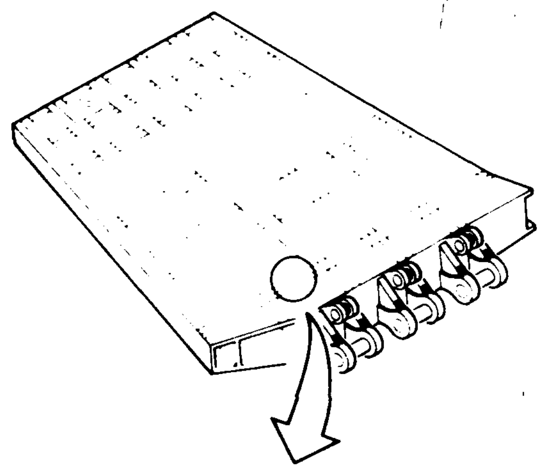
MCDONNELL AIRCRAFT COMPANY

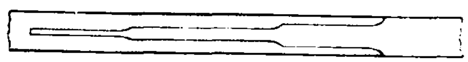
MCDONNELL DOUGLAS CORPORATION

GP23-0758-1

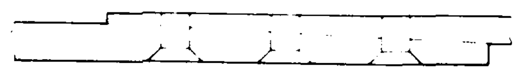
OBJECTIVE:

- DESIGN, SELECT AND EVALUATE METAL-TO-COMPOSITE BOLTED JOINTS AS AN ALTERNATIVE TO HIGH-LOAD TRANSFER ADHESIVE BONDED STEP-LAP JOINTS



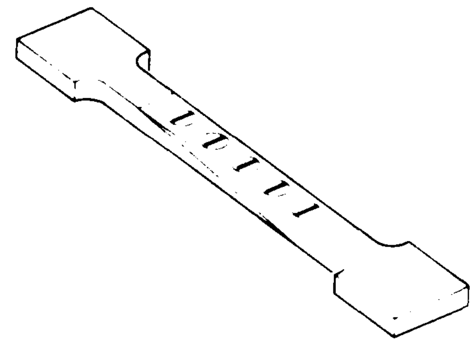


STEP-LAP
BONDED JOINT



MULTIFASTENED SPLICE

LAMINATE TAILORING



TASK 1
DOCUMENT BASELINE CONFIGURATION

TASK 2
EVALUATE POSSIBLE DESIGN ALTERNATIVES

TASK 3
VERIFY DESIGN BY SPECIMEN TESTING

TASK 4
PAYOFF EVALUATIONS AND RECOMMENDATIONS FOR FULL-SCALE STRUCTURAL TEST PROGRAM

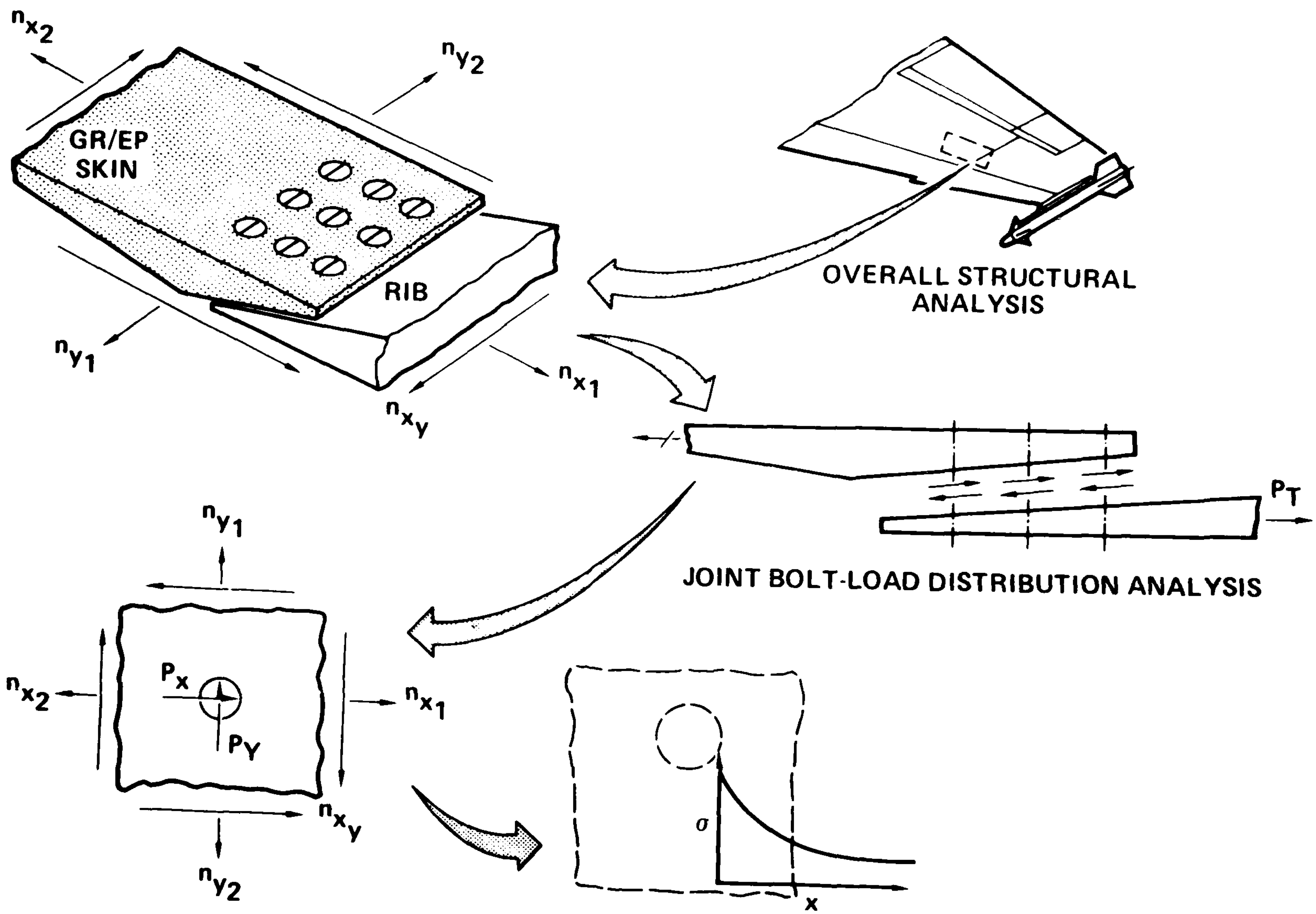
GP13-0770-2

CONCLUSIONS

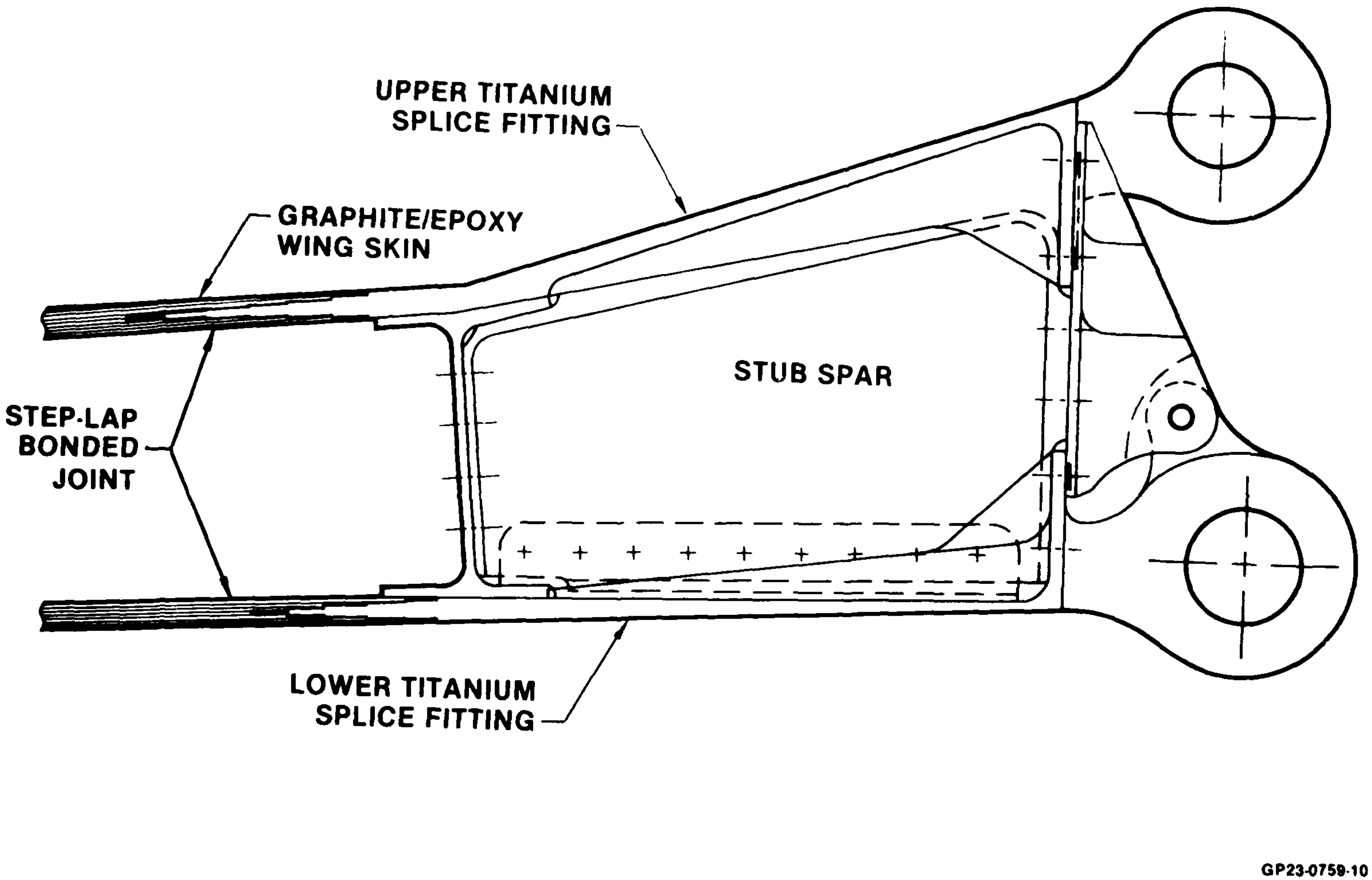
- **METHODOLOGY VERIFIED**
- **SOFTENED LAMINATES REDUCE WEIGHT**
- **BOLTED JOINTS COST COMPETITIVE**
  - LESS FABRICATION AND ASSEMBLY COSTS
  - TITANIUM COSTS SIGNIFICANT
- **FULL SCALE ASPECTS REQUIRE VERIFICATION**

GP23-0759-11

BOLTED JOINT DESIGN SEQUENCE



F/A-18 WING ROOT ATTACHMENT CONFIGURATION



BJSFM PROGRAM DESCRIPTION

INPUT DATA

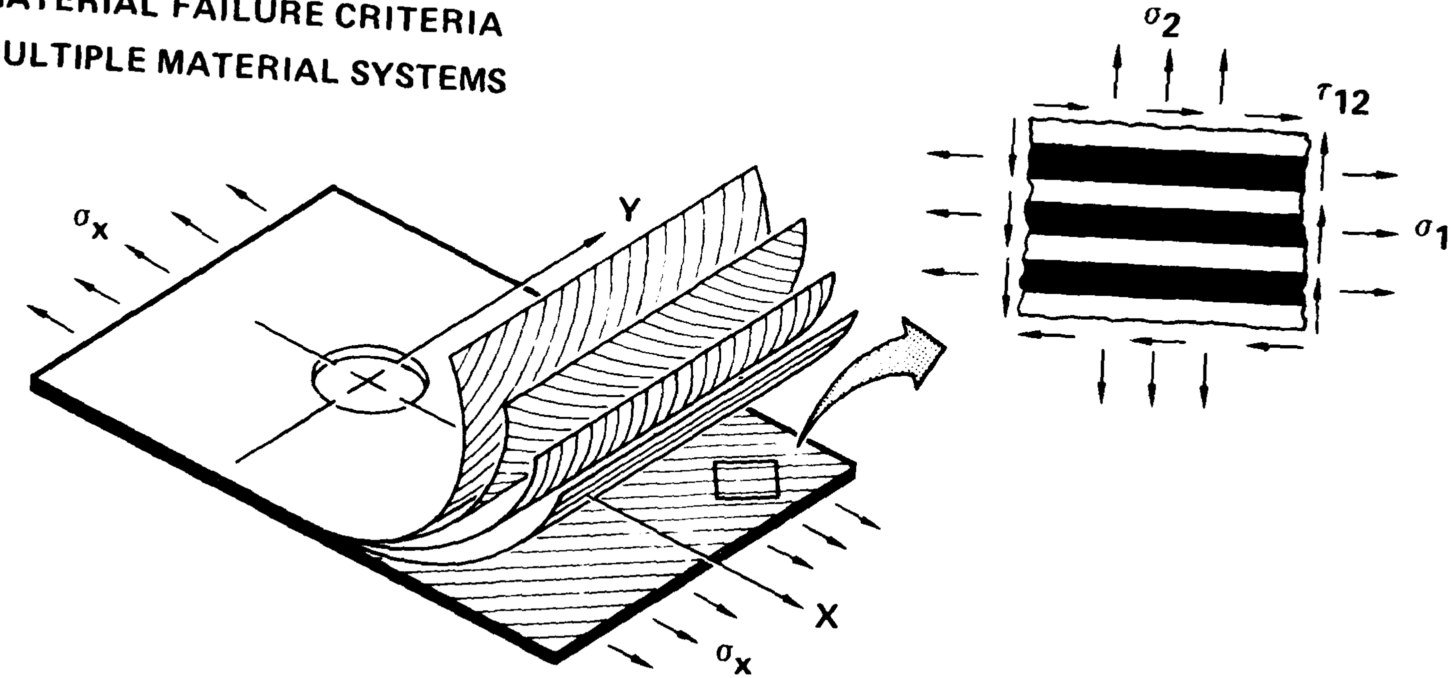
- UNIDIRECTIONAL MECHANICAL PROPERTIES
- GEOMETRIES
- LOADINGS

OUTPUT DATA

- STRESS/STRAIN DISTRIBUTIONS
- FAILURE ANALYSIS

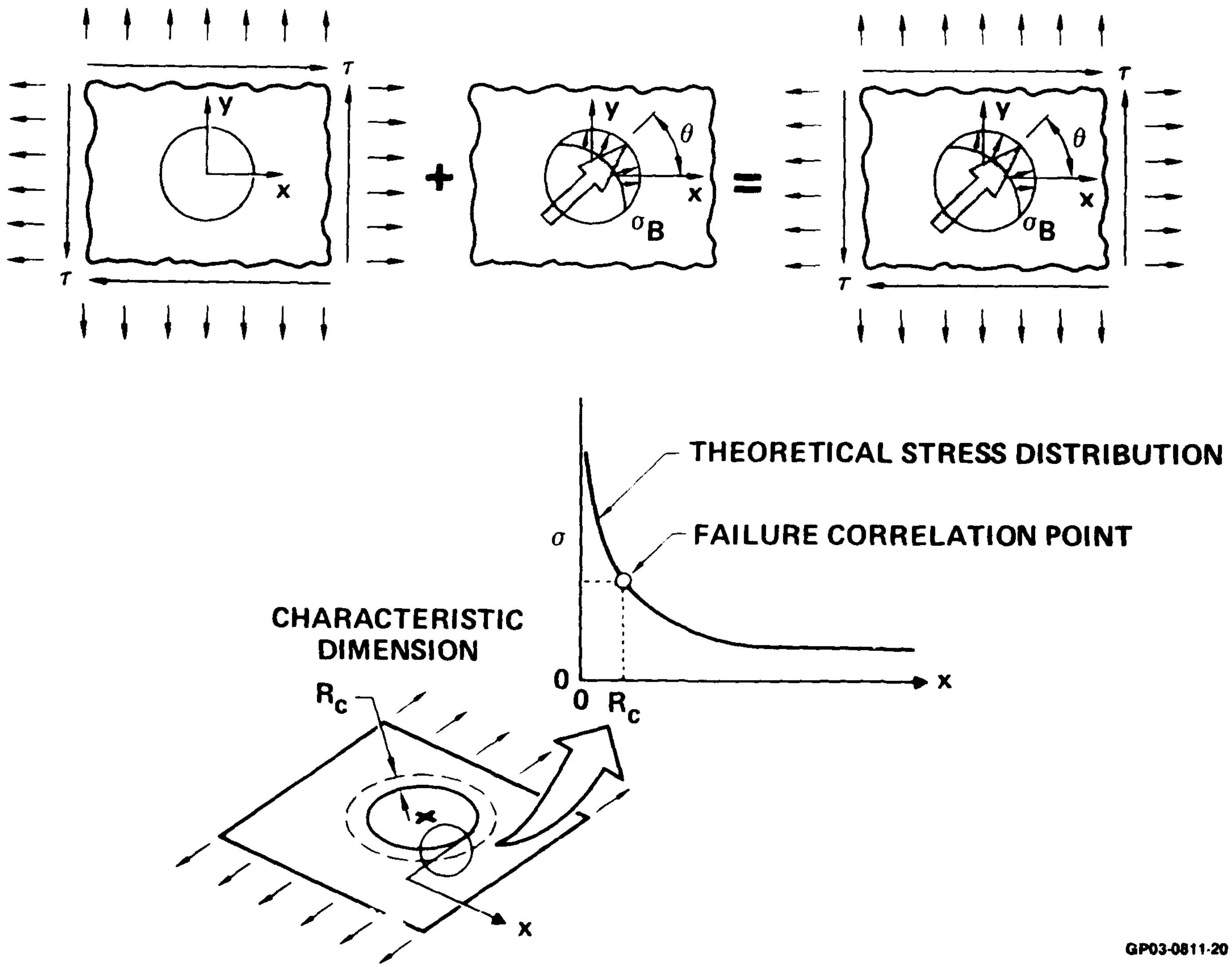
OPTIONS

- OUTPUT DATA
- MATERIAL FAILURE CRITERIA
- MULTIPLE MATERIAL SYSTEMS

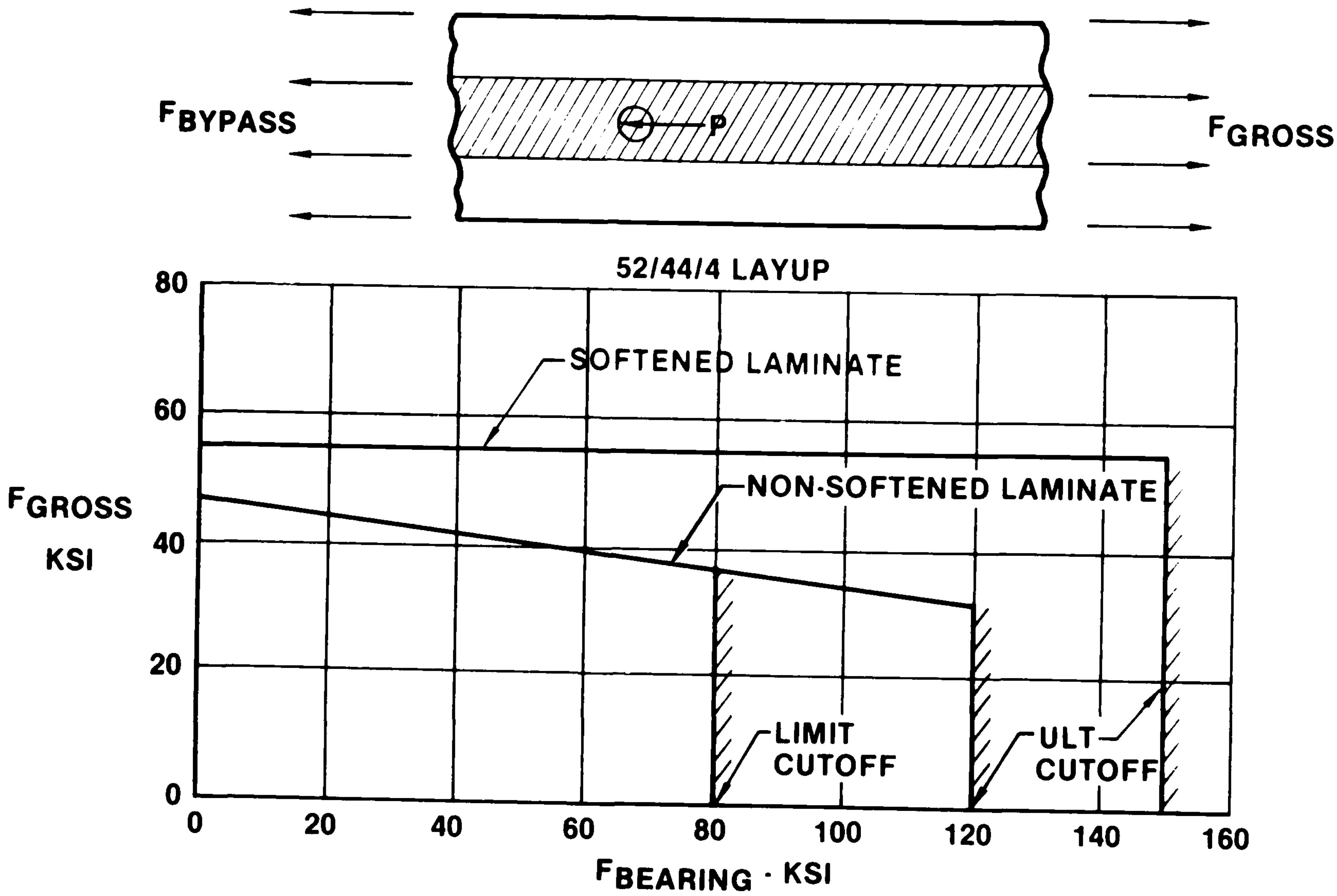


GP79-0938-3

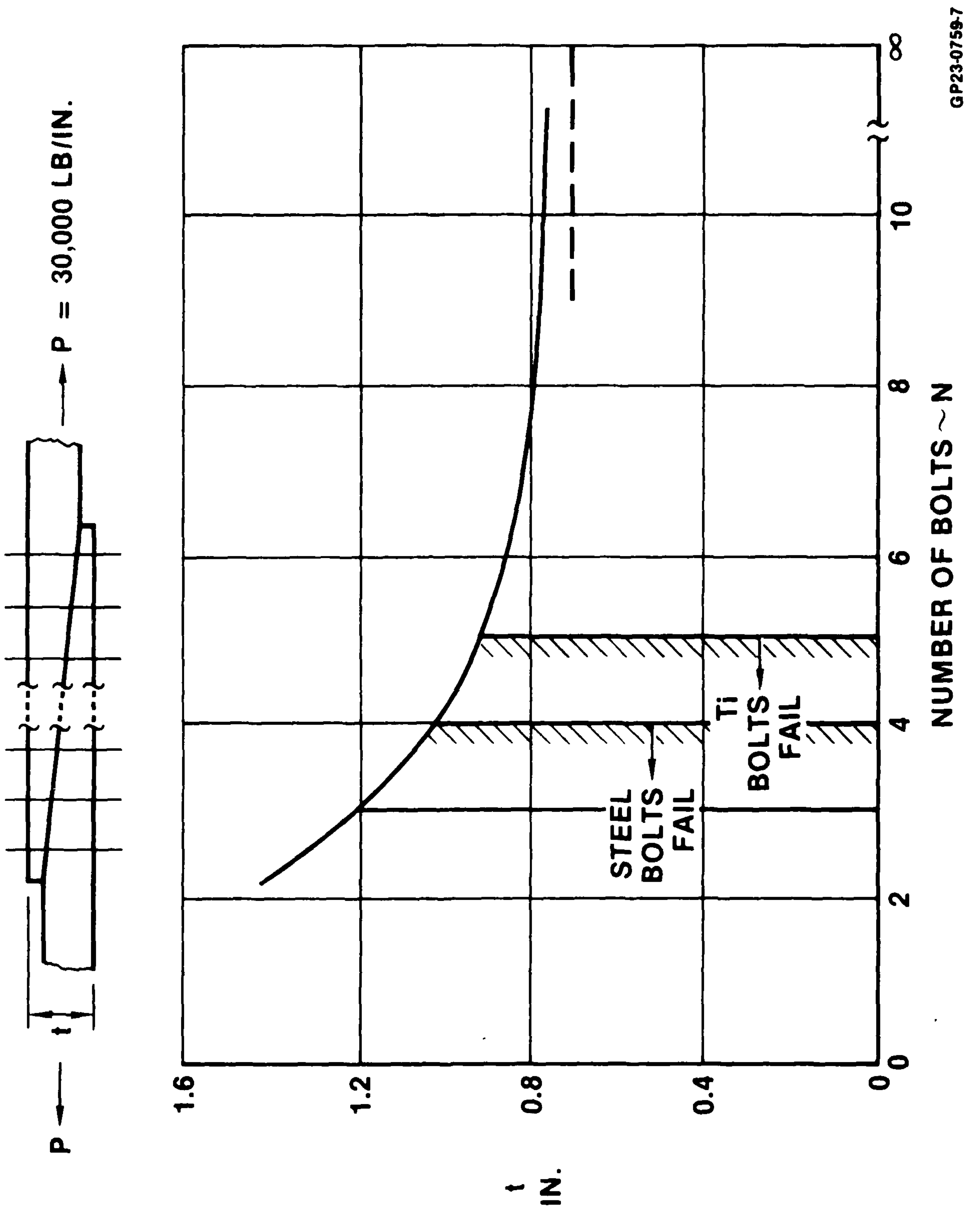
FAILURE PREDICTION OF BOLTED COMPOSITE JOINTS

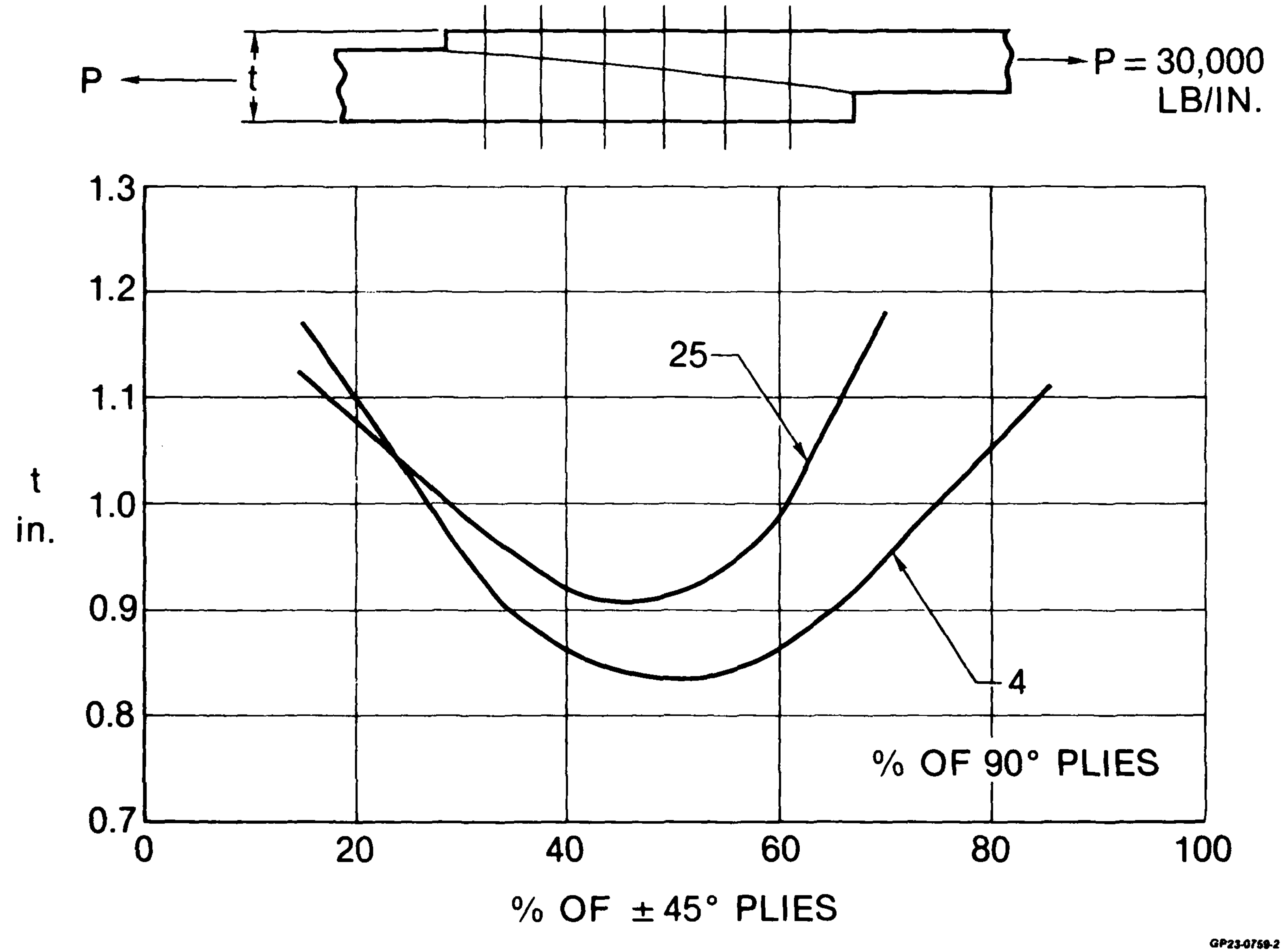


GP03-0811-20

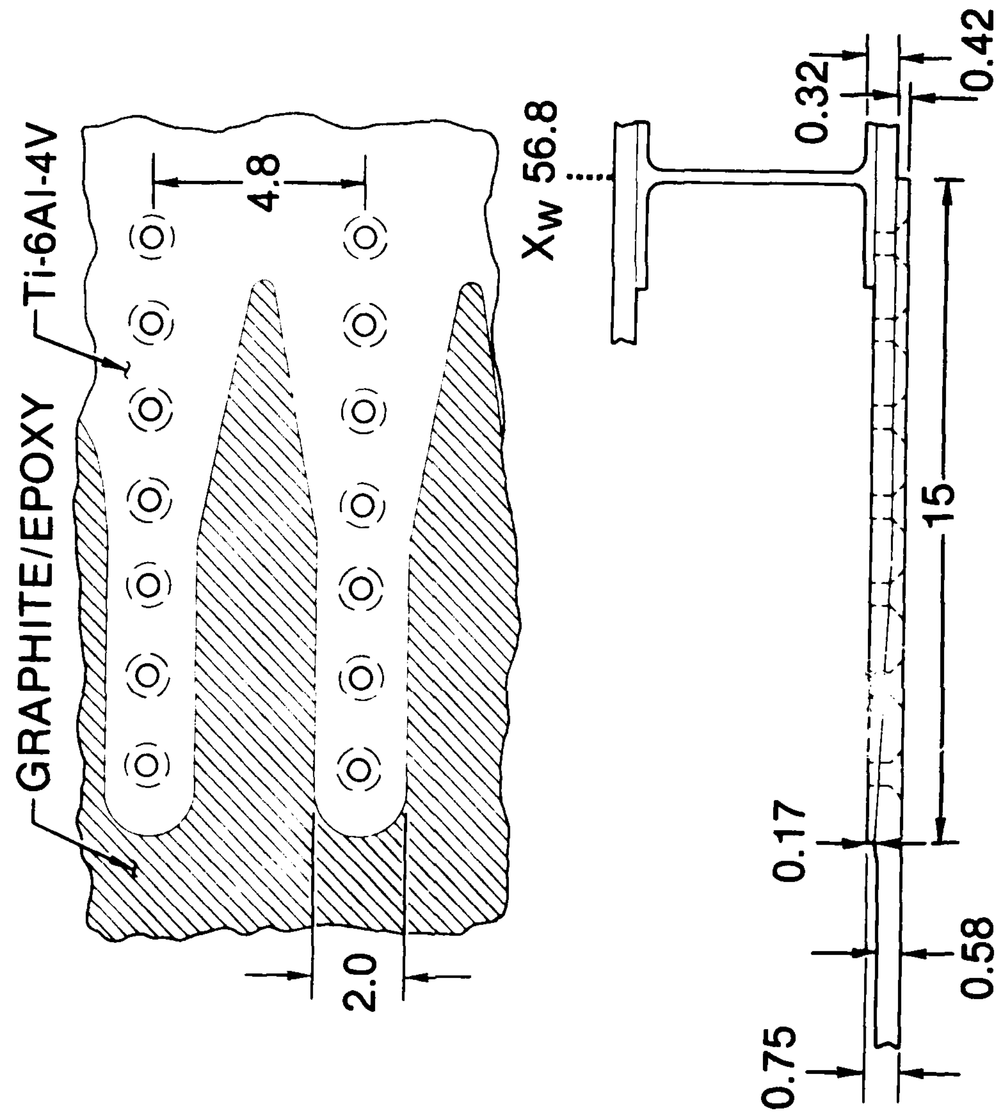


FBYPASS
P
FGROSS
52/44/4 LAYUP
SOFTENED LAMINATE
NON-SOFTENED LAMINATE
LIMIT CUTOFF
ULT CUTOFF
FGROSS KSI
FBEARING - KSI
0
20
40
60
80
100
120
140
160
GP23-0759-6


P
P = 30,000 LB/IN.
t
t
IN.
1.6
1.2
0.8
0.4
0
0
2
4
6
8
10
∞
STEEL
BOLTS
FAIL
Ti
BOLTS
FAIL
NUMBER OF BOLTS ~ N
GP23-0759-7

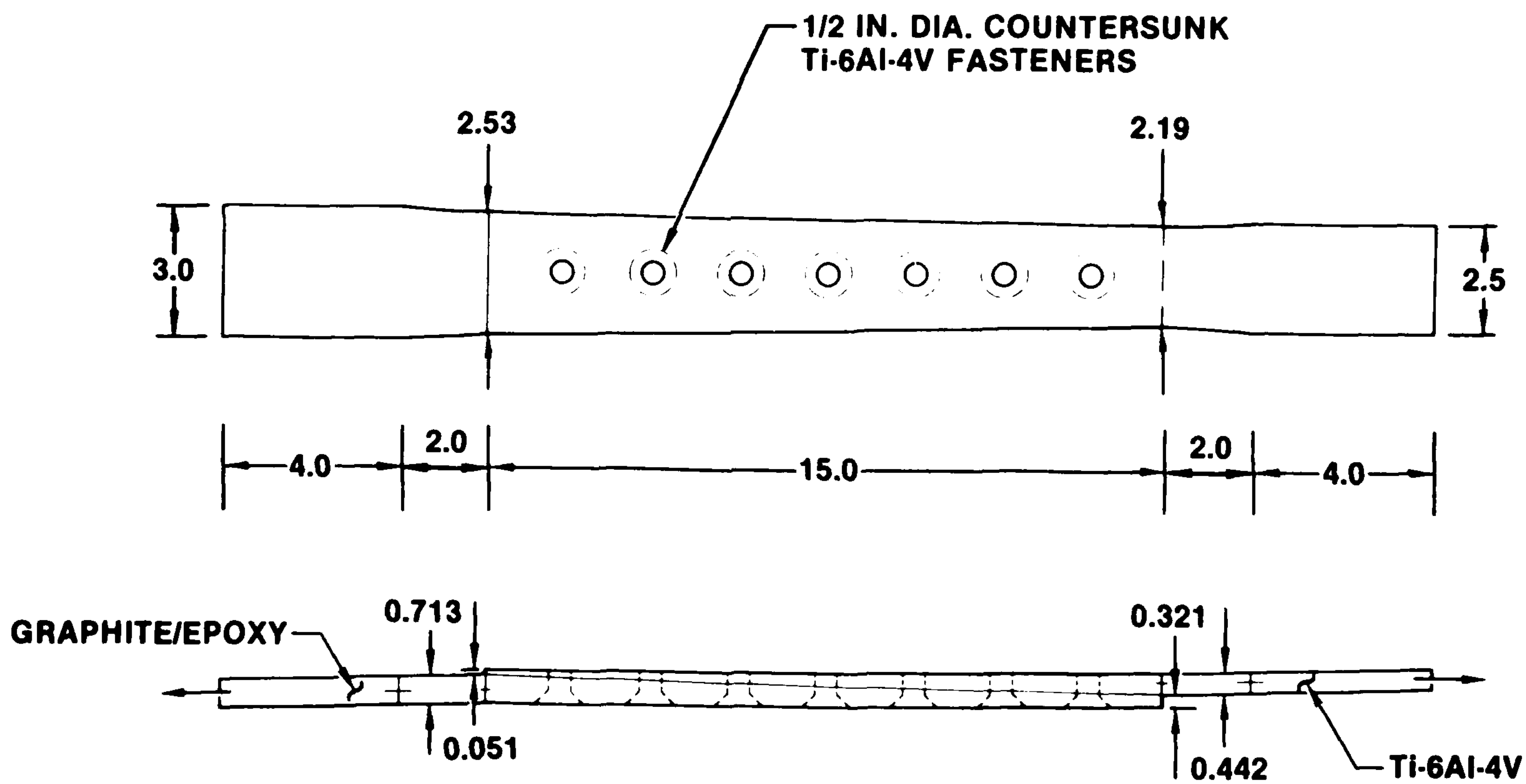


P
t
P = 30,000
LB/IN.
1.3
1.2
1.1
1.0
0.9
0.8
0.7
t
in.
0
20
40
60
80
100
25
4
% OF 90° PLIES
% OF ± 45° PLIES
GP23-0759-2



GRAPHITE/EPOXY
Ti-6Al-4V
4.8
2.0
X_W 56.8
0.32
0.42
15
0.17
0.58
0.75
GP23-0759-8

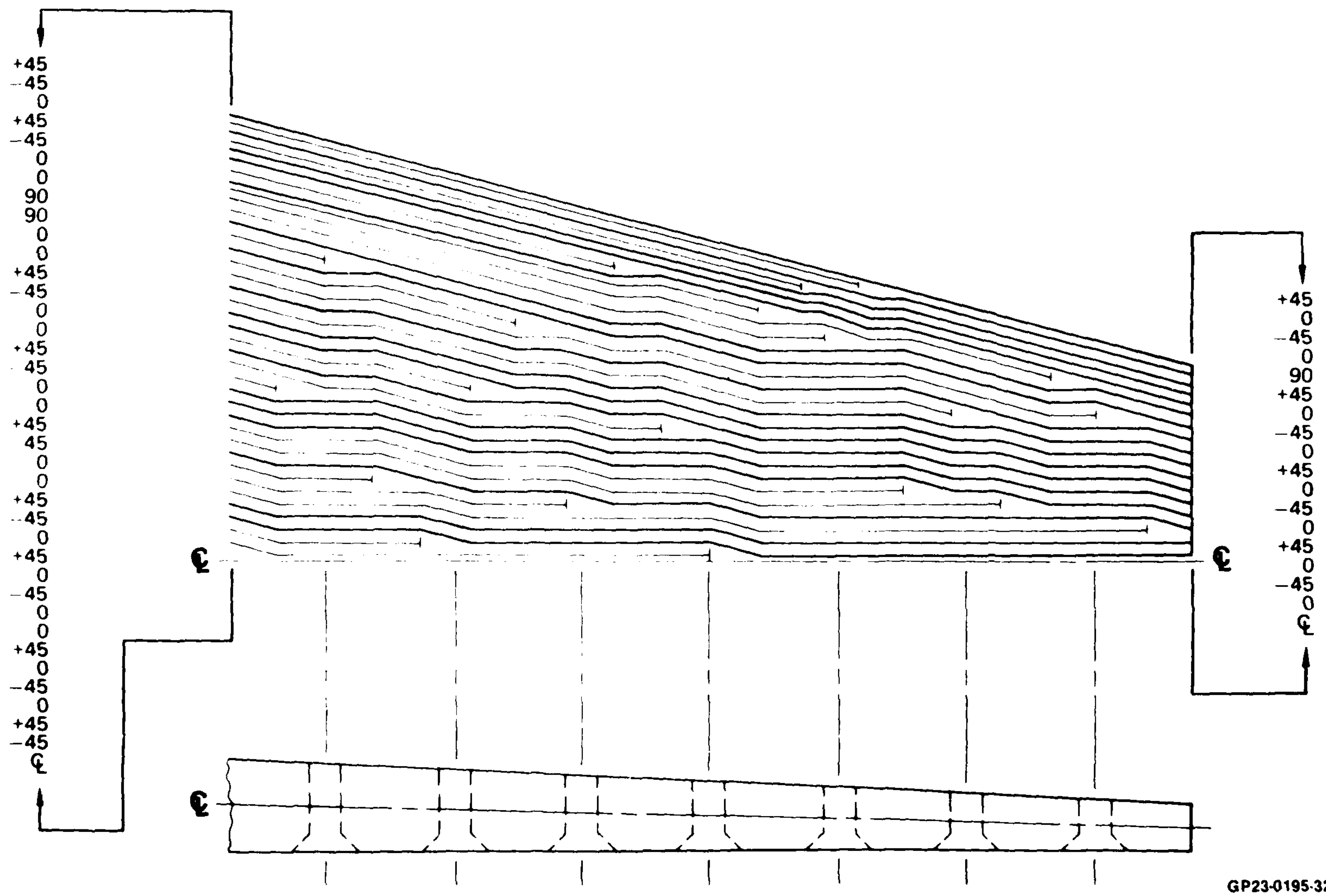
SCARFED MONOLITHIC-LAMINATE BOLTED JOINT TEST SPECIMEN



Note: All dimensions are in inches

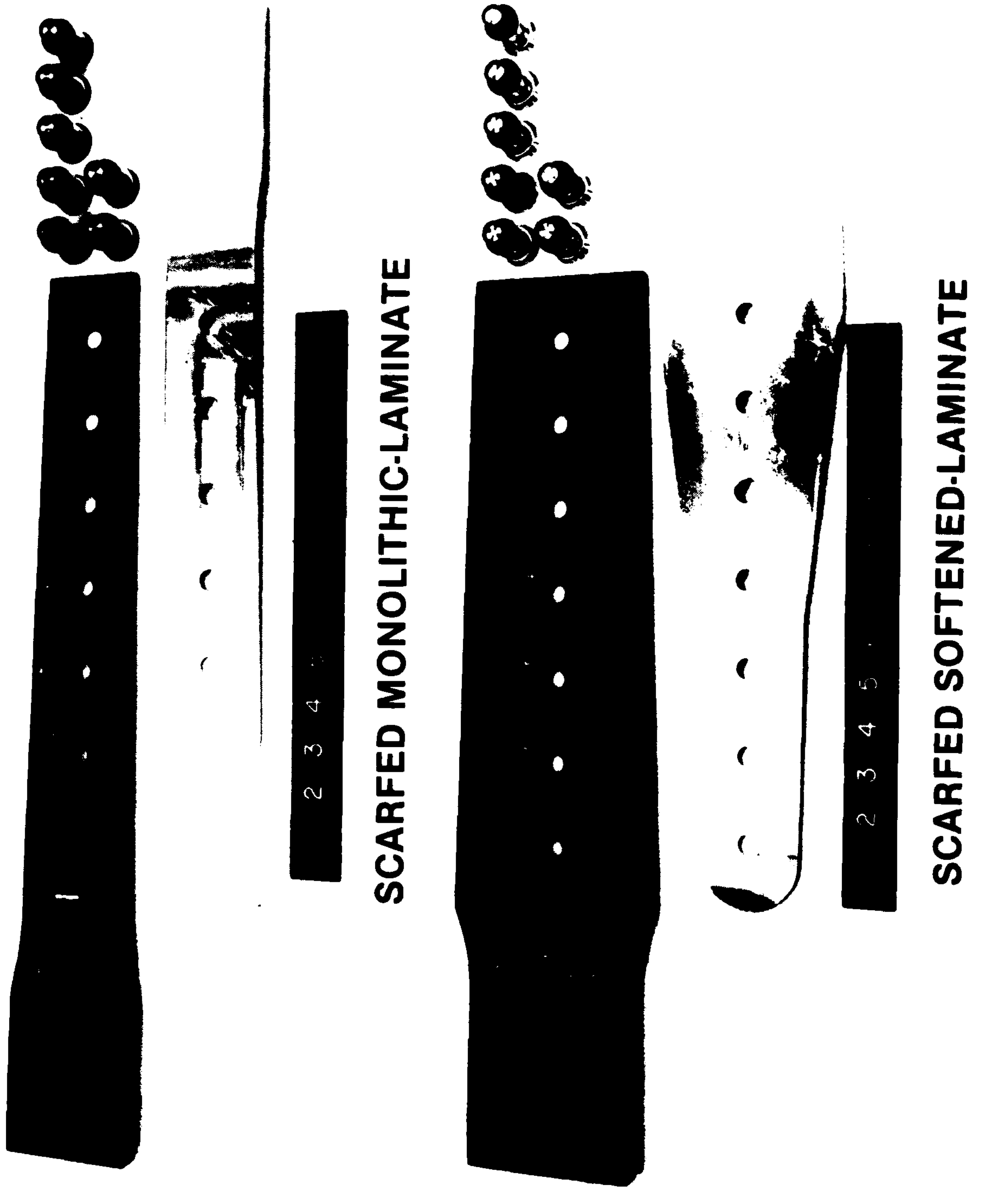
GP23-0195-18

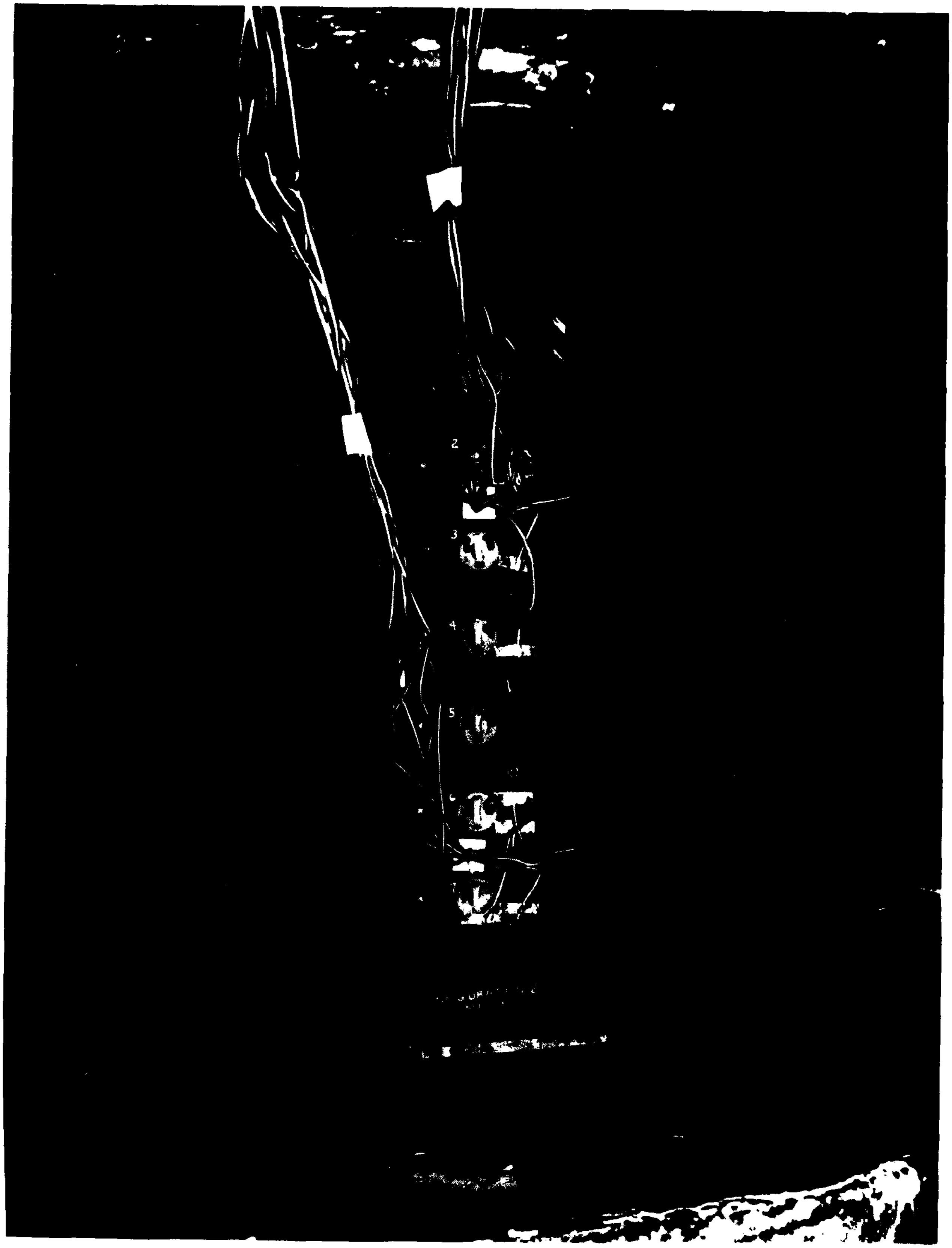
SCARFED MONOLITHIC-LAMINATE PLY LAYUP



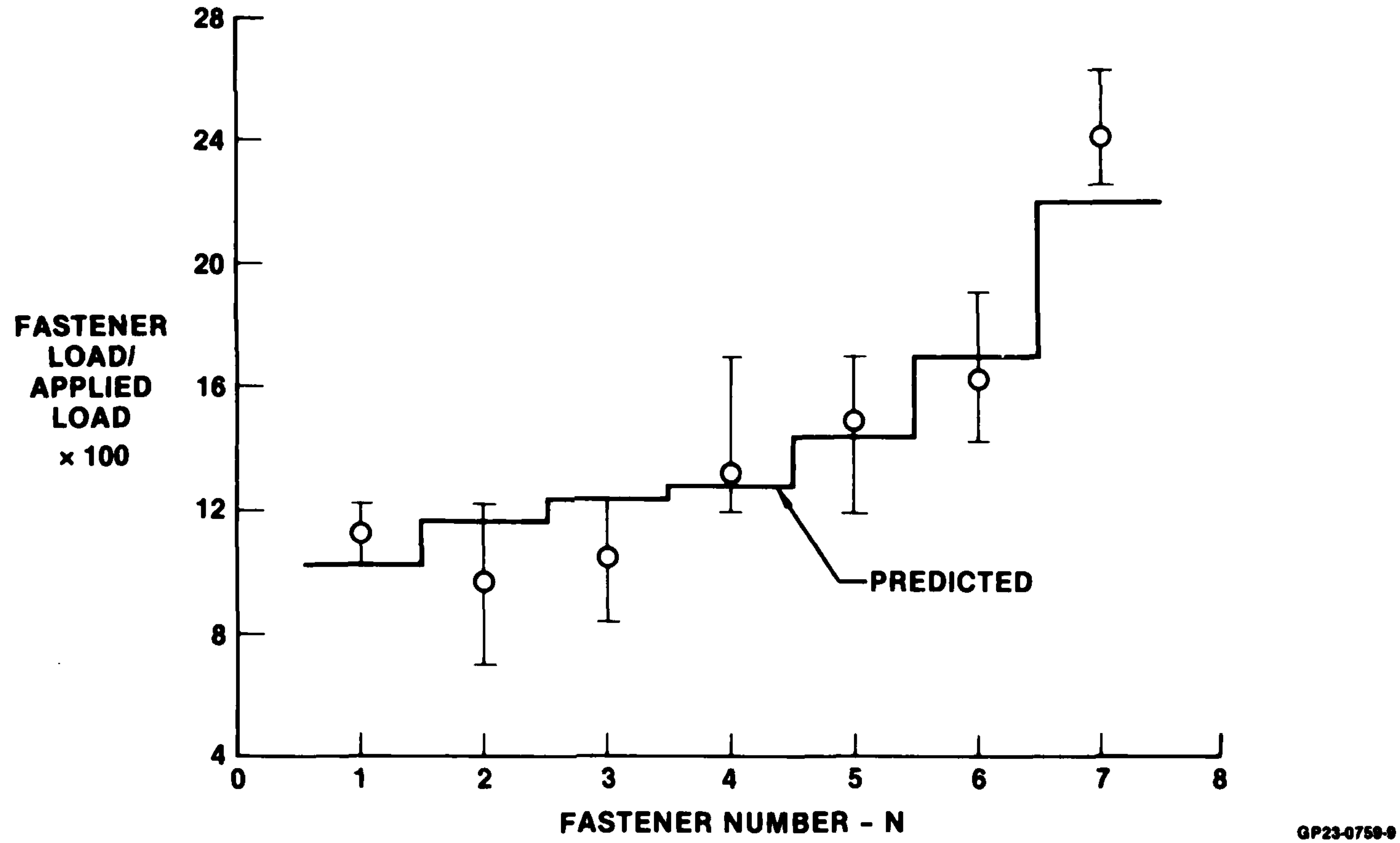
GP23-0195-32

GP23-0759-4

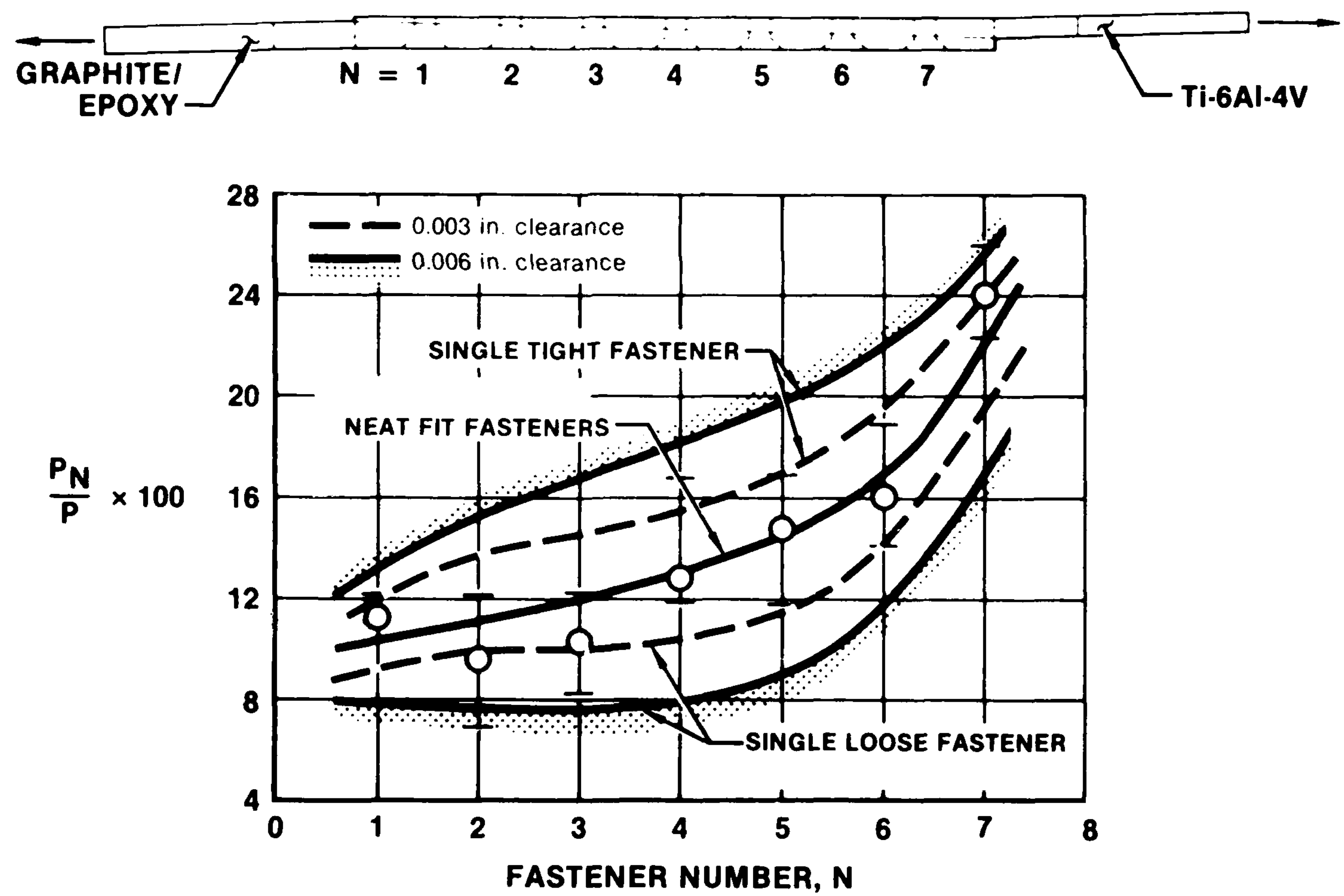



2
3
4
5

SCARFED MONOLITHIC-LAMINATE THEORY/TEST CORRELATION



EFFECT OF HOLE CLEARANCES ON BOLT LOADS



GP23-0195-8

COST AND WEIGHT COMPARISONS

PART	ITEM	COST STUDY DESIGNS F/A-18	COST STUDY DESIGNS NO. 1	COST STUDY DESIGNS NO. 2	WEIGHT STUDY DESIGNS F/A-18	WEIGHT STUDY DESIGNS NO. 1	WEIGHT STUDY DESIGNS NO. 2
GRAPHITE/ EPOXY	MATERIAL	0.208	0.212	0.205			
	FAB	0.102	0.086	0.100	0.41	0.46	0.38
	ASSEMBLY	0.020	0.010	0.008			
	SUBTOTAL	0.330	0.308	0.313			
Ti-6Al-4Vn	MATERIAL	0.458	0.491	0.491			
	FAB	0.204	0.182	0.193	0.59	0.76	0.76
	ASSEMBLY	0.008	0.011	0.007			
	SUBTOTAL	0.670	0.679	0.691			
FASTENERS		—	—	—	0	0.10	0.09
	TOTALS	1.000	0.992	1.004	1.00	1.32	1.23

Notes:
No. 1 Monolithic - laminate
No. 2 Softened - laminate

GP23-0759-3

AD P001258

DESIGN METHODOLOGY FOR BONDED-BOLTED COMPOSITE JOINTS

by

L. J. HART-SMITH

AIR FORCE FLIGHT DYNAMICS LABORATORY CONTRACT F33615-79-C-3212
AIR FORCE PROJECT ENGINEERS: V. B. VENKAYYA AND P. J. CONRAD

DOUGLAS AIRCRAFT COMPANY
MCDONNELL DOUGLAS CORPORATION

82-PAB 768

OBJECTIVES

DEVELOPMENT OF NEW ANALYSIS/DESIGN COMPUTER PROGRAMS FOR COMPOSITE JOINTS

A4EI FOR BONDED JOINTS

- ELASTIC-PLASTIC ADHESIVE BEHAVIOR
- STEPPED OR UNIFORM ADHERENDS
- VARIABLE ADHESIVE THICKNESS
- VARIABLE ADHESIVE PROPERTIES (FLAWS AND POROSITY)
- RESIDUAL THERMAL STRESSES

A4EJ FOR BOLTED JOINTS

- LOAD SHARING IN MULTIROW JOINTS
- BEARING/BYPASS INTERACTION FOR STRENGTH PREDICTION
- NONLINEAR FASTENER DEFLECTION CHARACTERISTICS
- NONLINEAR ADHEREND BEHAVIOR
- STEPPED OR UNIFORM MEMBERS

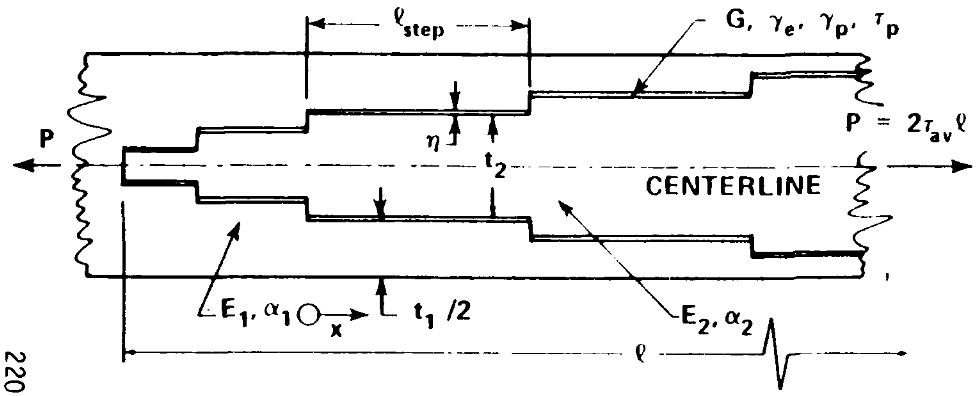
A4EK FOR BONDED-BOLTED JOINTS –

- ALL OF ABOVE FEATURES, EXCEPT YIELDING ADHERENDS

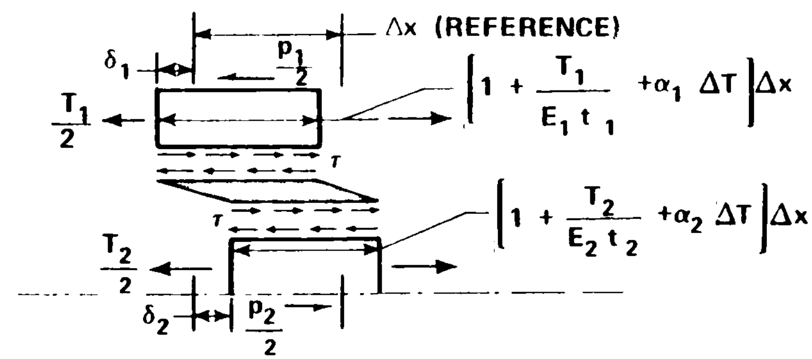
ILLUSTRATION OF ANALYSIS CAPABILITIES BY SAMPLE SOLUTIONS

82 PAB 769

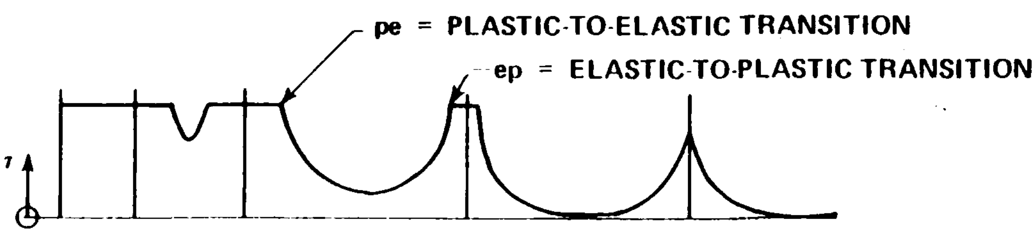
NOTATION AND GEOMETRY FOR ADHESIVE-BONDED STEPPED-LAP JOINT ANALYSIS



JOINT GEOMETRY



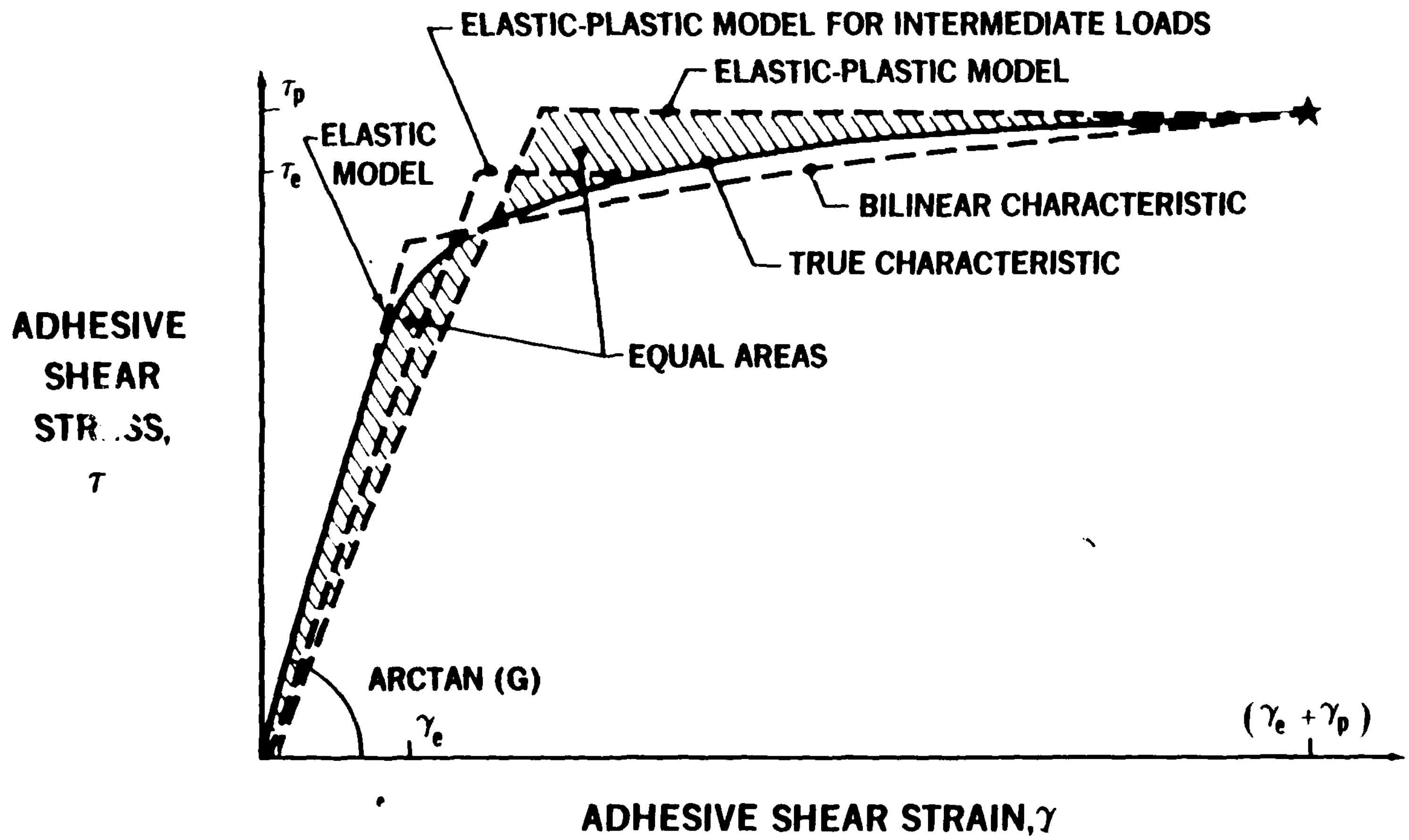
DISPLACEMENTS AND ELEMENT LOADS



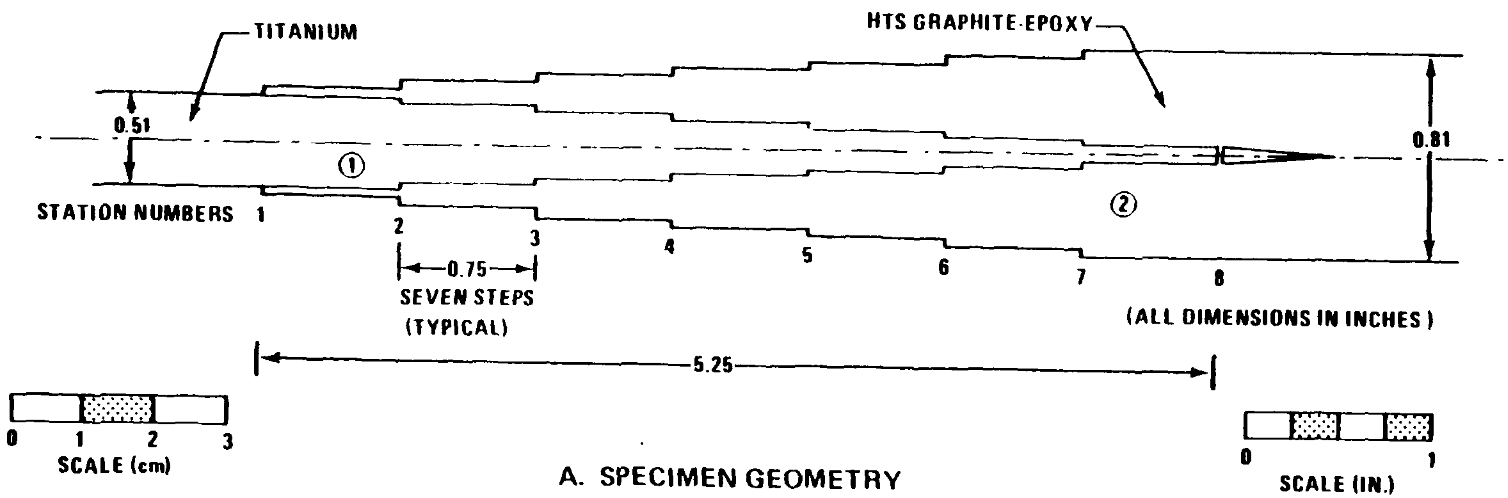
ADHESIVE SHEAR STRESS DISTRIBUTION

9 DP 8322

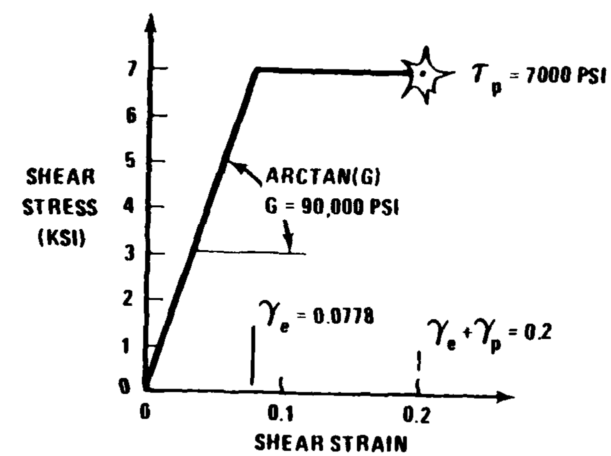
REPRESENTATIONS OF ADHESIVE NONLINEAR SHEAR BEHAVIOR



STEPPED-LAP ADHESIVE-BONDED JOINT



A. SPECIMEN GEOMETRY



ADHESIVE THICKNESS η = 0.005 IN.

B. ADHESIVE PROPERTIES

ADHEREND 1

MATERIAL: TITANIUM

$E = 16.0 \times 10^6$ PSI

F_{tu} = 130 KSI

α = 0.000006/°F

ADHERENDS 2

MATERIAL: HTS GRAPHITE

($\simeq$45% 0° PLIES)

$E = 10.0 \times 10^6$ PSI

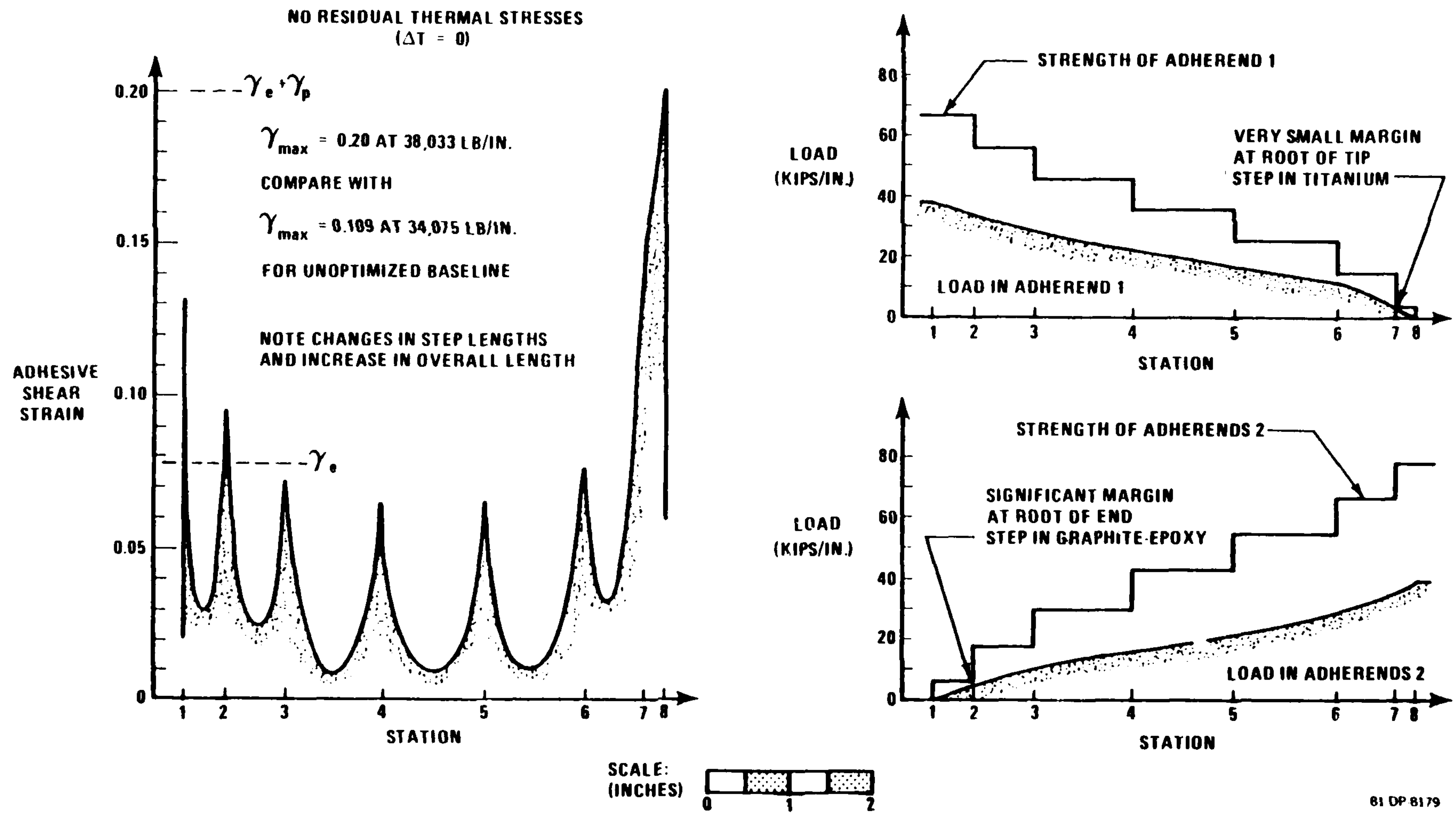
F_{tu} = 100 KSI

α = 0

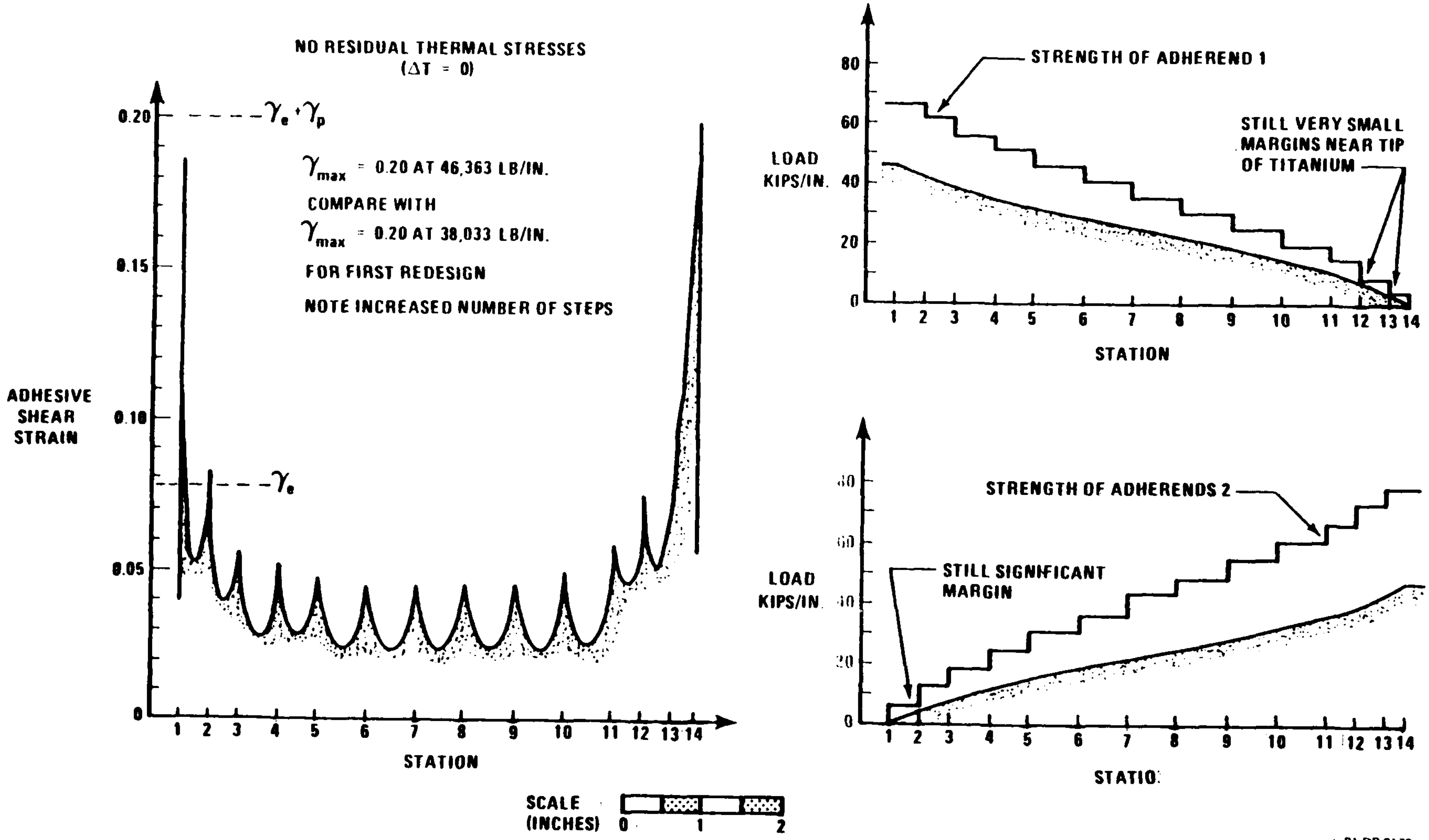
C. ADHEREND PROPERTIES

81 DP 8190

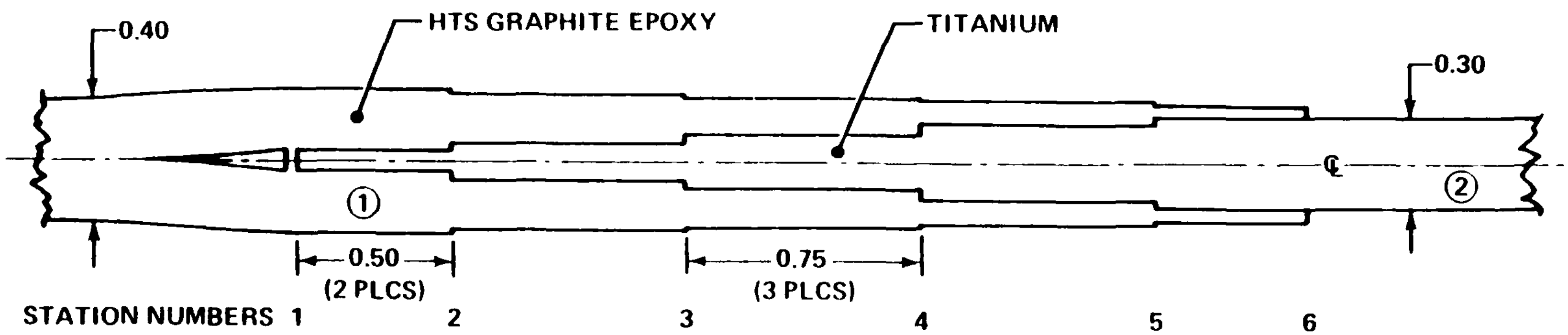
IMPROVEMENTS DUE TO FIRST REDESIGN OF STEPPED-LAP BONDED JOINT



IMPROVEMENTS DUE TO SECOND REDESIGN OF STEPPED-LAP BONDED JOINT

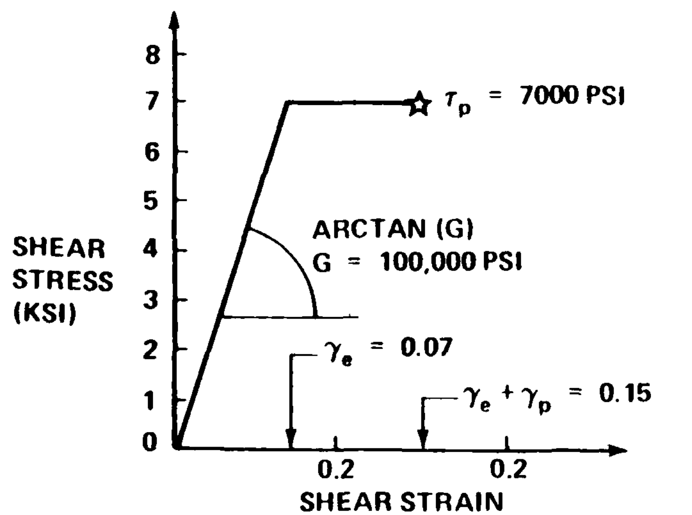


TYPICAL STEPPED-LAP ADHESIVE-BONDED JOINT



A. SPECIMEN GEOMETRY

(ALL DIMENSIONS IN INCHES)



NOMINAL ADHESIVE THICKNESS η = 0.005 IN.

B. ADHESIVE PROPERTIES

ADHERENDS "1"
MATERIAL: TITANIUM
$E = 12 \times 10^6$ PSI
F_{tu} = 120 KSI
$\alpha = 0$

ADHERENDS "2"
MATERIAL: TITANIUM
$E = 16 \times 10^6$ PSI
F_{tu} = 130 KSI
$\alpha = 0.000006/°F$

C. ADHEREND PROPERTIES

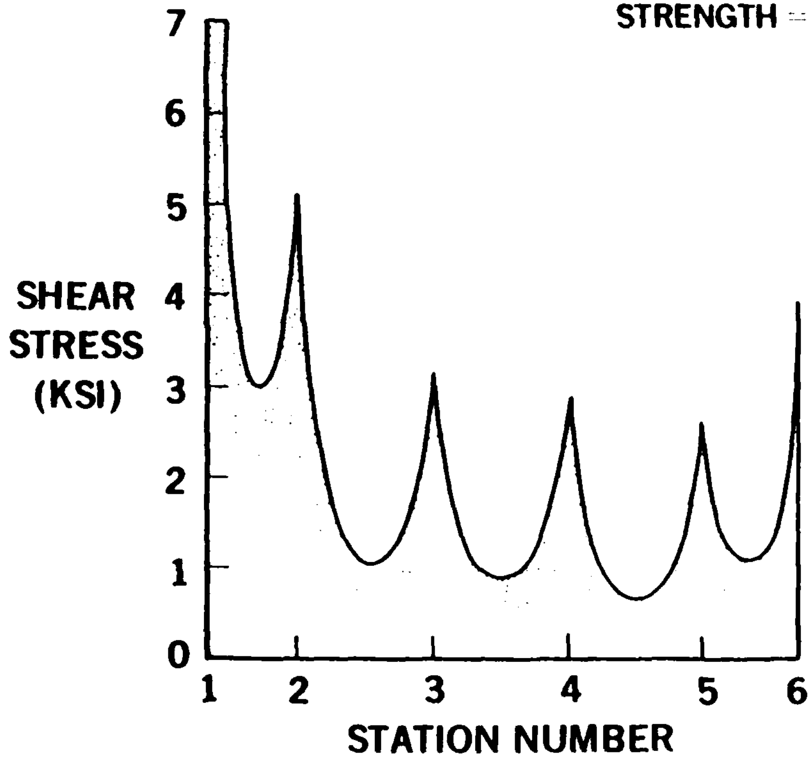
81-DP-8596

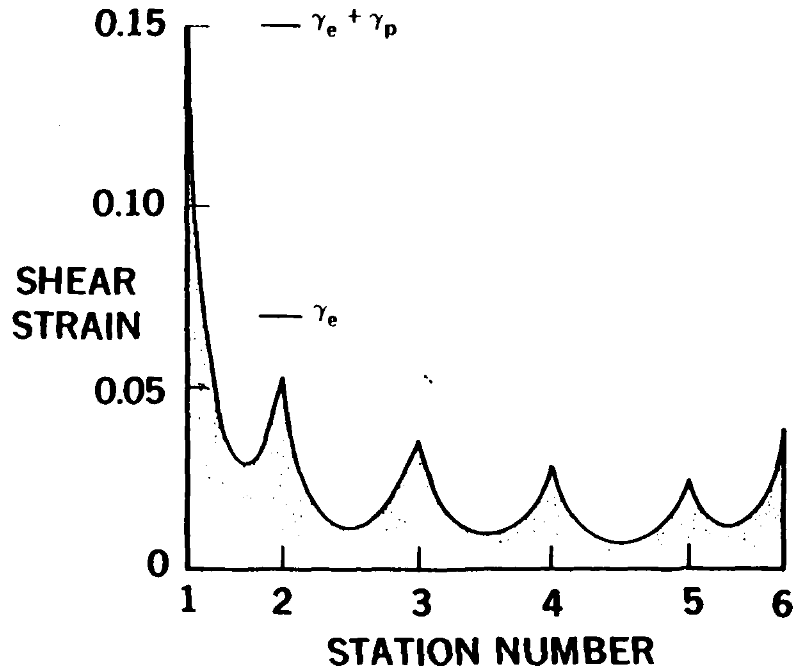
ADHESIVE SHEAR STRESSES AND STRAINS IN UNFLAWED JOINT

TENSILE LOADING

($\Delta T = -250°F$)

STRENGTH = 13,749 LB/IN.



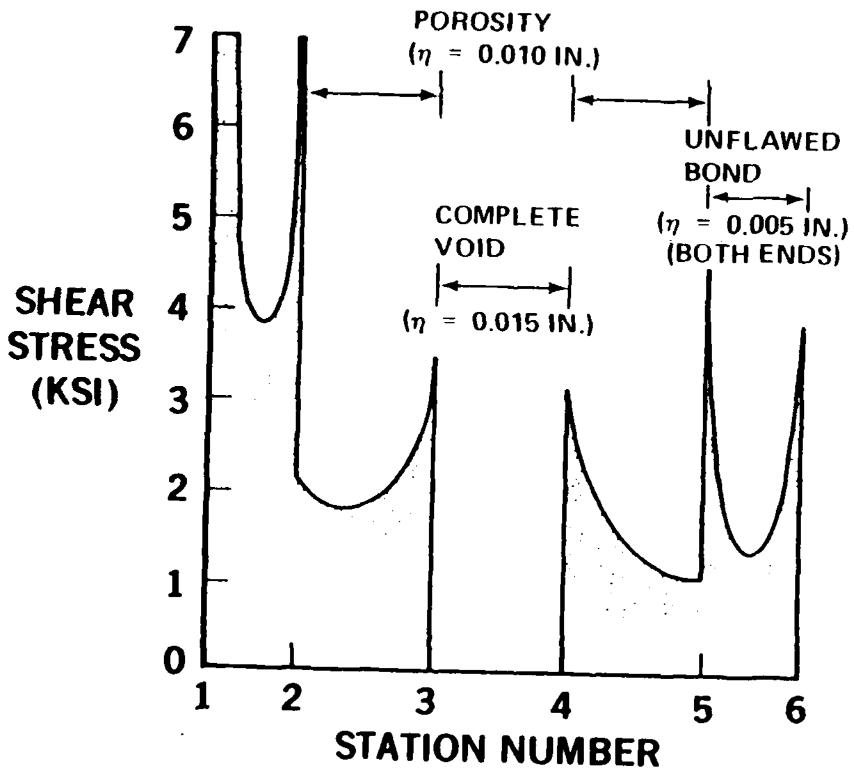


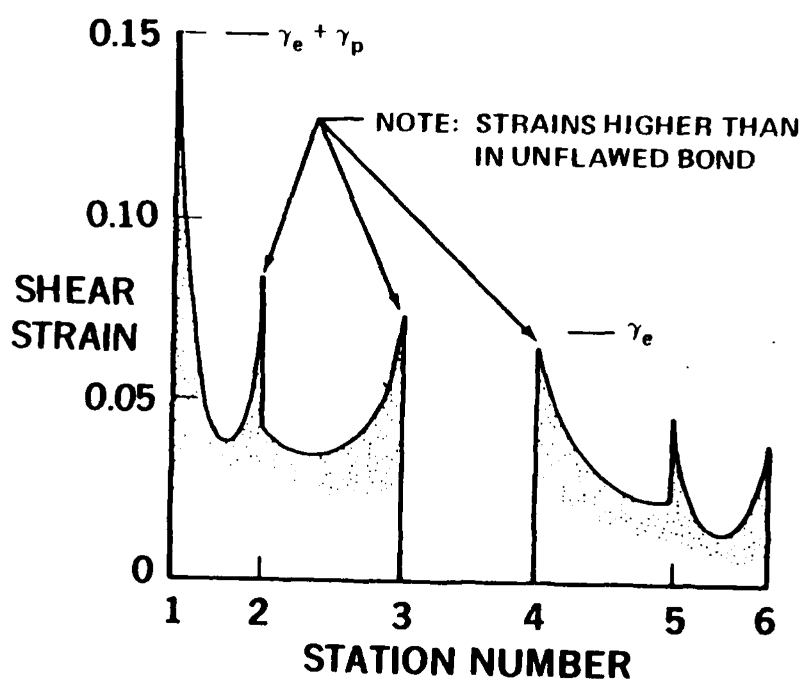
81 DP 8597

SHEAR STRESSES AND STRAINS IN FLAWED, POROUS BONDED JOINT

TENSILE LOAD
($\Delta T = -250°F$)

STRENGTH = 13,350 LB/IN.





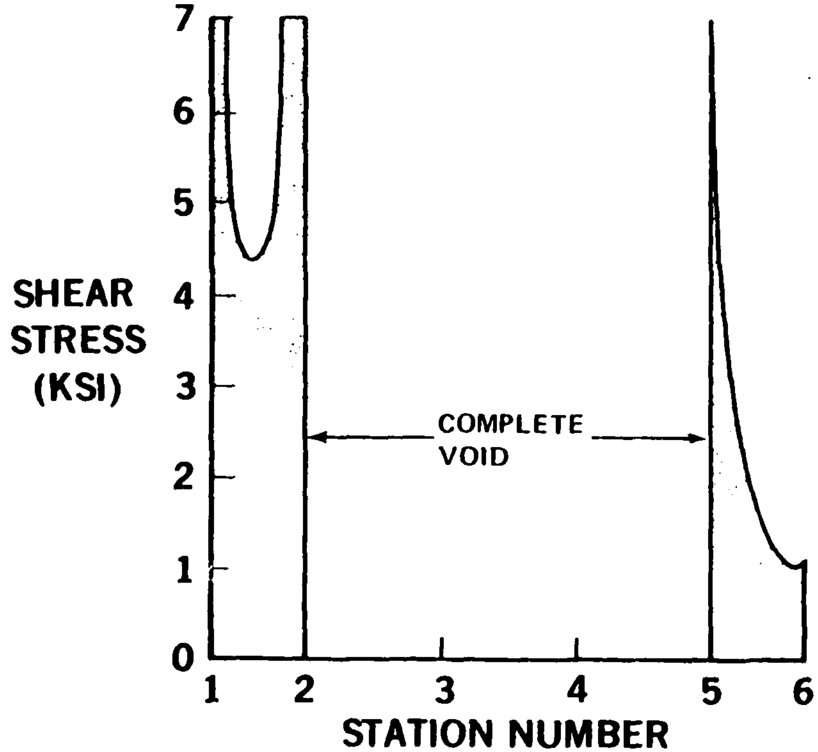
81 DP 8595

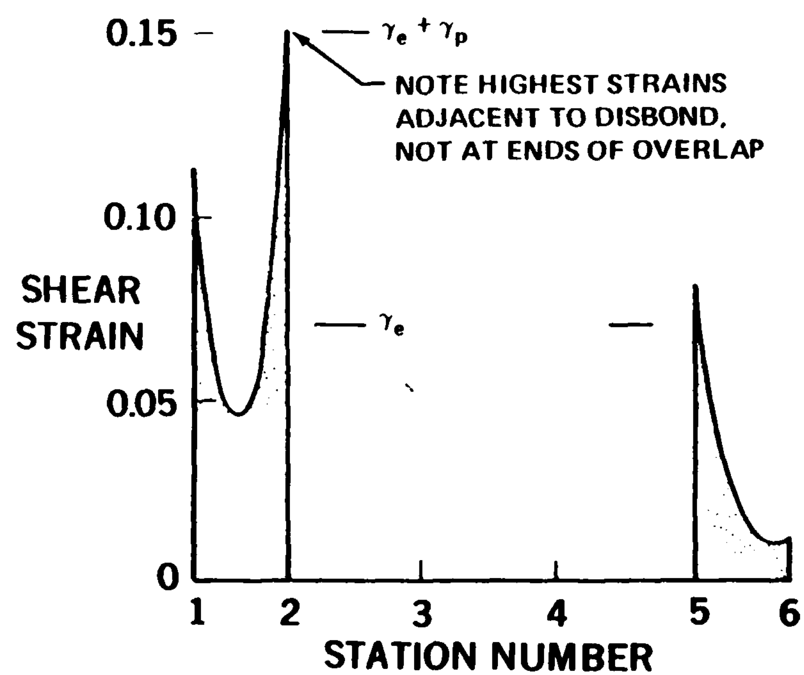
SHEAR STRESSES AND STRAINS IN FLAWED ADHESIVE-BONDED JOINT

TENSILE LOADING

($\Delta T = -250°F$)

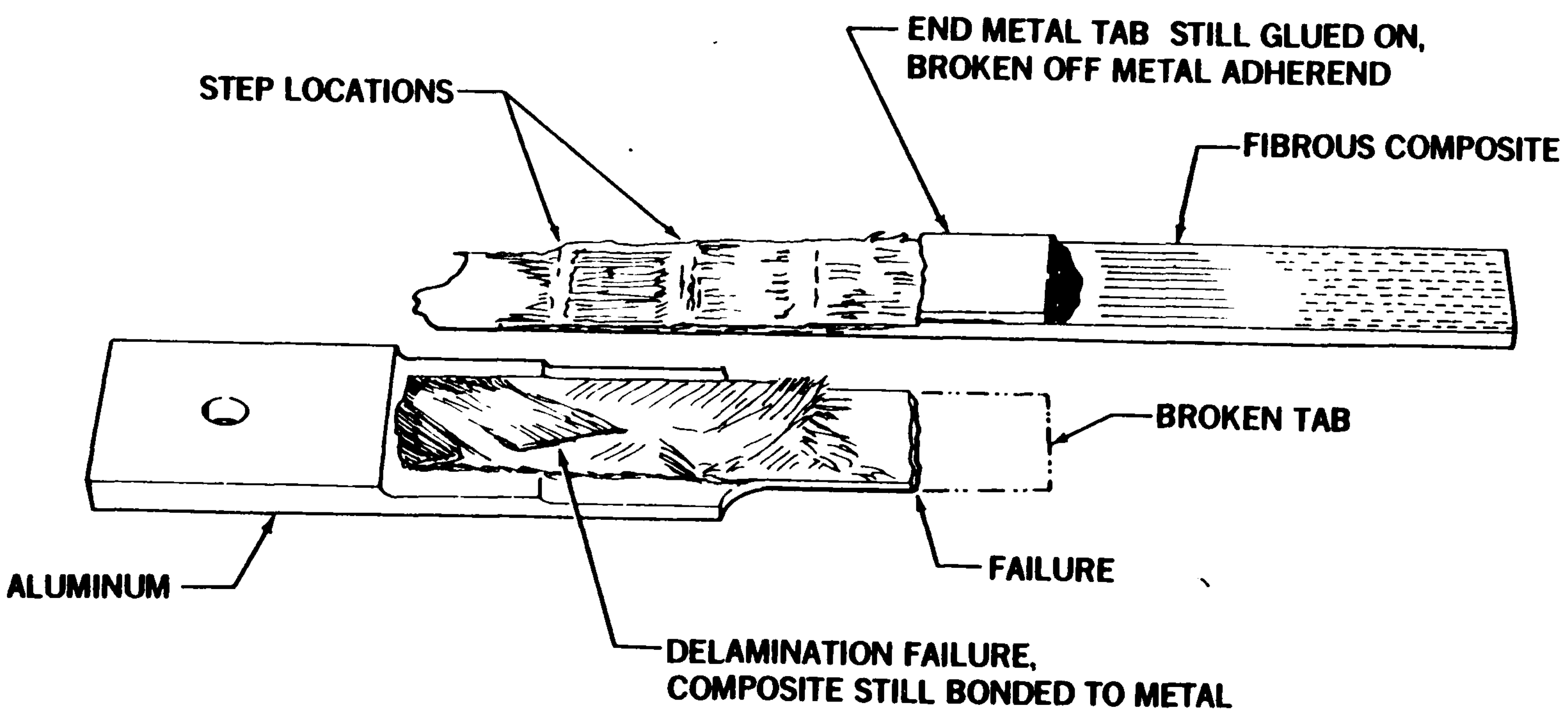
STRENGTH = 8350 LB/IN.





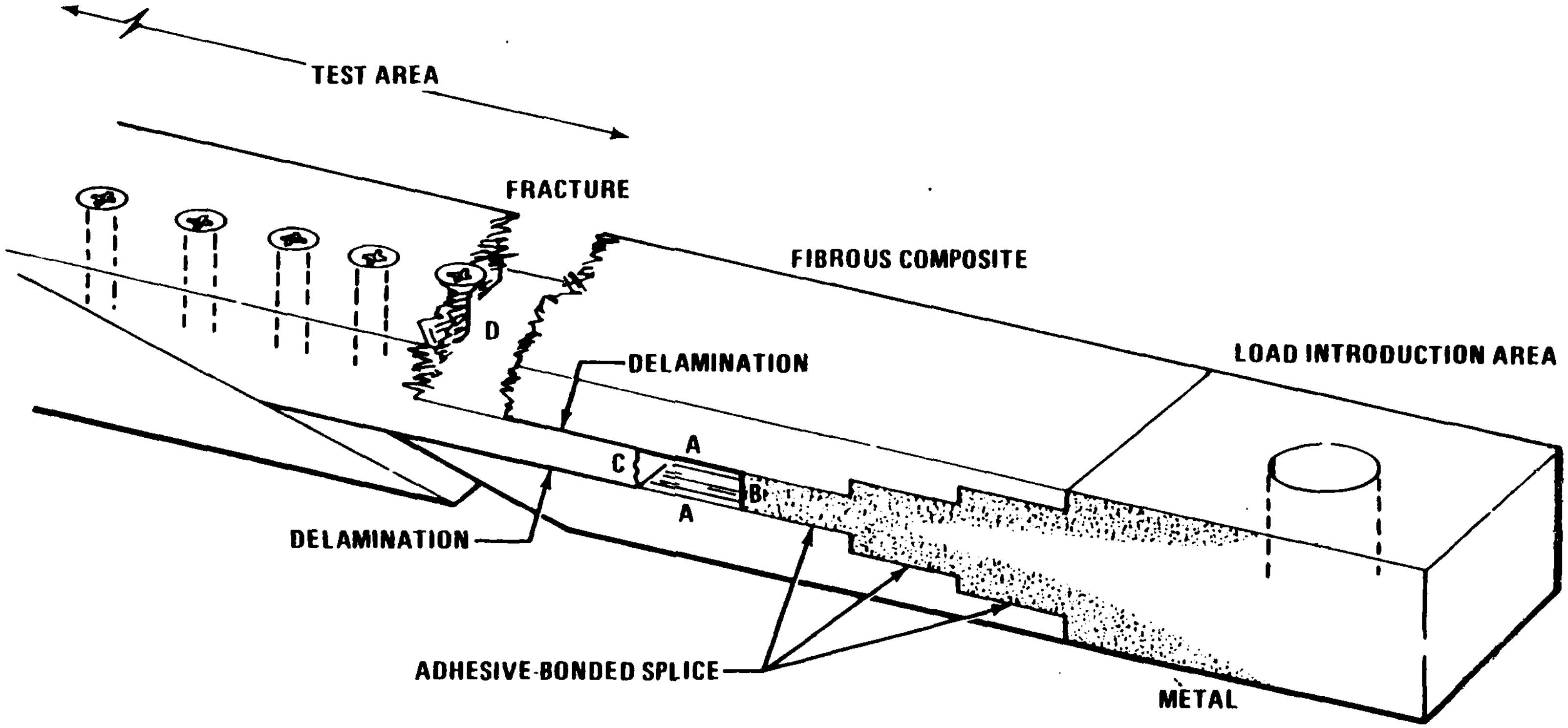
81 DP 8594

FAILURE OF STEPPED-LAP ADHESIVE-BONDED JOINT



81-DP 8193

PREMATURE FAILURE OF STEPPED-LAP BONDED JOINT BY DELAMINATION

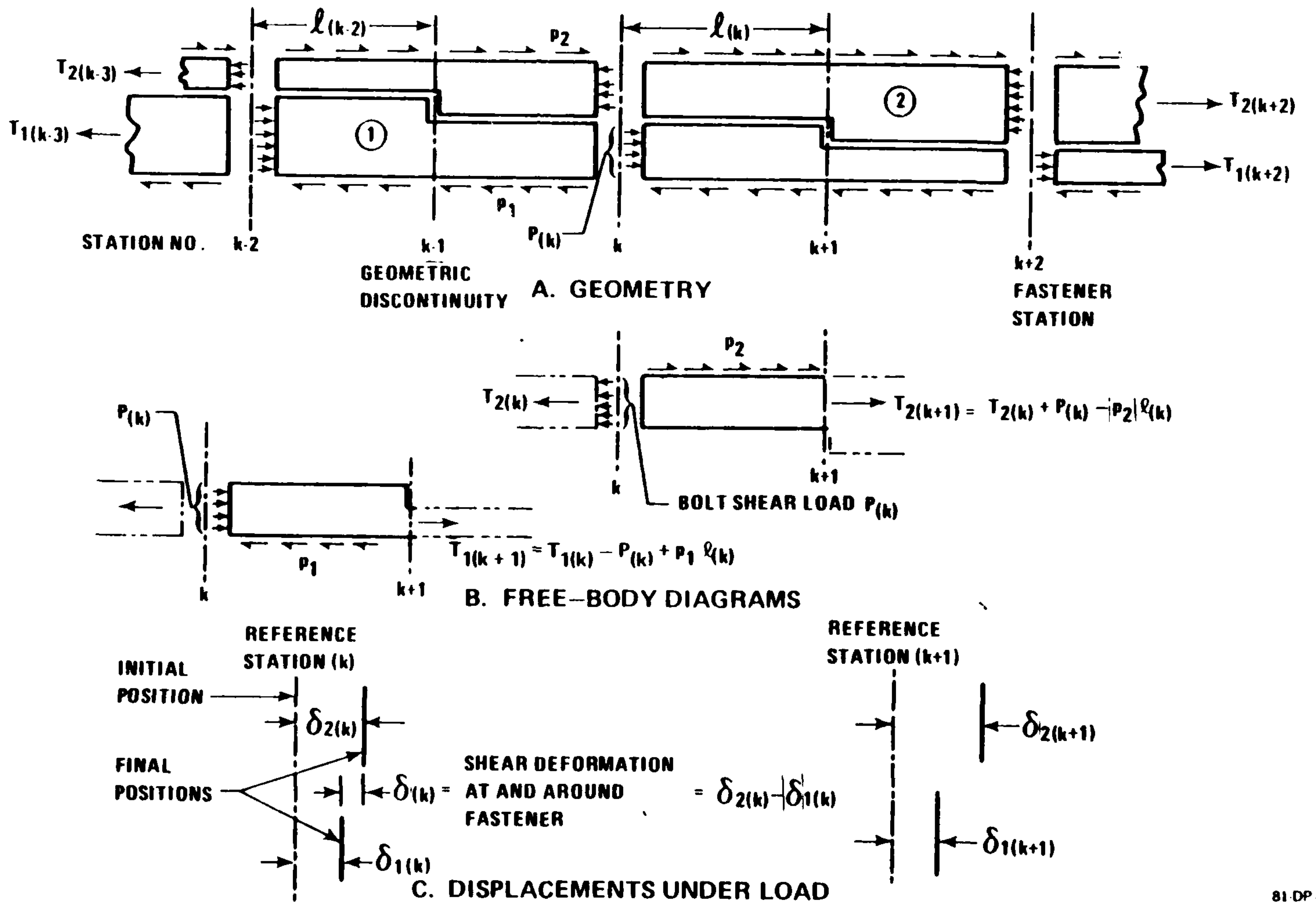


INITIAL FAILURE AT "A" BECAUSE THICKNESS "B" IS EXCESSIVE AND LOAD IN FIBERS "C" CANNOT BE UNLOADED THROUGH RESIN MATRIX

FINAL FAILURE, AT "D", IS BY NET SECTION TENSION ON THE TOP FACE AND SHEAROUT (NOT SHOWN) ON THE LOWER FACE

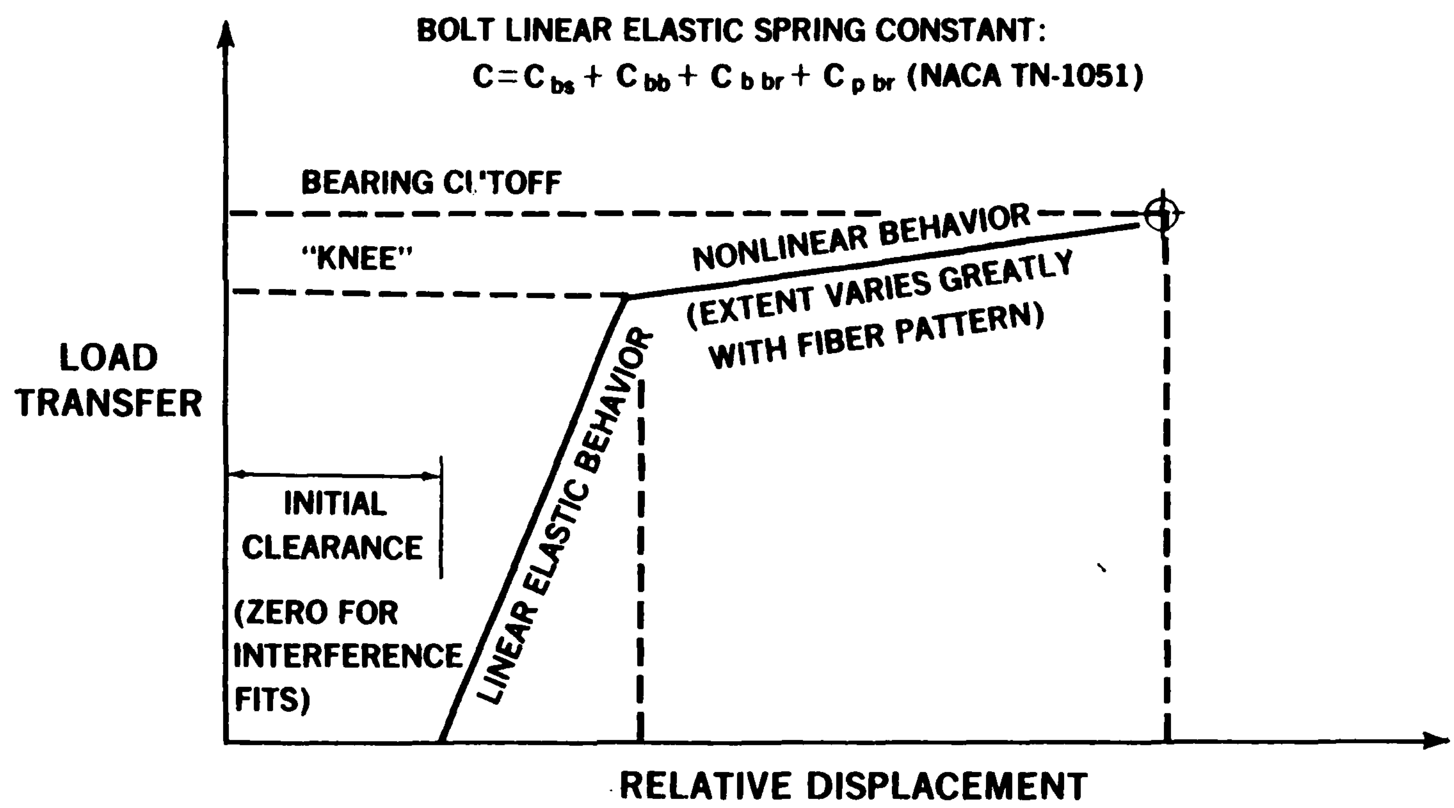
81 DP-8192

LOADS AND DEFORMATIONS ON ELEMENTS OF BOLTED JOINT



81-DP 8173

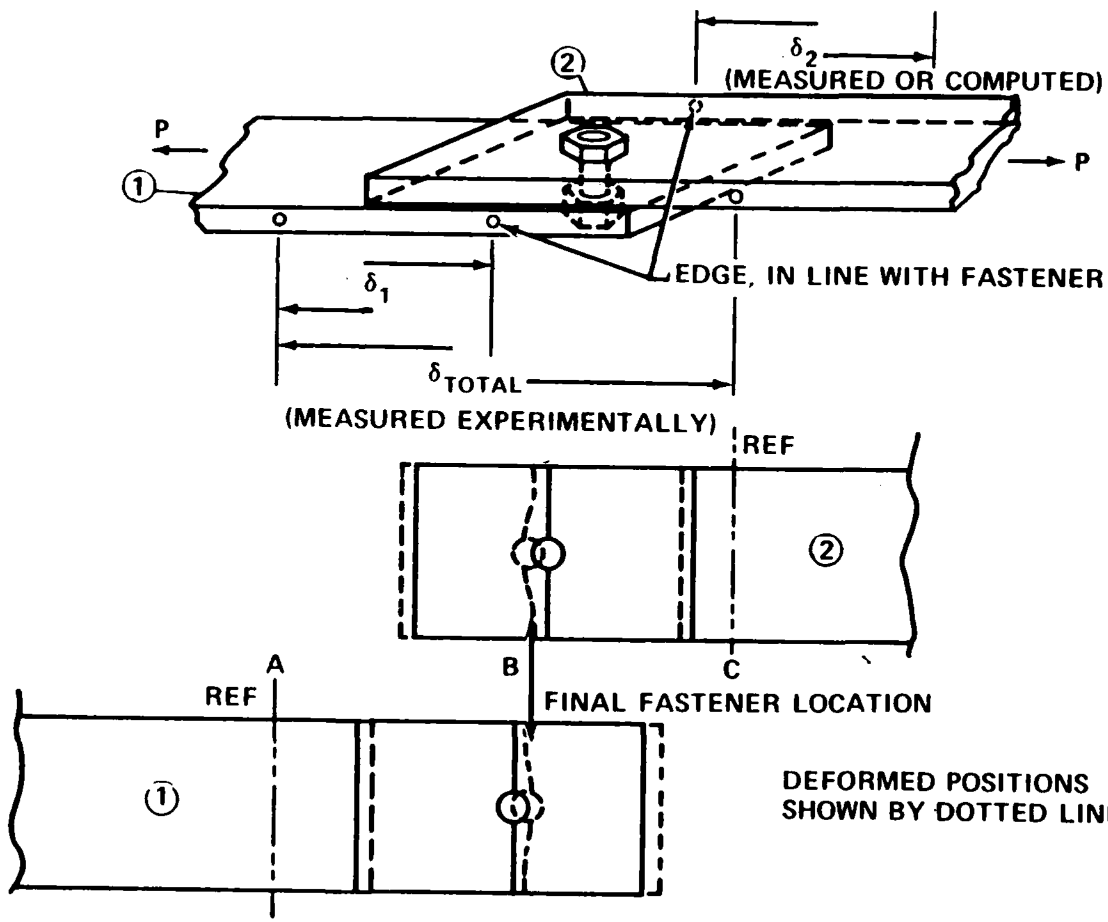
FASTENER LOAD-DEFLECTION CHARACTERISTICS



80-GEN 28102 A

DEFORMATIONS IN MECHANICALLY FASTENED JOINT

DELTAS SHOWN
(δ_1, δ_2, δ_{TOTAL})
REPRESENT CHANGES
IN LENGTH BETWEEN
STATIONS SHOWN,
NOT THE ACTUAL
LENGTHS



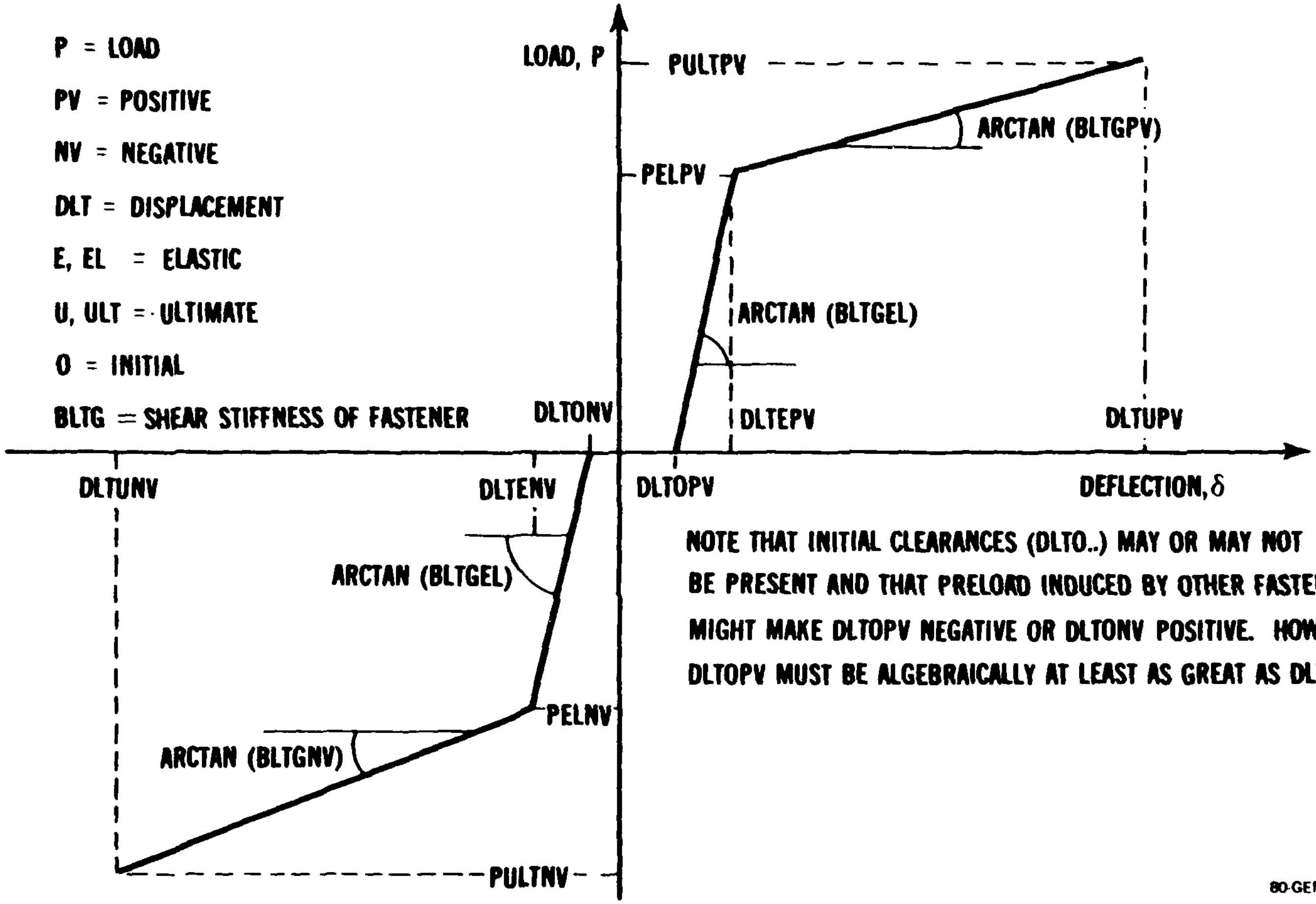
DEFORMED POSITIONS
SHOWN BY DOTTED LINES

EFFECTIVE FASTENER RELATIVE DISPLACEMENT $= \delta_{TOTAL} - \delta_1 - \delta_2$
AND INCLUDES DISTORTION OF CROSS SECTION AT FASTENER STATION

80 GEN 28105 A

IDEALIZED FASTENER
LOAD-DEFLECTION CHARACTERISTICS

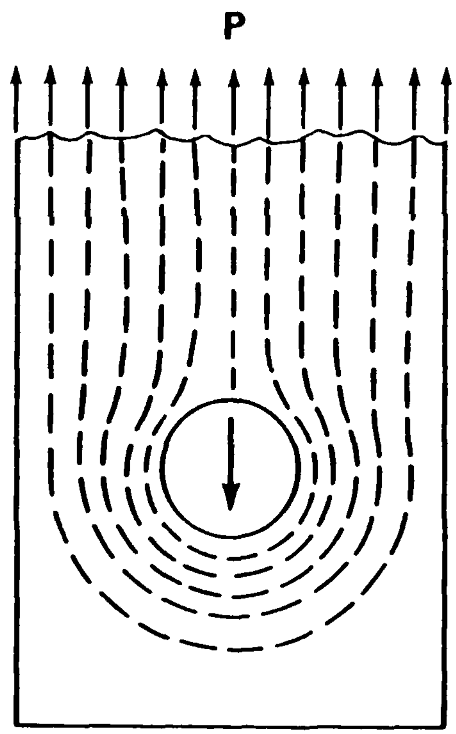
P = LOAD
PV = POSITIVE
NV = NEGATIVE
DLT = DISPLACEMENT
E, EL = ELASTIC
U, ULT = ULTIMATE
O = INITIAL
BLTG = SHEAR STIFFNESS OF FASTENER



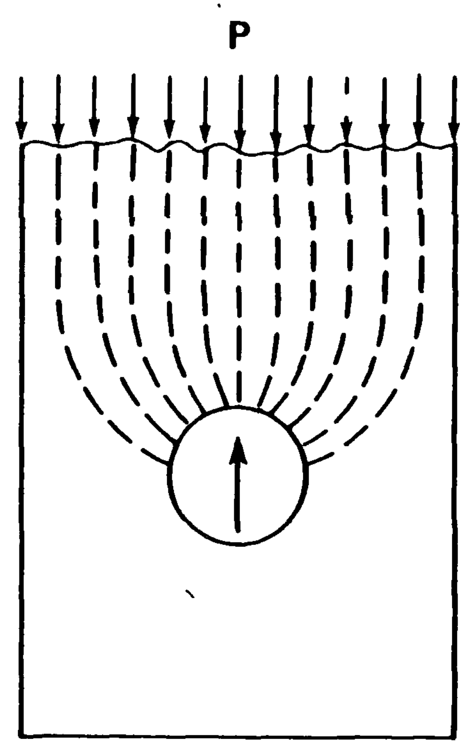
NOTE THAT INITIAL CLEARANCES (DLTO..) MAY OR MAY NOT BE PRESENT AND THAT PRELOAD INDUCED BY OTHER FASTENERS MIGHT MAKE DLTOPV NEGATIVE OR DLTONV POSITIVE. HOWEVER, DLTOPV MUST BE ALGEBRAICALLY AT LEAST AS GREAT AS DLTONV.

80-GEN-28100 A

STRESS TRAJECTORIES AROUND BOLTS FOR TENSILE AND COMPRESSIVE LAP SHEAR



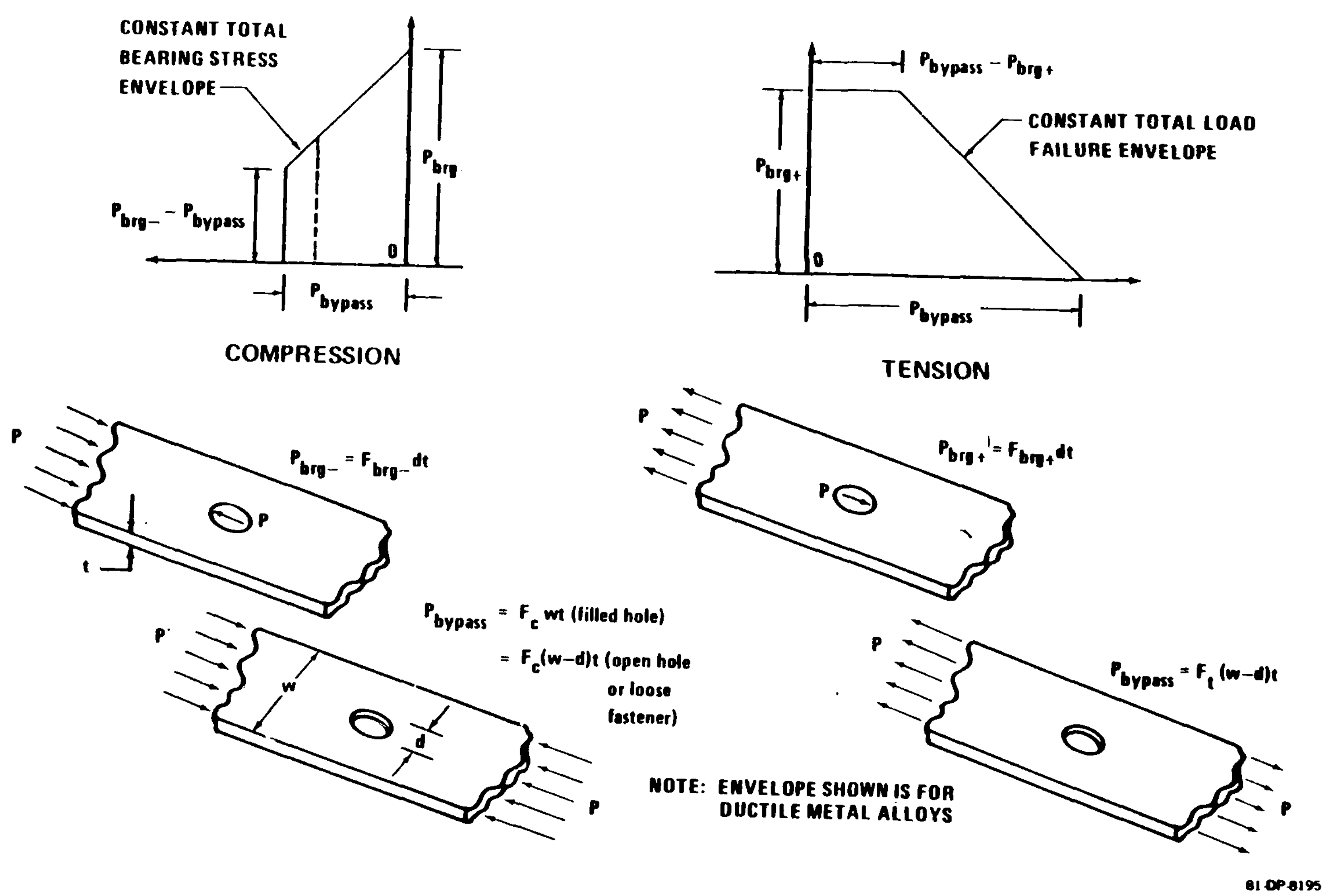
A. TENSILE LAP SHEAR



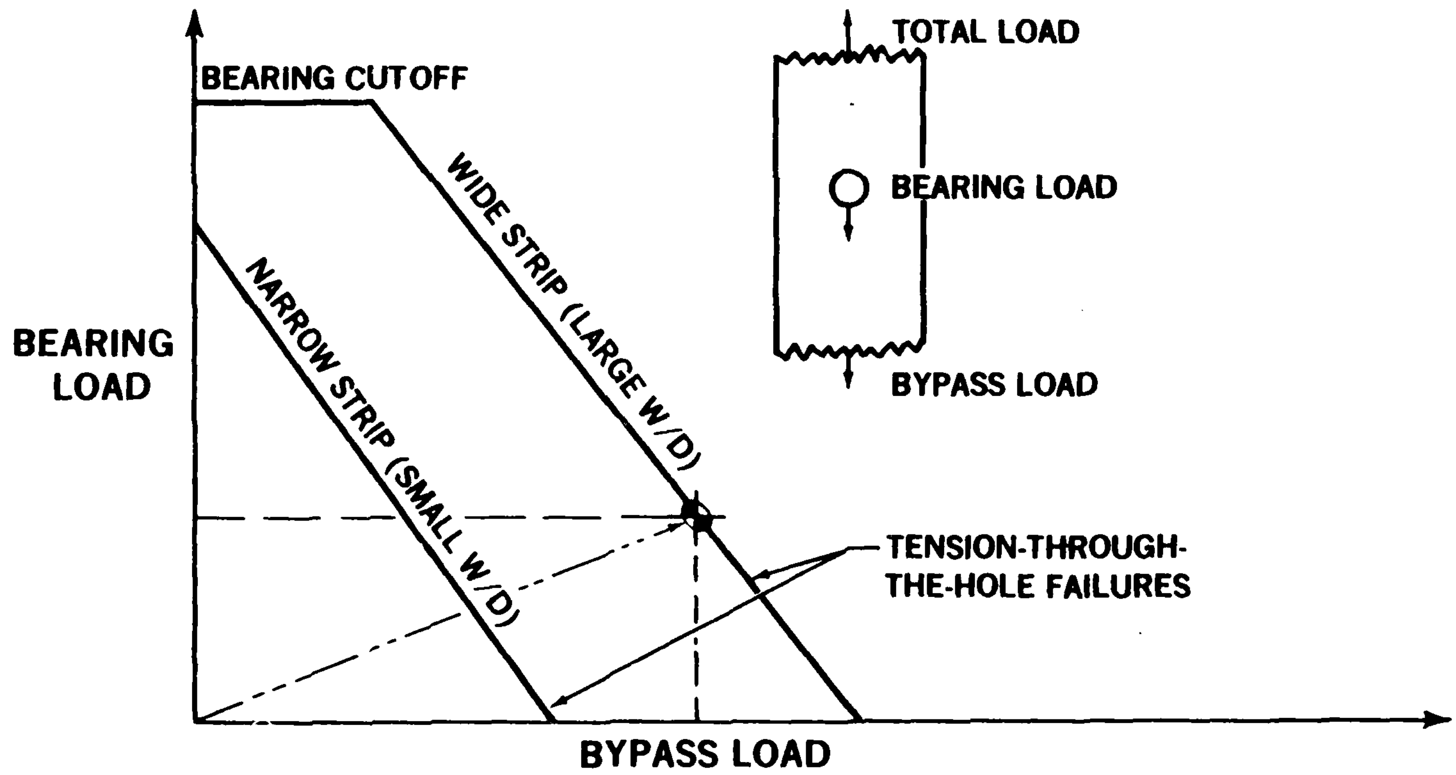
B. COMPRESSIVE LAP SHEAR

81 DP 8182

OUTER ENVELOPE OF BEARING-BYPASS LOAD INTERACTIONS

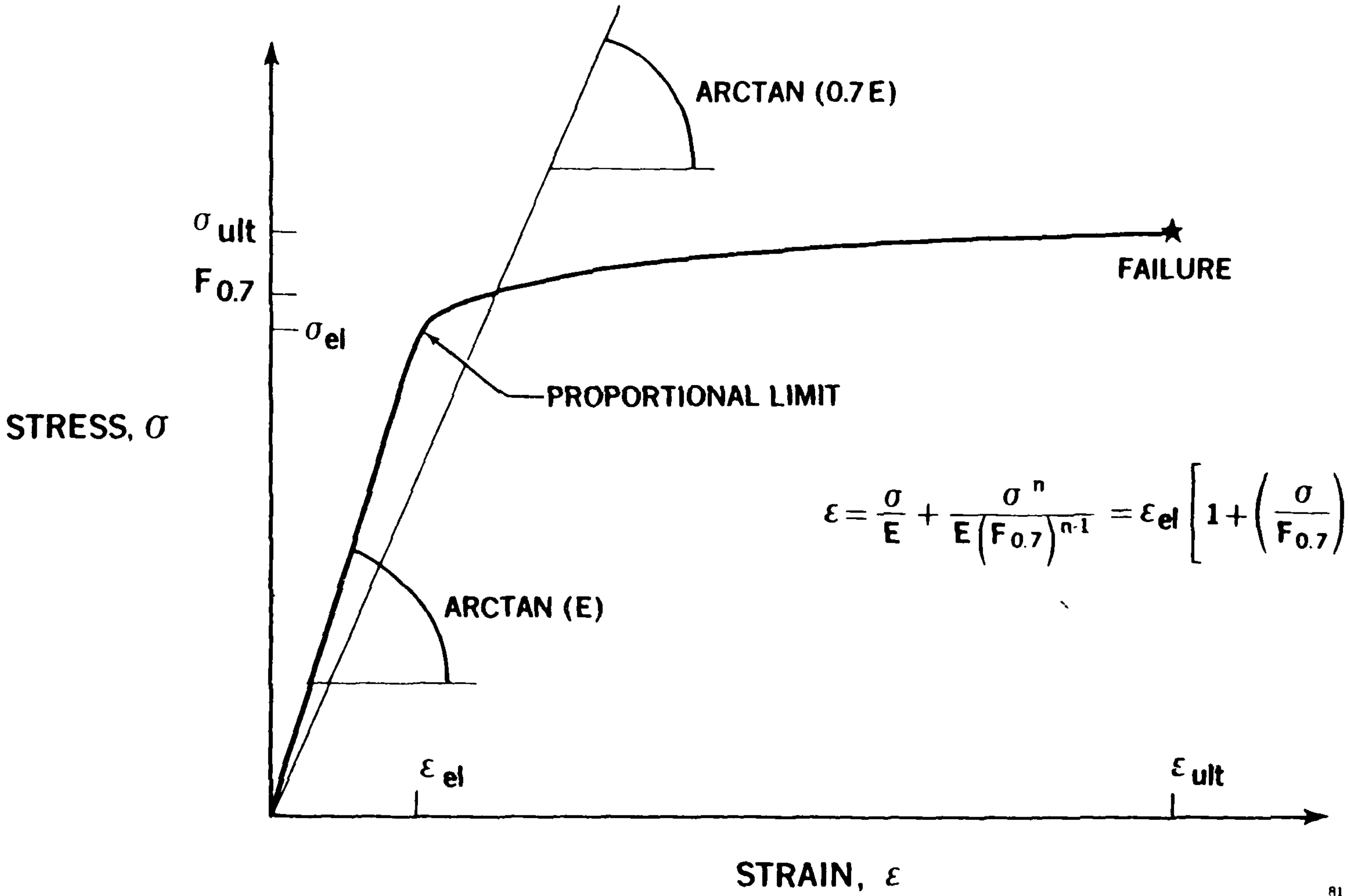


BEARING/BYPASS LOAD INTERACTION FOR LOADED BOLTS IN ADVANCED COMPOSITES



80 GEN-28101

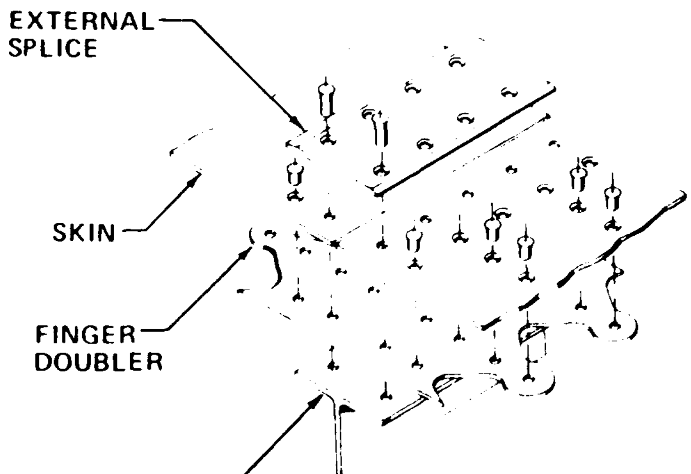
RAMBERG-OSGOOD NONLINEAR CHARACTERIZATION OF STRESS-STRAIN BEHAVIOR



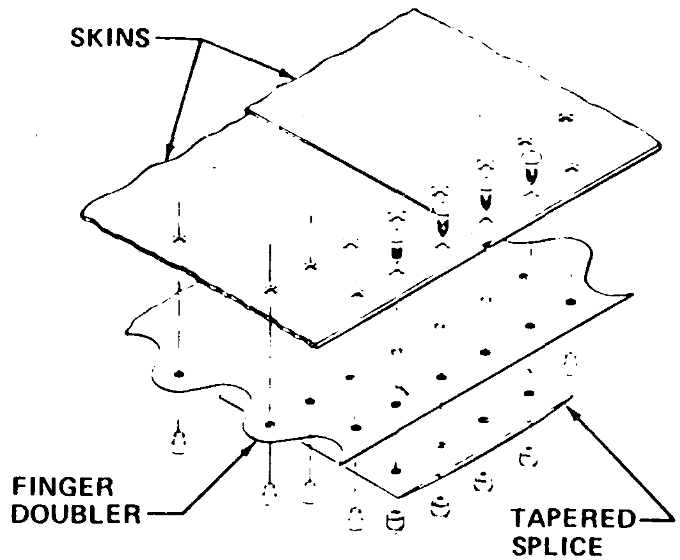
$$\varepsilon = \frac{\sigma}{E} + \frac{\sigma^n}{E\left(F_{0.7}\right)^{n-1}} = \varepsilon_{el}\left[1 + \left(\frac{\sigma}{F_{0.7}}\right)^{n-1}\right]$$

81 DP 8197

RIVETED FUSELAGE SKIN SPLICES



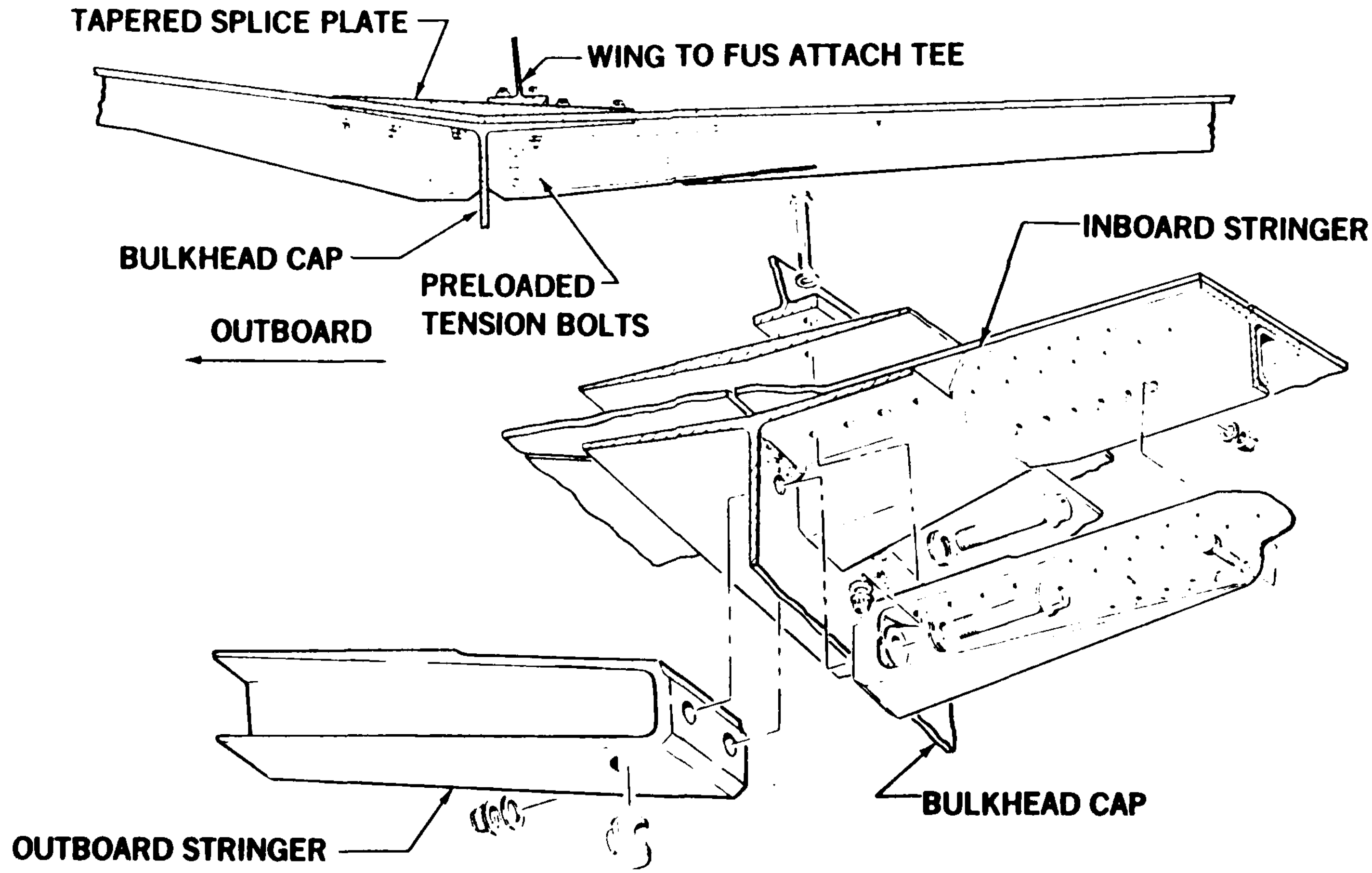
A. TYPICAL LONGITUDINAL SKIN SPLICE



B. TYPICAL TRANSVERSE SKIN SPLICE

81 DP 8313

WING PANEL JOINT AT SIDE OF FUSELAGE

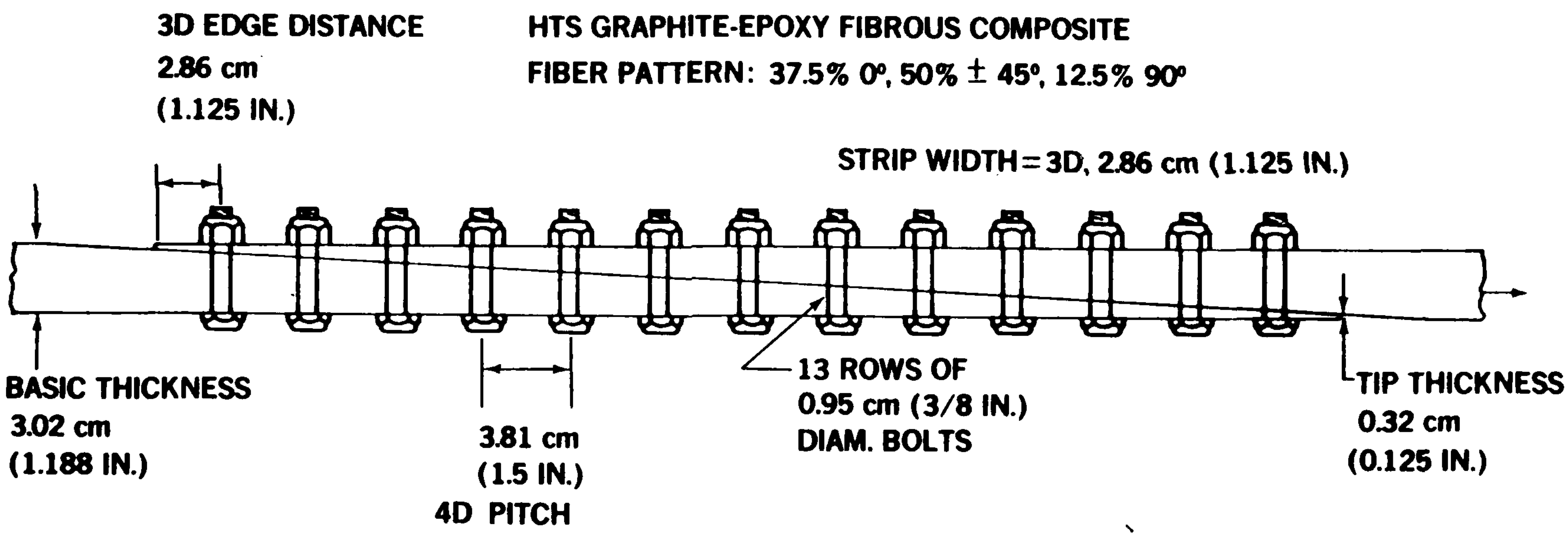


81 DP 8314

BOLTED COMPOSITE JOINT
SPAR CAP SPLICE

HTS GRAPHITE-EPOXY FIBROUS COMPOSITE

FIBER PATTERN: 37.5% 0°, 50% ± 45°, 12.5% 90°



PREDICTED FAILURE LOAD = 225.5 kN (50703 LB)

PREDICTED GROSS SECTION FAILURE STRAIN = 0.0039

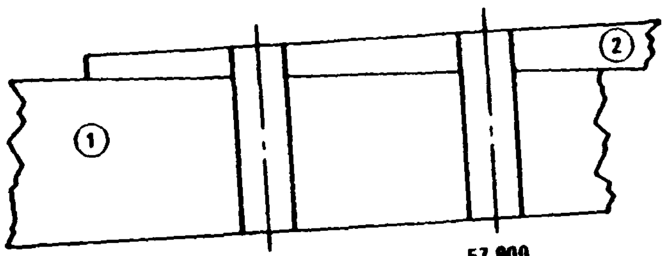
PREDICTED NET SECTION FAILURE STRESS = 349.4 MPa (50.7 ksi)

PREDICTED FAILURE MODE: TENSION THROUGH FIRST FASTENER HOLE

PEAK BOLT LOAD (AT ENDS) = 22.7 kN (5093 LB)
MINIMUM BOLT LOAD (IN MIDDLE) = 14.8 kN (3317 LB)
: RATIO 1.5

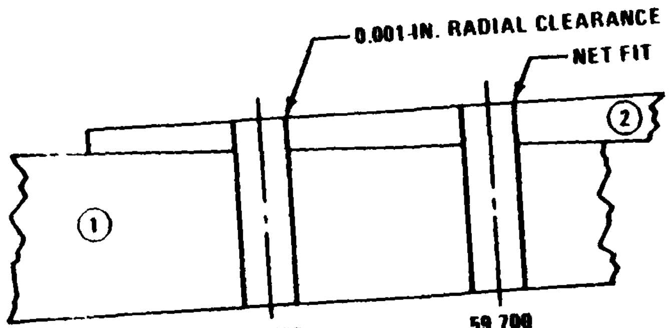
80-GEN-28107

INFLUENCE OF HOLE CLEARANCE ON STI ENGTH OF BOLTED JOINTS



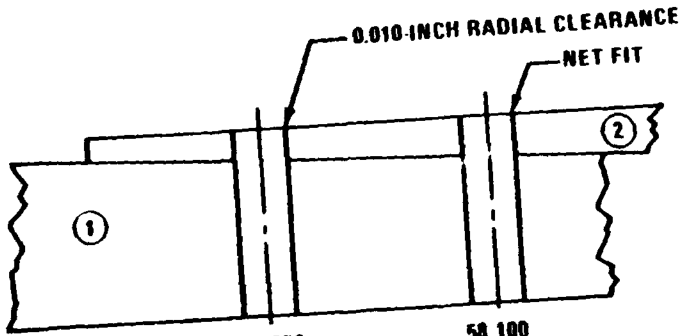
$\sigma_{net\ 1}$ (PSI)	56,900	57,900
$\epsilon_{gross\ 1}$	0.0039	0.0040
$\sigma_{brg\ 2}$ (PSI)	73,800	45,270

A. NET FIT FASTENERS



$\sigma_{net\ 1}$ (PSI)	52,700	59,700
$\epsilon_{gross\ 1}$	0.0038	0.0041
$\sigma_{brg\ 2}$ (PSI)	27,410	57,440

C. LOOSE FIT FASTENERS



$\sigma_{net\ 1}$ (PSI)	56,950	58,100
$\epsilon_{gross\ 1}$	0.0039	0.0040
$\sigma_{brg\ 2}$ (PSI)	72,300	45,770

B. CLOSE FIT FASTENERS

ULTIMATE JOINT STRENGTHS

A. 50,703 LB

B. 50,725 LB

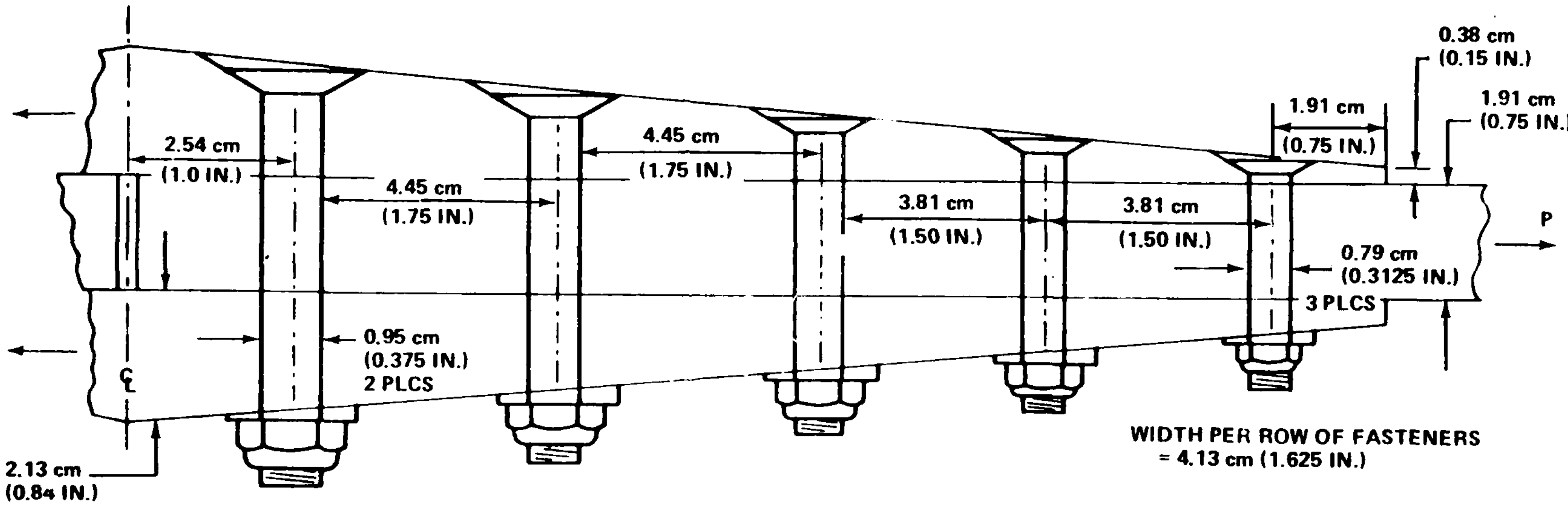
C. 45,858 LB

NOTE: THIS ILLUSTRATION IS AN ENLARGED DETAIL AT THE END OF THE JOINT IN THE PRECEDING FIGURE

81 DP 8174

BOLTED METAL JOINT

WING SKIN TEST SPLICE



• RESULTS OF NONLINEAR ANALYSIS AT APPLIED GROSS SECTION STRESS OF 275.8 MPa (40 KSI)

FASTENER	(kN)	51.6	37.0	34.2	39.8	54.4	
LOADS	(LB)	11,605	8,323	7,694	8,947	12,232	

NOTE BALANCE BETWEEN END FASTENER LOADS, DUE TO THICKENING OF SPLICE STRAPS

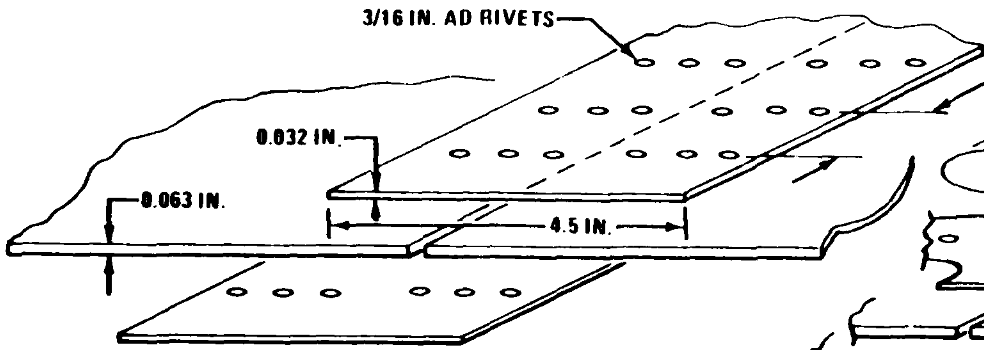
SKIN NET	(MPa)	85.4	146.6	193.5	256.2	341.9	276.1
SECTION STRESSES	(KSI)	12,379	21,256	28,060	37,149	49,575	40,041
SPLICE STRAP	(MPa)	179.5	169.6	164.7	165.6	151.6	
NET SECTION STRESSES	(KSI)	26,027	24,592	23,885	24,012	21,979	

PEAK BEARING STRESSES ON RIGHT END FASTENER: 360.0 MPa (52.2 KSI) IN SKIN; 636.5 MPa (92.3 KSI) IN SPLICE

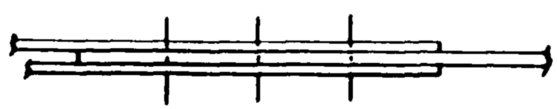
- TEST FAILURE IN SKIN THROUGH RIGHT END FASTENER, WHERE BOTH NET TENSION AND BEARING STRESSES ARE PREDICTED TO BE HIGHEST
- ANALYSIS PREDICTED ULTIMATE FAILURE AT GROSS SECTION SKIN STRESS OF 350.3 MPa (50.8 KSI)

80-GEN-28108

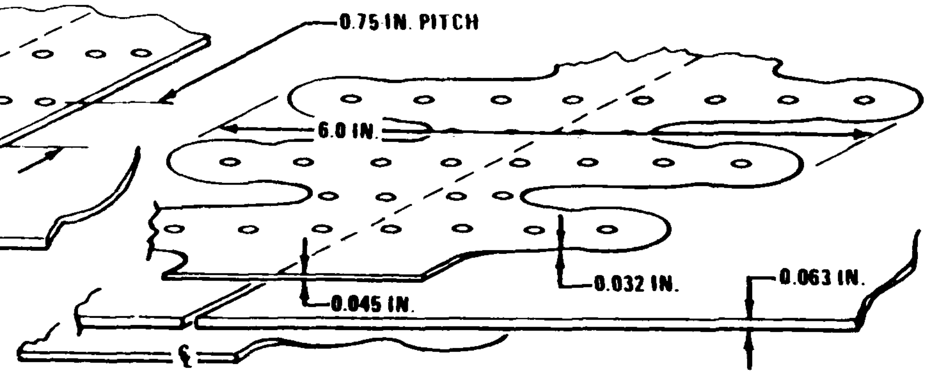
COMPARISON BETWEEN BASIC AND REFINED FUSELAGE SKIN SPLICES



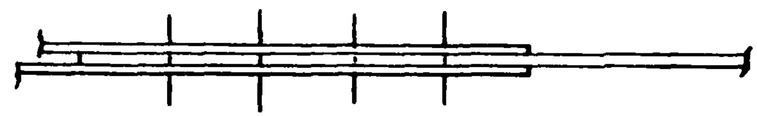
3 FULL ROWS OF RIVETS PER SIDE



RIVET LOAD (LB)	224	182	224	
BEARING STRESS (KSI)	19	15	19	
SKIN STRESS (KSI)	6.3	11.5	17.8	13.3



2 FULL AND 2 HALF ROWS OF RIVETS PER SIDE



RIVET LOADS (LB)	213	188	183	285	
SKIN BEARING STRESS (KSI)	18	16	16	24	
SPLICE BEARING STRESS (KSI)	12	11	16	24	
SKIN STRESS (KSI)	5.0	9.6	11.8	15.2	13.3
SPLICE STRESS (KSI)	10.7	14.9	9.0	5.6	

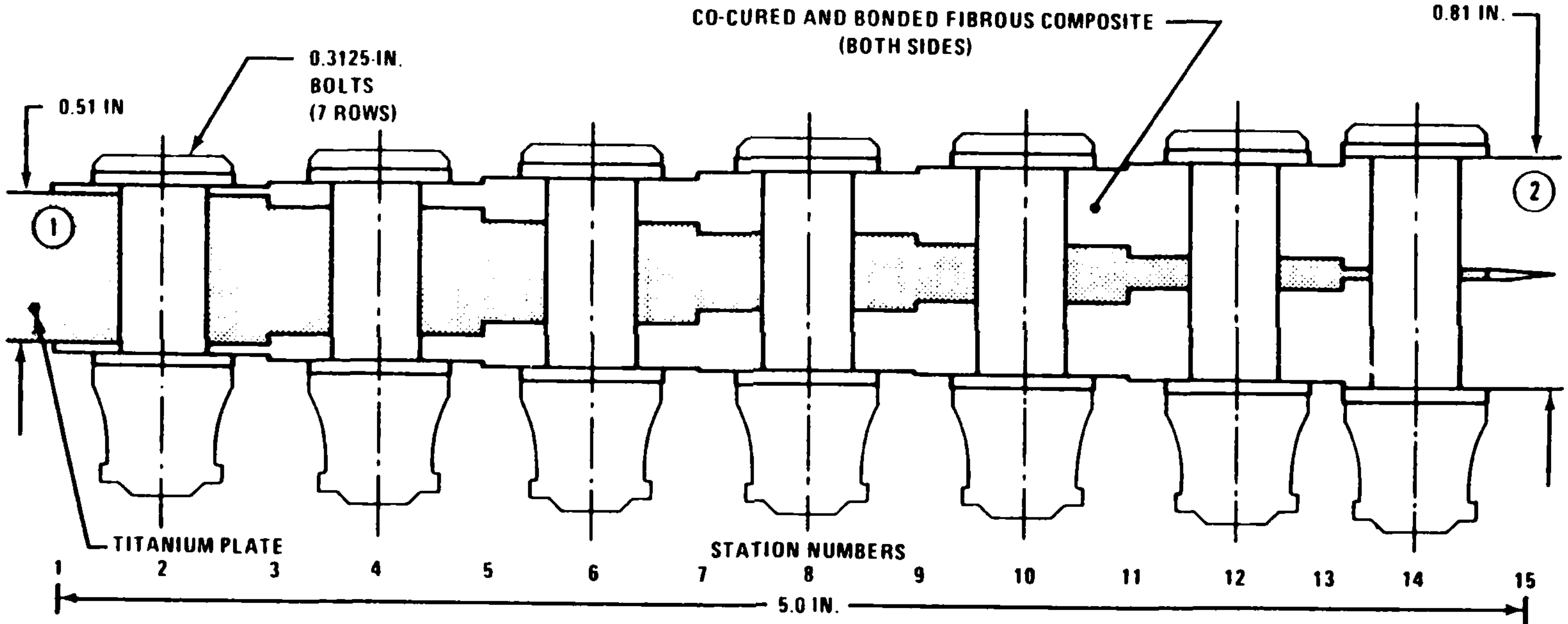
INTERNAL LOADS FOR COMMON 1-P LOAD OF 840 LB/IN.

NOTES:

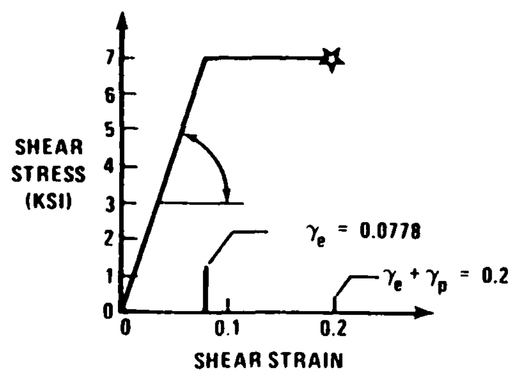
1. ULTIMATE STRENGTHS ARE 2929 LB/IN. FOR BASIC JOINT AND 3418 LB/IN. FOR LOLLIPOPPED SPLICE, CORRESPONDING WITH REMOTE SKIN STRESSES OF 46 AND 54 KSI, RESPECTIVELY.
2. EACH SPLICE HAS THE SAME NUMBER OF RIVETS.
3. FURTHER OTIMIZATION OF REFINED SPLICE COULD ENHANCE FATIGUE LIFE BY REDUCING END FASTENER LOAD, BUT ULTIMATE STRENGTH WOULD NOT BE IMPROVED.

81 DP 8169

STEPPED-LAP BONDED/BOLTED JOINT



A. SPECIMEN GEOMETRY



ADHESIVE THICKNESS = 0.005 IN.

B. ADHESIVE PROPERTIES

ADHEREND 1

MATERIAL: TITANIUM

$E = 16.0 \times 10^6$ PSI

$F_{tu} = 130$ KSI

$\alpha = 0.000006/^{\circ}F$

ADHERENDS 2

MATERIAL: HTS GRAPHITE-EPOXY ($\approx$ 45-PERCENT 0-DEG PLIES)

$E = 10.0 \times 10^6$ PSI

$F_{tu} = 100$ KSI

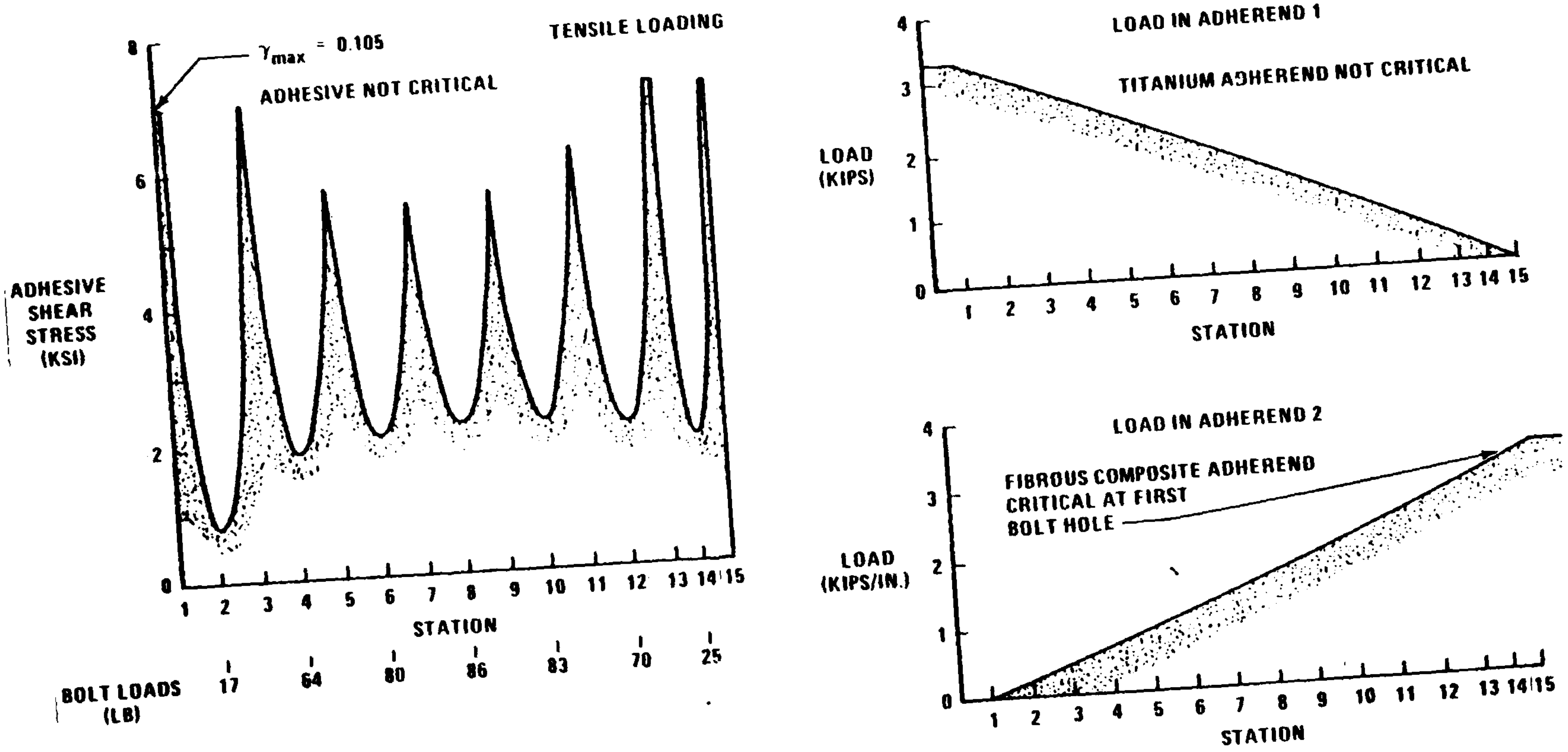
$\alpha = 0$

C. ADHEREND PROPERTIES

81-DP-8249

LOAD TRANSFER THROUGH BONDED/BOLTED STEPPED-LAP JOINT WITH NO FLAWS

NO RESIDUAL THERMAL STRESSES ACCOUNTED FOR ($\Delta T = 0$)

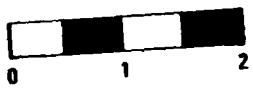


TOTAL LOAD TRANSFER = 33,096 LB (ON 1.0-IN. WIDTH)

ADHESIVE LOAD TRANSFER = 32,671 LB

BOLT LOAD TRANSFER = 425 LB (BARELY 1 PERCENT OF TOTAL)

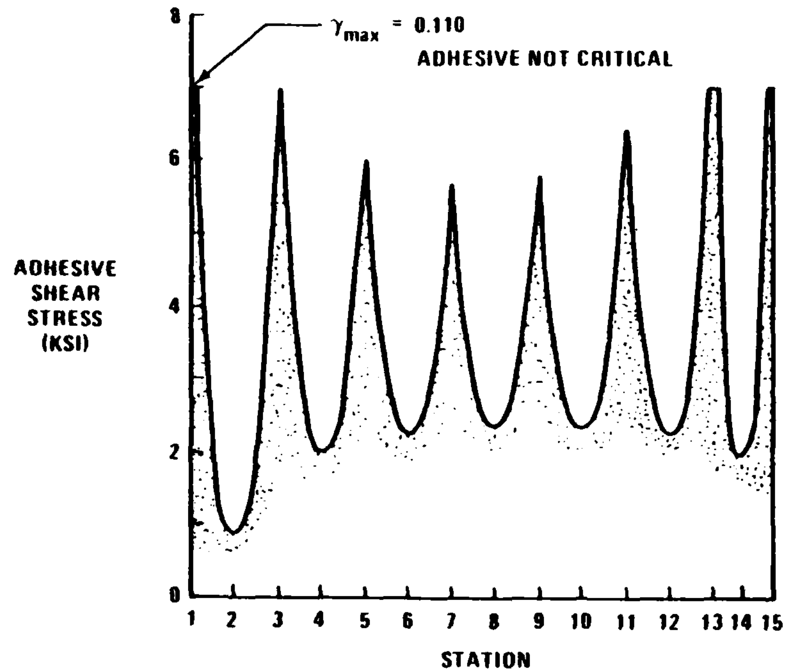
SCALE: (INCHES)

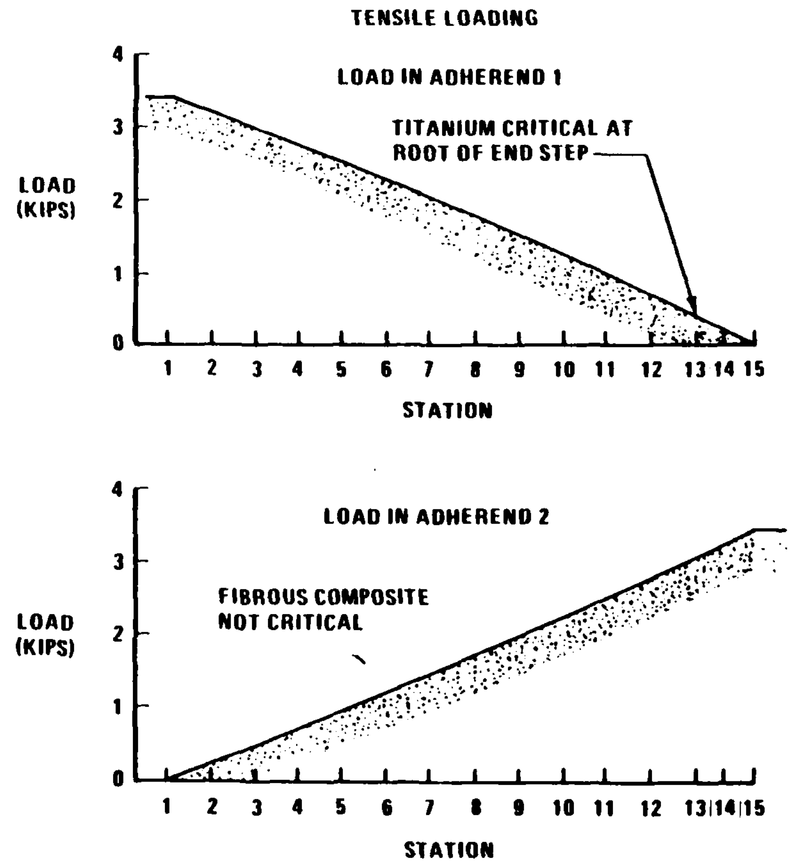


81 DP 8278

LOAD TRANSFER THROUGH ADHESIVE-BONDED STEPPED-LAP JOINT WITH NO FASTENERS

NO RESIDUAL THERMAL STRESSES ACCOUNTED FOR ($\Delta T=0$)





LOAD TRANSFER = 34,322 LB
(ON 1.0-IN. WIDTH)

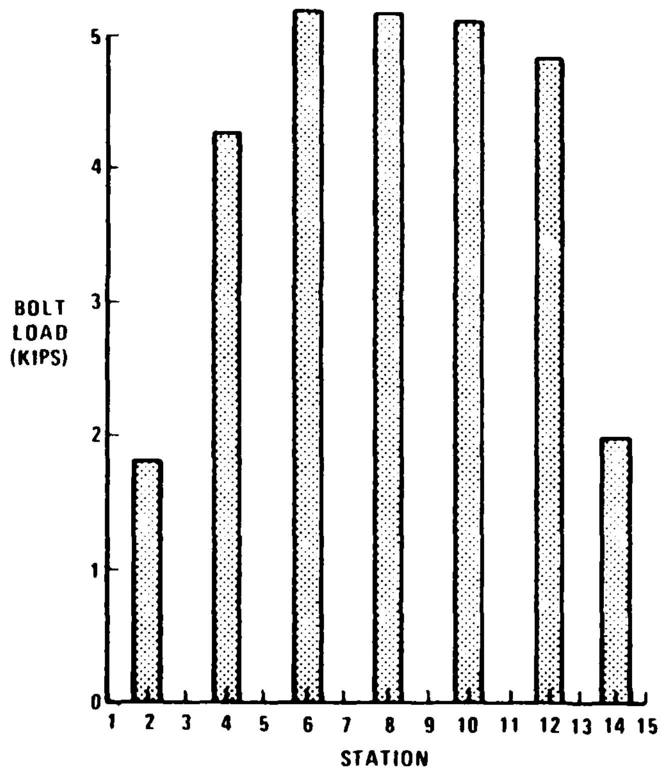
LOAD TRANSFER OF SAME JOINT, WITH SEVEN 0.3215-IN. FASTENERS ADDED = 33,096 LB

SLIGHT REDUCTION IN STRENGTH DUE TO BOLT HOLE IN THICK END OF FIBROUS COMPOSITE

SCALE: (INCHES) 0 1 2

81 DP 8274

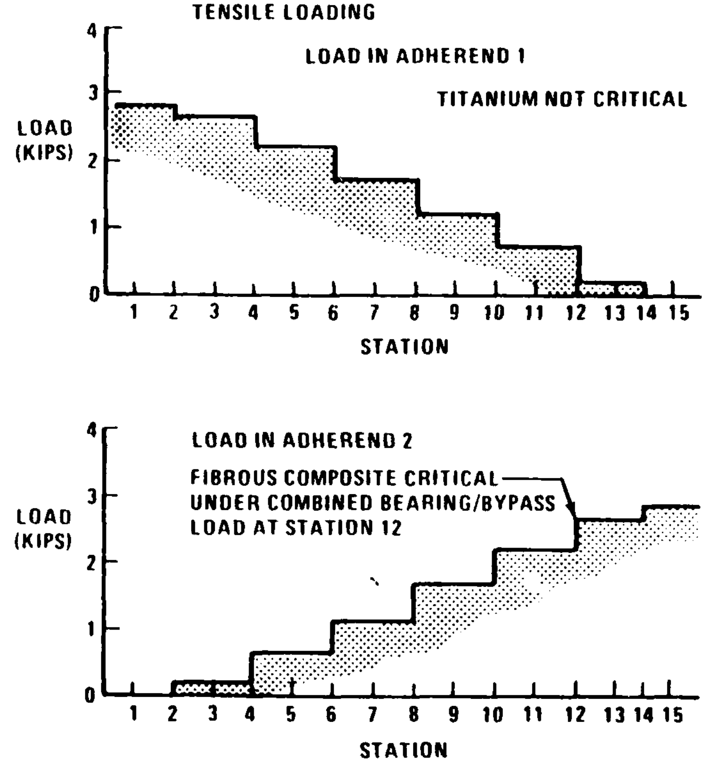
LOAD TRANSFER THROUGH BOLTED JOINT WITHOUT ANY ADHESIVE BOND



JOINT STRENGTH = 28,380 LB (ON 1.0-IN. WIDTH)

FASTENERS NOT CRITICAL
SHEAR STRENGTH = 5750 LB

NO RESIDUAL THERMAL STRESSES ACCOUNTED FOR ($\Delta T = 0$)

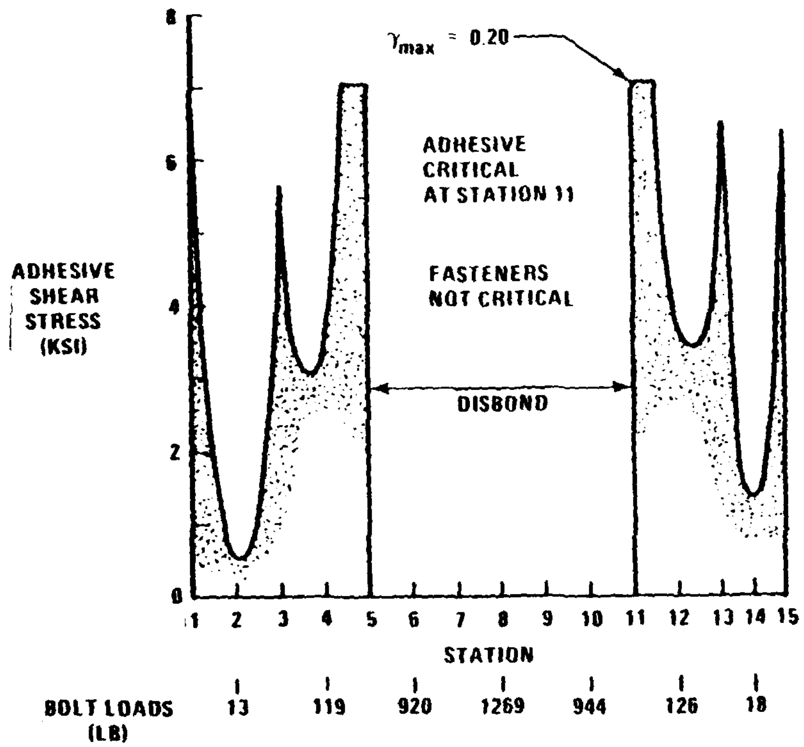


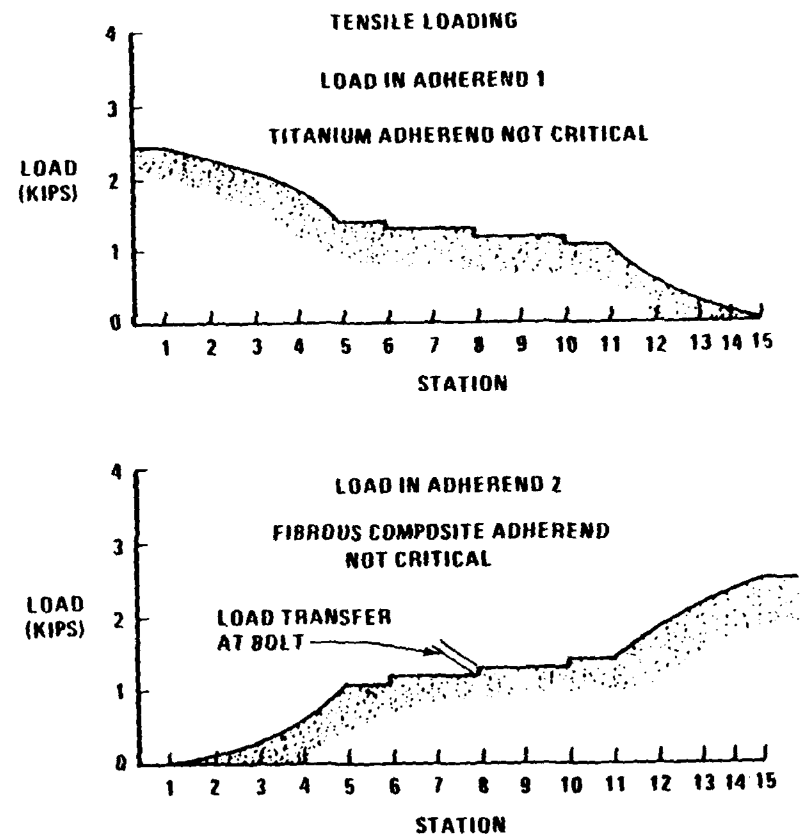
SCALE: (INCHES) 0 1 2

81 DP 8250

LOAD TRANSFER THROUGH FLAWED BONDED JOINT REINFORCED BY BOLTS

NO RESIDUAL THERMAL STRESSES ACCOUNTED FOR ($\Delta T = 0$)





TOTAL LOAD TRANSFER = 24,027 LB
(ON 1.0 IN. WIDTH)

ADHESIVE LOAD TRANSFER = 20,618 LB

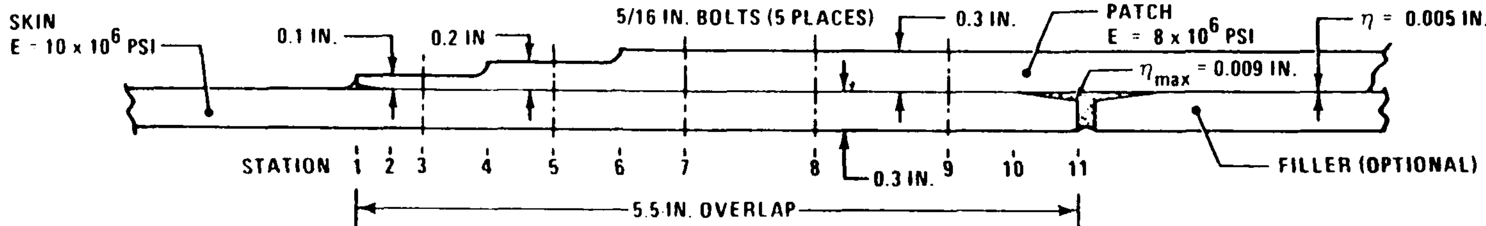
BOLT LOAD TRANSFER = 3409 LB
(14 PERCENT OF TOTAL)

COMPARABLE STRENGTH WITH ADHESIVE ALONE = 18,705 LB
STRENGTH OF UNFLAWED JOINT = 33,096 LB

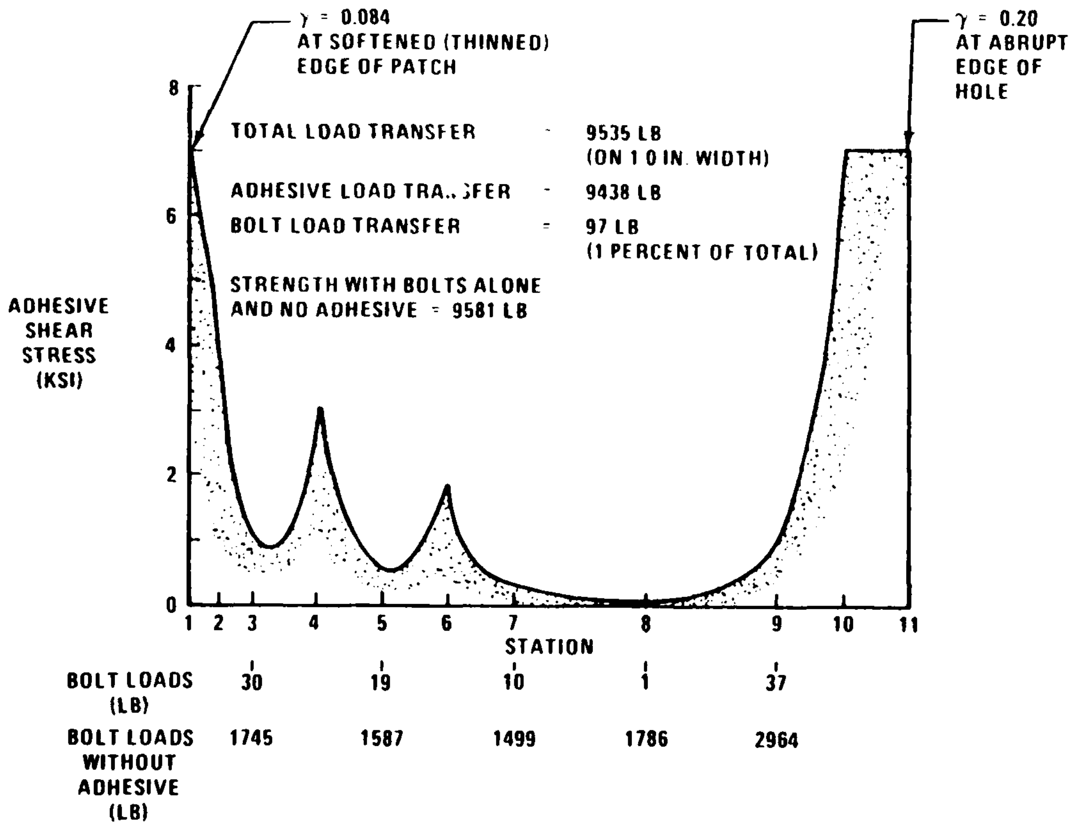
SCALE: (INCHES) 0 1 2

81 OP 8276

LOAD TRANSFER THROUGH BONDED/BOLTED FIBROUS COMPOSITE PATCH



PATCH GEOMETRY



STIFFNESS OF LOAD PATH ACROSS HOLE, THROUGH PATCH
$\infty E_{patch} = 8 \times 10^6$

STIFFNESS OF LOAD PATH AROUND HOLE, IN SKIN
$\infty E_{skin}/3 = 3.3 \times 10^6$

SO PATCH ACCEPTS 8/11.3 OF LOAD

HENCE, FAILURE PREDICTED AT LOAD OF
9,535 x 11.3/8 = 13,468 LB/IN.

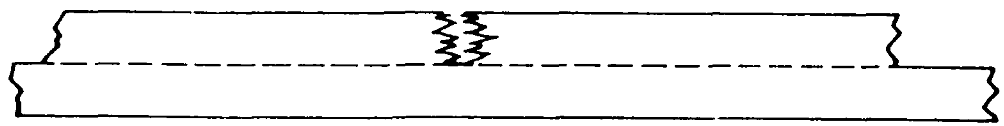
CORRESPONDING STRAIN IN SKIN

$$= \frac{13,468}{0.3 \times 10^7} = 0.0045$$

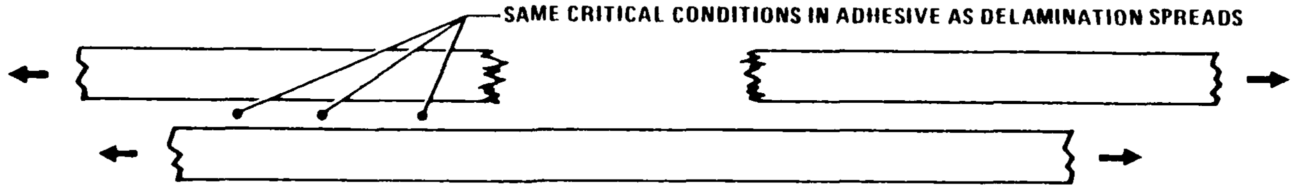
REPAIR IS THUS BACK TO NORMAL DESIGN OPERATING STRAINS, BUT NOT PARENT MATERIAL CAPABILITY

81 DP-8275

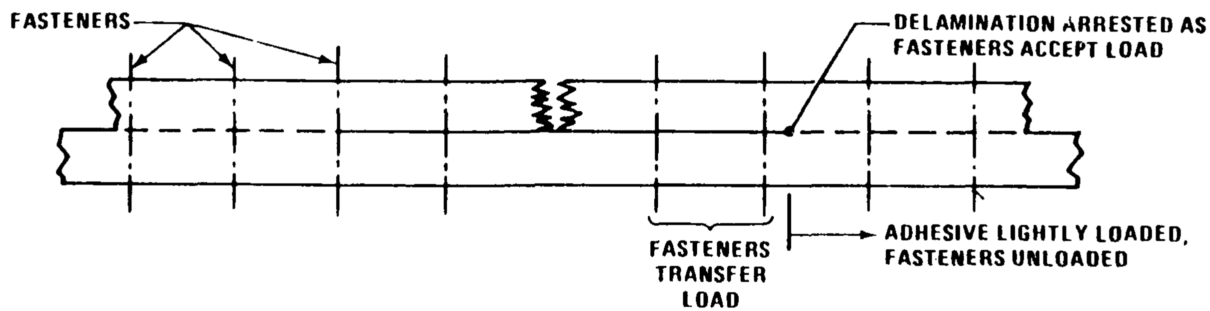
NEED FOR FAIL-SAFE FASTENERS IN THICK BONDED STRUCTURES



A. PARTIAL FRACTURE OF ADHESIVE-BONDED STRUCTURE



B. GROSS DELAMINATION OF THICK BONDED STRUCTURE WHEN ADHERENDS ARE STRONGER THAN ADHESIVE

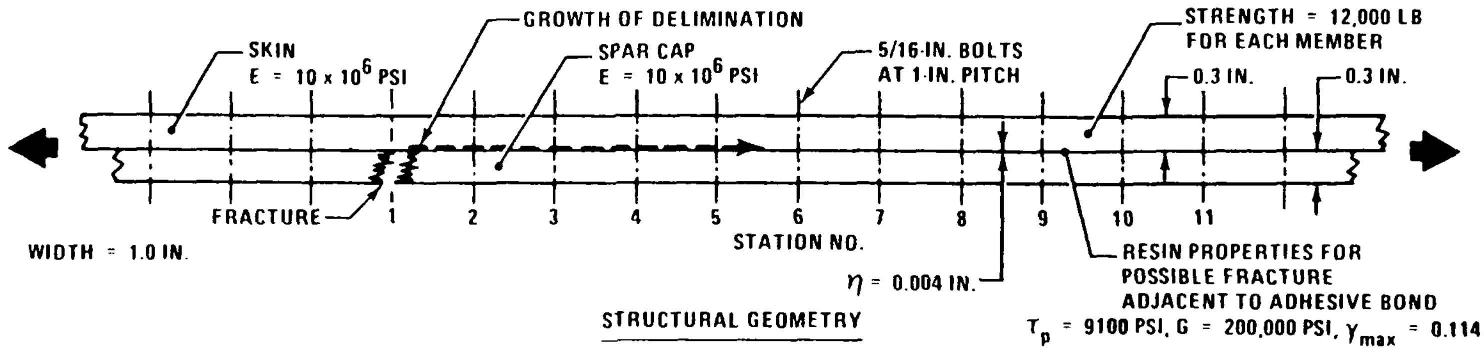


C. SELF-ARREST OF DELAMINATION WHEN FASTENERS ARE USED IN COMBINATION WITH ADHESIVE FOR HIGHLY LOADED STRUCTURE

NOTE: IF ADHESIVE WERE STRONGER THAN ADHERENDS, THERE WOULD BE NO DELAMINATIONS. THE SECOND MEMBER WOULD SIMPLY FAIL AT THE SAME LOCATION AS THE FIRST. IN THAT SITUATION, FASTENERS WOULD NEVER FEEL ANY LOAD AND WOULD BE SUPERFLUOUS

81 DP 8254

USE OF BOLTS AS FAIL-SAFE LOAD PATHS IN BONDED STRUCTURES



STRUCTURAL GEOMETRY

STATION NUMBER	1	2	3	4	5	6	7	8	9	10	11
1. ZERO DELAMINATION; FAILURE AT 6313 LB											
ADHESIVE SHEAR STRESS (KSI)	9100	74	0	0	0	0	0	0	0	0	0
FASTENER LOAD (LB)	0	1	0	0	0	0	0	0	0	0	0
2. 1.0-IN. DELAMINATION; FAILURE AT 6903 LB											
ADHESIVE SHEAR STRESS (KSI)	0	0	9100	74	0	0	0	0	0	0	0
FASTENER LOAD (LB)	0	295	1	0	0	0	0	0	0	0	0
3. 2.0-IN. DELAMINATION; FAILURE AT 10,468 LB											
ADHESIVE SHEAR STRESS (KSI)	0	0	0	9100	74	0	0	0	0	0	0
FASTENER LOAD (LB)	0	1783	295	1	0	0	0	0	0	0	0
4. 3.0-IN. DELAMINATION; FAILURE AT 12,000 LB*											
ADHESIVE SHEAR STRESS (KSI)	0	0	0	0	9100	38	0	0	0	0	0
FASTENER LOAD (LB)	0	2602	1137	163	0	0	0	0	0	0	0
5. 4.0-IN. DELAMINATION; FAILURE AT 12,000 LB*											
ADHESIVE SHEAR STRESS (KSI)	0	0	0	0	0	6530	20	0	0	0	0
FASTENER LOAD (LB)	0	2782	1394	608	84	0	0	0	0	0	0
6. NO ADHESIVE BOND; FAILURE AT 12,000 LB*											
FASTENER LOAD (LB)	0	2853	1496	785	412	216	113	58	29	12	0

*LIMITED BY ADHEREND STRENGTH

81-DP-8260

CONCLUSIONS

A4EI, A4EJ AND A4EK NONLINEAR COMPUTER PROGRAMS OFFER SIGNIFICANT INCREASE IN ANALYSIS AND DESIGN CAPABILITY FOR COMPOSITE JOINTS

VARIABLE ADHESIVE PROPERTIES — FLAWS, POROSITY, NONUNIFORM THICKNESS — ARE NEEDED TO CHARACTERIZE IMPERFECT BONDS

LOAD REDISTRIBUTION ASSOCIATED WITH BOND FLAWS IS CONFINED TO IMMEDIATE VICINITY OF FLAWS AND MAY NOT AFFECT CRITICAL CONDITIONS ELSEWHERE

BONDED JOINTS ARE FAR MORE TOLERANT OF FLAWS THAN NDI WOULD SUGGEST

STRENGTH OF MULTIROW BOLTED JOINTS DEPENDS CRITICALLY ON BEARING LOAD OF OUTERMOST FASTENERS, SO ACCURATE LOAD-SHARING ANALYSIS IS VITAL

NONLINEAR BEHAVIOR IN BOLTED JOINTS IS SIGNIFICANT FOR BOTH METAL AND COMPOSITE STRUCTURES

ADHESIVE BONDING AND MECHANICAL FASTENING DO NOT ACT IN UNISON BECAUSE OF DIFFERENT STIFFNESS — CANNOT SUM SEPARATE STRENGTHS

FASTENERS ARE USEFUL FOR REPAIRING DEFECTIVE BONDED JOINTS AND PROVIDING DAMAGE TOLERANCE FOR BONDED THICK STRUCTURES

ADHESIVE BONDING PROVIDES FAIL SAFETY FOR RIVETED THIN STRUCTURES

82 PAB 770

INTERPLY LAYER PROGRESSIVE WEAKENING EFFECTS ON COMPOSITE STRUCTURAL RESPONSE

BY

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8TH ANNUAL MECHANICS OF COMPOSITES REVIEW
OCTOBER 5 - 7, 1982, DAYTON, OHIO

OBJECTIVE:

COMPUTATIONALLY DETERMINE AND ASSESS THE EFFECTS OF INTERPLY LAYER PROGRESSIVE DEGRADATION ON THE STRUCTURAL RESPONSE OF A LAYERED COMPOSITE BEAM.

THE STRUCTURAL RESPONSE OF INTEREST INCLUDES:

- BENDING/FLEXURAL (3-POINT)
- BUCKLING
- FREE VIBRATIONS
- PERIODIC EXCITATIONS (FORCED VIBRATIONS)
- IMPACT

CONCLUSIONS:

GENERAL:

THE PROGRESSIVE DEGRATION EFFECTS OF THE INTERPLY LAYER ON COMPOSITE STRUCTURAL RESPONSE CAN BE DETERMINED AND ASSESSED USING FINITE ELEMENT ANALYSIS

SPECIFIC:

- INTERPLY DEGRADATION EFFECTS ON STRUCTURAL RESPONSE ARE NEGLIGIBLE FOR $E > 0.2 \times 10^6$ psi AND INCREASE VERY RAPIDLY FOR $E < 0.2 \times 10^6$ psi
- THE COMPOSITE RESPONDS AS INDIVIDUAL LAYERS FOR $E \leq 0.1 \times 10^6$ psi
- THE BUCKLING MODE IS NOT AFFECTED BY INTERPLY DEGRADATION EVEN FOR $E \approx 0.01 \times 10^6$ psi
- THROUGH THE THICKNESS VIBRATION MODES ARE NEGLIGIBLE FOR $E > 0.01 \times 10^6$ psi

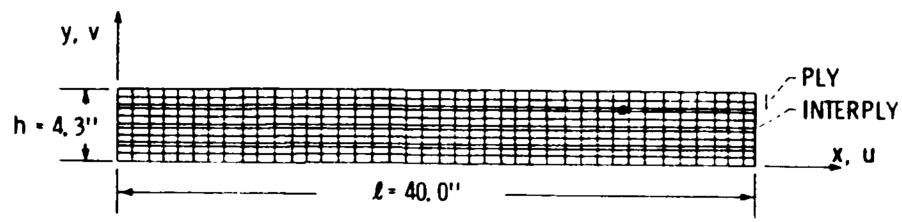
SPECIFIC CONCLUSIONS CONT'D.

- INTERPLY DEGRADATION HAS NEGLIGIBLE EFFECTS ON FORCED VIBRATION RESPONSE FOR $E > 0.2 \times 10^6$ psi WHEN DAMPING IS INCLUDED
- INTERPLY DEGRADATION HAS NOTICEABLE EFFECTS ON IMPACT RESPONSE FOR $E \leq 0.1 \times 10^6$ psi

<u>FINALLY</u>:

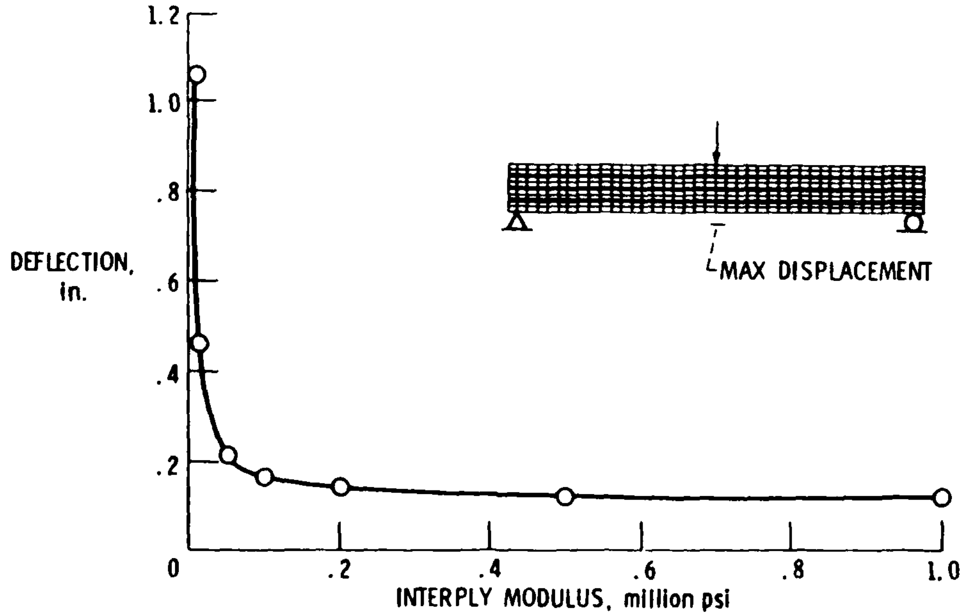
- CONSIDERABLE INTERPLY DEGRADATION MUST OCCUR (REDUCE INTERPLY MODULUS TO $\approx 0.01 \times 10^6$ psi) IN ORDER TO APPRECIABLY AFFECT/DEGRADE THE STRUCTURAL INTEGRITY OF LAYERED COMPOSITES AS BY: BENDING, BUCKLING, VIBRATION (FATIGUE), AND IMPACT

FINITE ELEMENT MODEL AND COMPOSITE



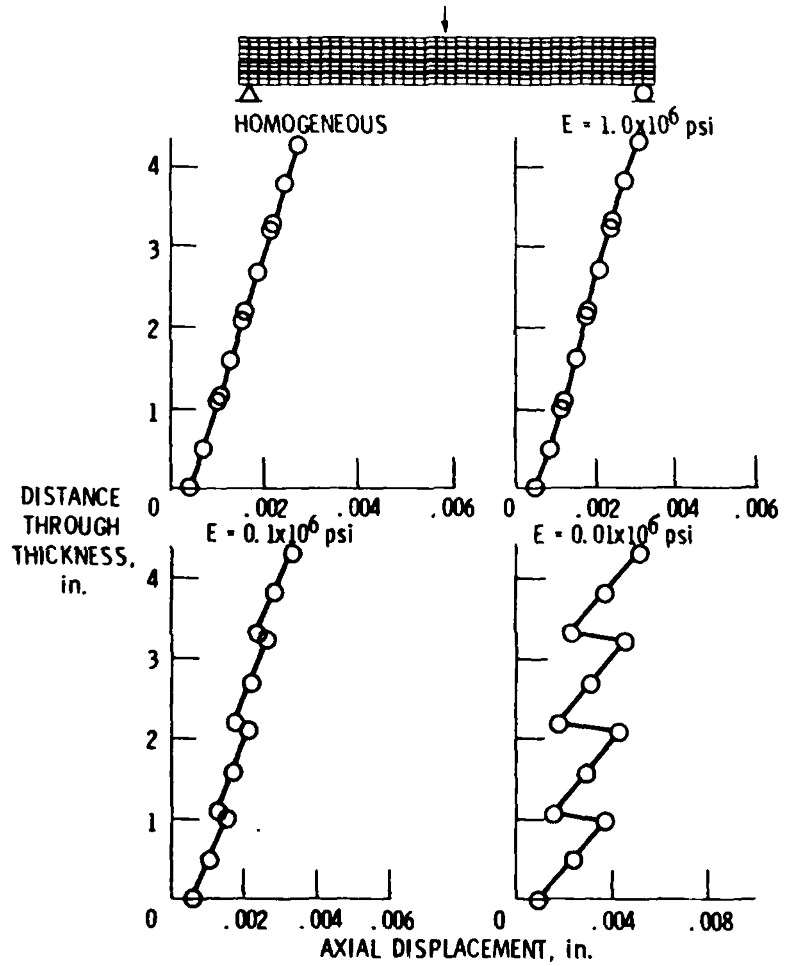
ELEMENTS:	440
NODES	492
ELEMENT ASPECT RATIO	PLY: 2/1 INTERPLY: 10/1
SUPPORT:	$x = y = 0$: $u, v = 0$ $x = \ell$, $y = 0$: $v = 0$
PLY:	$E_x = 20 \times 10^6$ psi; $E_y = 2 \times 10^6$ psi; $G = 0.8 \times 10^6$ psi
INTERPLY:	$1000 \text{ psi} \leq E \leq 1 \times 10^6$ psi

INTERPLY DEGRADATION EFFECTS ON MAXIMUM BENDING DISPLACEMENT



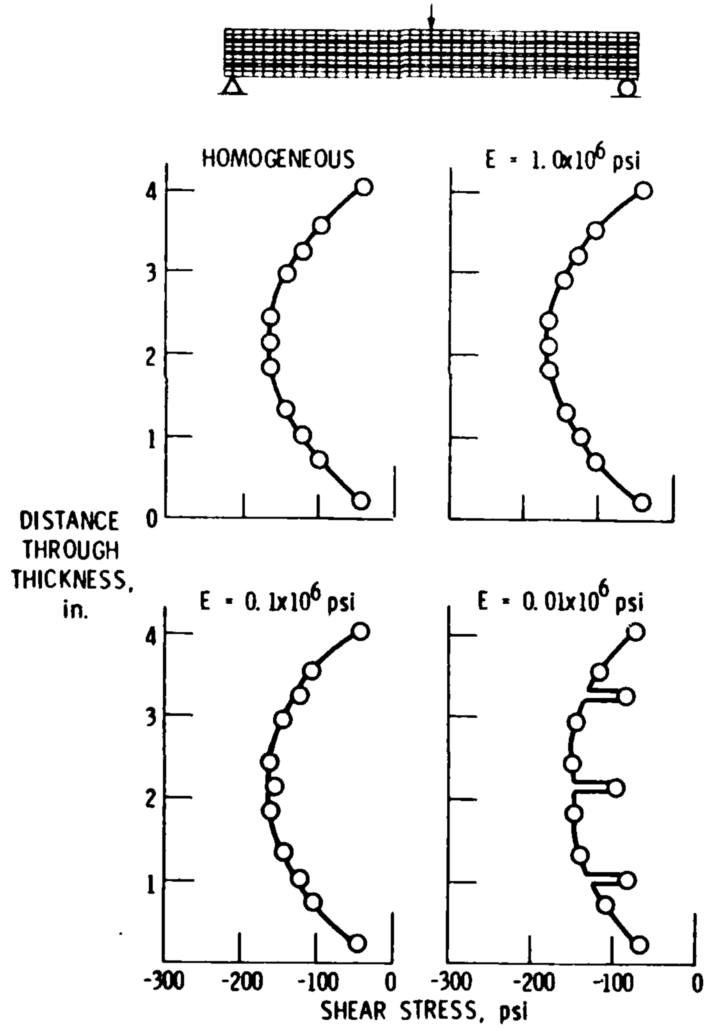
INTERPLY DEGRADATION EFFECTS ON AXIAL DISPLACEMENT

THROUGH A SECTION AT THE QUARTER SPAN

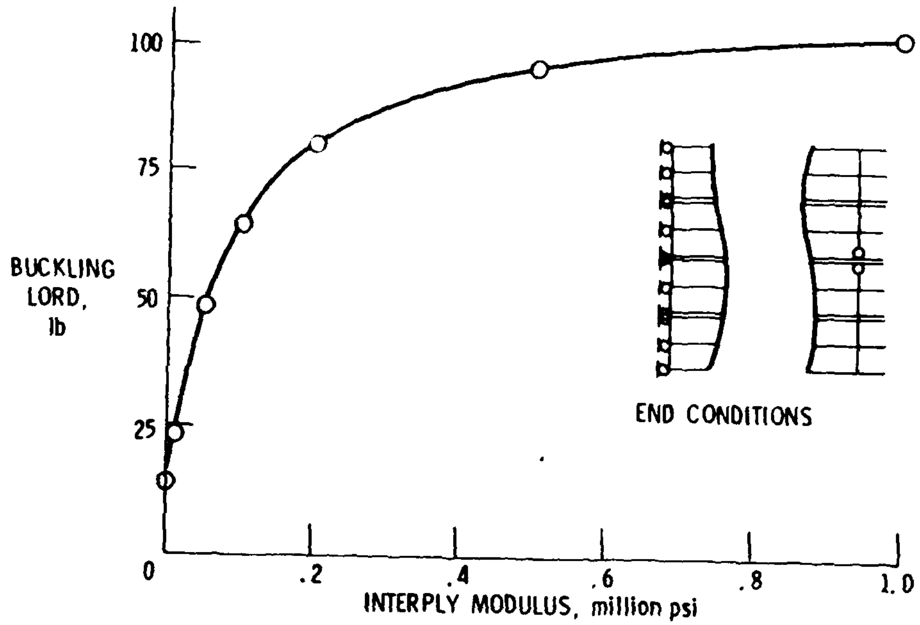


INTERPLY DEGRADATION EFFECTS ON SHEAR STRESS

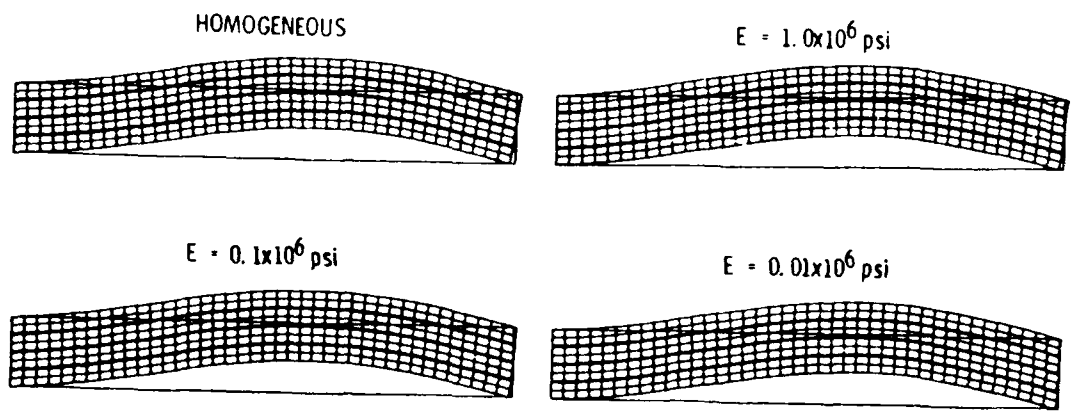
SECTION AT QUARTER SPAN



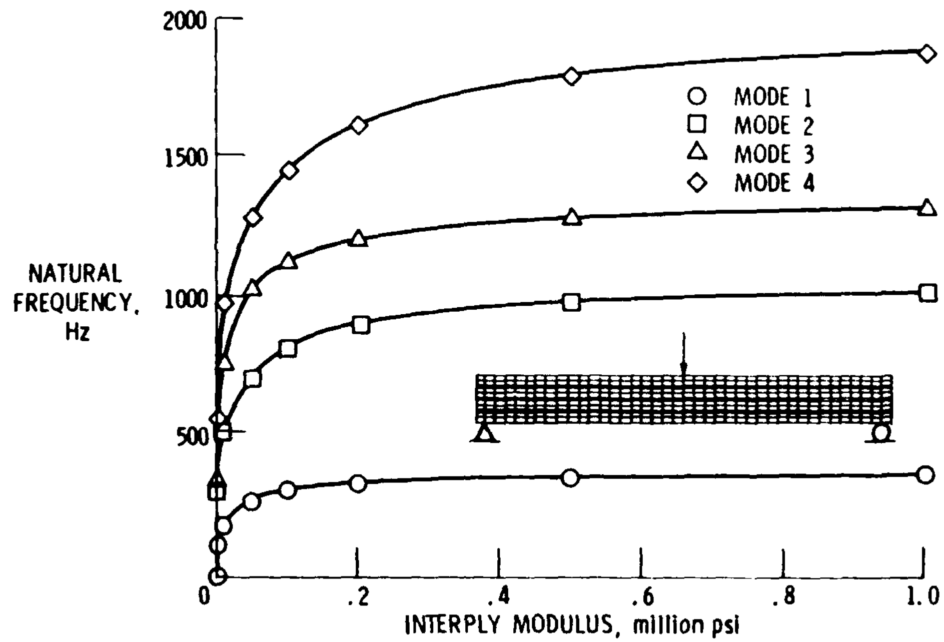
INTERPLY DEGRADATION EFFECTS ON BUCKLING LOAD



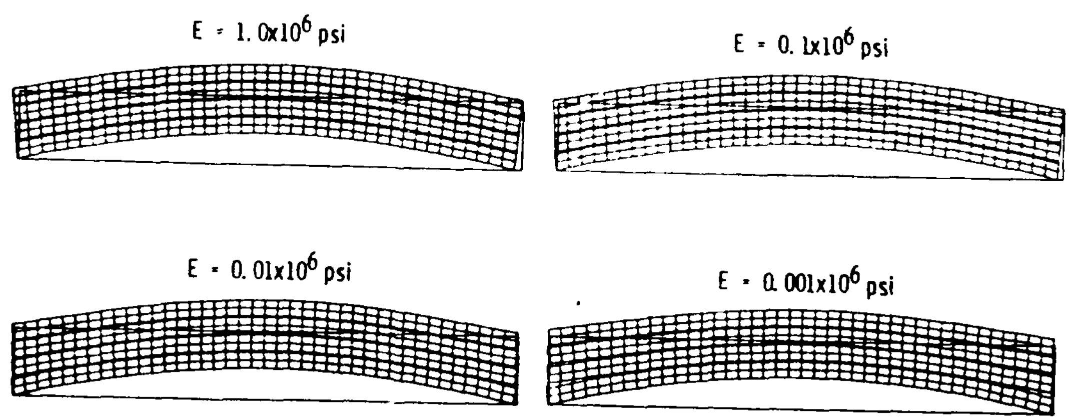
INTERPLY DEGRADATION EFFECTS ON BUCKLING MODES



INTERPLY DEGRADATION EFFECTS ON VIBRATION FREQUENCIES

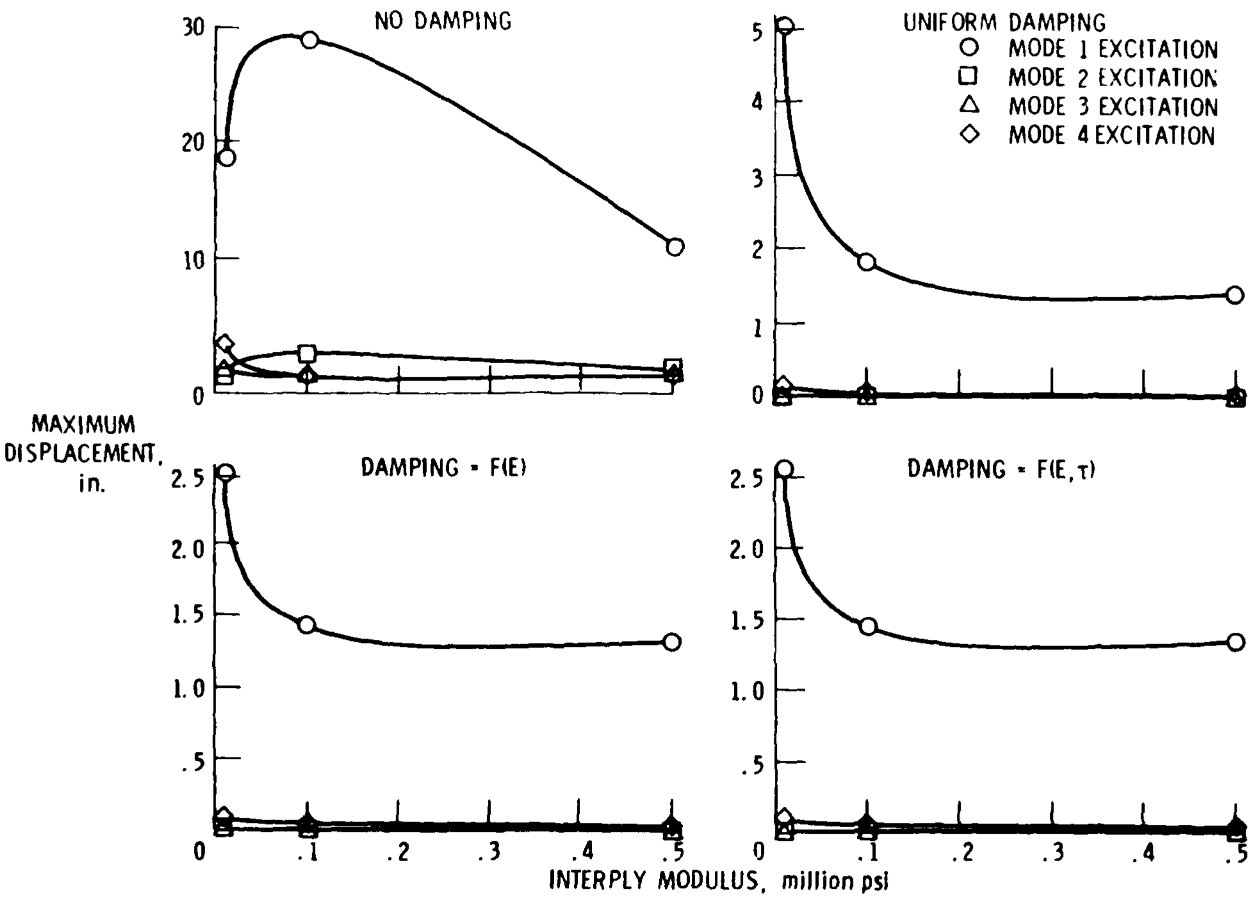


INTERPLY DEGRADATION EFFECTS ON THE 1st VIBRATION MODE

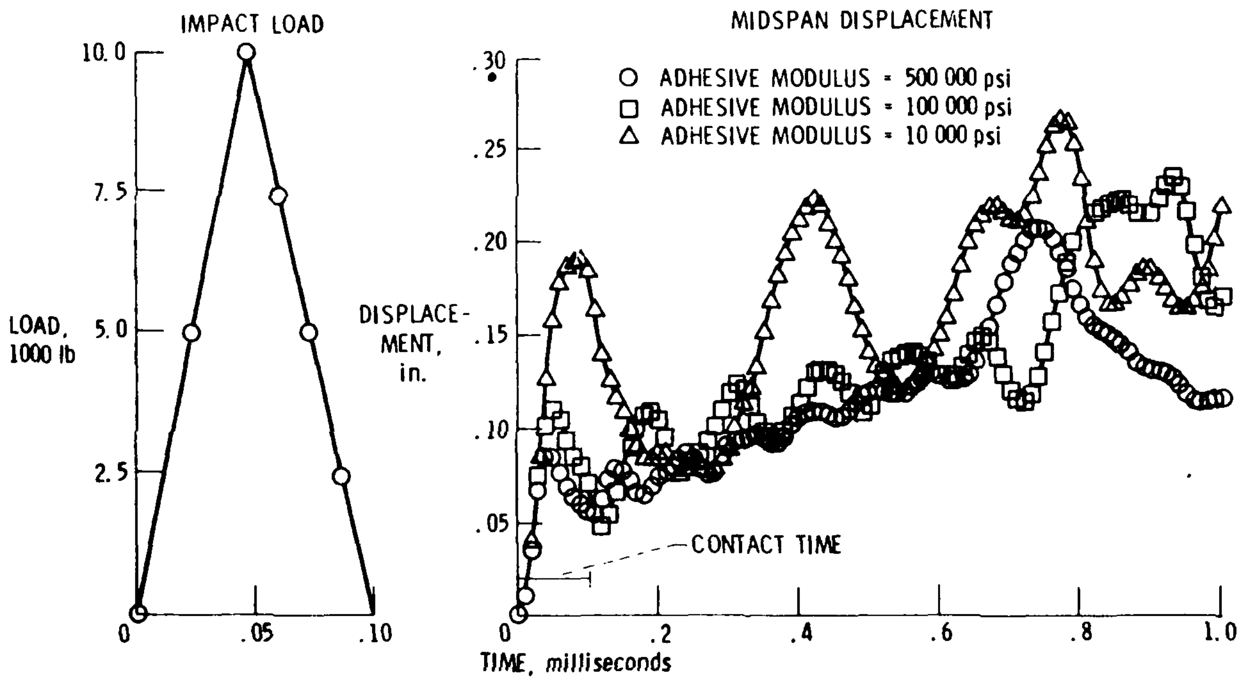


INTERPLY DEGRADATION EFFECTS ON FORCED VIBRATION RESPONSE

MAXIMUM DISPLACEMENT



INTERPLY DEGRADATION EFFECTS ON IMPACT RESPONSE



RESEARCH ON COMPOSITE MATERIALS FOR STRUCTURAL DESIGN

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Sponsored by the Air Force Office of Scientific Research under Contract No. F49620-82-C-0057

OBJECTIVES

GENERAL: STUDY DEFORMATION AND FRACTURE BEHAVIOR OF ADVANCED STRUCTURAL COMPOSITES WITH EMPHASIS ON POLYMER MATRIX DOMINATED PHENOMENA. MOST TOPICS WERE SELECTED SO THAT WORK COULD BE DONE AS PART OF A ONE-YEAR M.S. DEGREE PROGRAM.

M.S. THESES

STUDENT/ FACULTY ADVISOR TITLE	SPECIFIC OBJECTIVES
1. R.N. COHEN/ W.L. BRADLEY "EFFECT OF RESIN TOUGHNESS ON FRACTURE BEHAVIOR OF GRAPHITE/EPOXY COMPOSITES"	INVESTIGATE EXPERIMENTALLY TRANSVERSE AND DELAMINATION FRACTURE TOUGHNESS OF THREE GRAPHITE/EPOXY SYSTEMS. DETERMINE MICROMECHANISM OF ENERGY DISSIPATION USING IN-SITU FRACTURE IN SEM AND FRACTOGRAPHY ON RESULTING FRACTURED SURFACES.
2. H. RAZI/ R.A. SCHAPERY "SLOW, STABLE DELAMINATION IN GRAPHITE/EPOXY COMPOSITES"	USING UNIDIRECTIONAL SPLIT LAMINATES, DETERMINE RELATION BETWEEN CRACK SPEED AND ENERGY RELEASE RATE UNDER FIXED-GRIP CONDITIONS.
3. J.R. WEATHERBY/ R.A. SCHAPERY "EVALUATION OF ENERGY RELEASE RATES IN UNIDIRECTIONAL SPLIT LAMINATE SPECIMENS"	DEVELOP IMPROVED BEAM THEORY WHICH ACCOUNTS FOR ROTATION AT THE CRACK TIP.

4. R.T. ARENBURG/ R.A. SCHAPERY "ANALYSIS OF THE EFFECT OF MATRIX DEGRADATION ON FATIGUE BEHAVIOR OF A GRAPHITE/EPOXY LAMINATE"

PREDICT MECHANICAL STATE AND ENERGY RELEASE RATE FOR DELAMINATION IN TENSILE COUPONS WITH VARIOUS AMOUNTS OF DELAMINATION AND MATRIX DEGRADATION; COMPARE RESULTS WITH FATIGUE-INDUCED DEFORMATION AND FRACTURE OF $[\pm 45/90_2]_S$ GRAPHITE/EPOXY LAMINATES.

PH.D. THESIS

5. B.D. HARPER/ Y. WEITSMAN "ON THE EFFECTS OF POST CURE COOL DOWN AND ENVIRONMENTAL CONDITIONING ON RESIDUAL STRESSES IN COMPOSITE LAMINATES"

STUDY THE APPLICABILITY OF LINEAR VISCOELASTICITY IN PREDICTING CURVATURES OF ANTI-SYMMETRIC CROSS-PLY GRAPHITE/EPOXY LAMINATES AFTER BEING COOLED FROM THEIR CURE TEMPERATURE AND DURING EXPOSURE TO BOTH CONSTANT AND FLUCTUATING RELATIVE HUMIDITIES. BOTH ELASTIC AND VISCOELASTIC ANALYTICAL PREDICTIONS ARE TO BE COMPARED WITH MEASURED CURVATURES OF AS4/3502 GRAPHITE/EPOXY LAMINATES, WITH EMPHASIS ON EFFECTS OF MOISTURE FLUCTUATIONS.

NOTE:
THESIS NOS. 1-3, COMPLETION EXPECTED BY DEC. 1982
THESIS NO. 4, COMPLETED MAY 1982
THESIS NO. 5, COMPLETION EXPECTED IN 1983

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CONCLUSIONS

(NUMBERS CORRESPOND TO THOSE FOR OBJECTIVES)

1. FRACTURE TOUGHNESS OF LAMINATES (24 PLIES):
   (I) THE HIGH TOUGHNESS RESIN SYSTEM HAS A CRITICAL ENERGY RELEASE RATE WHICH DECREASES WITH INCREASING PROPORTION OF MODE II COMPONENT. T6T145/F185 DECREASES FROM A G OF 2400 J/m^2 TO 1900 J/m^2 AS $G_{II}/(G_I + G_{II})$ INCREASES FROM 0.0 TO 0.40.
   (II) FOR LOWER TOUGHNESS SYSTEMS THE OPPOSITE BEHAVIOR APPEARS TO EXIST. AS4/3502 INCREASES FROM 200 TO 475 J/m^2 WHILE T6C190/F155 INCREASES FROM 550 TO 750 J/m^2 FOR THE SAME RELATIVE INCREASE OF G_{II}.
   (III) MODE RATIO HISTORY IN THE BRITTLE SYSTEMS HAS A PRONOUNCED EFFECT ON FRACTURE TOUGHNESS THAT IS NOT SEEN IN TOUGHER SYSTEM.
   (IV) FOR BOTH TYPES OF SYSTEMS, VALUES OF MODE I CRITICAL ENERGY RELEASE RATES FOR TRANSVERSE CRACKING AND DELAMINATION FRACTURE ARE SIMILAR.
   (V) THE DAMAGE ZONE AT THE CRACK TIP IN THE BRITTLE MATERIALS IS LIMITED TO THE RESIN RICH REGION BETWEEN PLYS WHEREAS IN THE TOUGH F185 RESIN SYSTEM IT EXTENDS THREE TO FIVE FIBER DIAMETERS FROM THE PRIMARY FRACTURE PLANE.
   (VI) WEAK INTERFACIAL BONDING AND SOME MATRIX DUCTILITY CONTRIBUTE TO SIGNIFICANT TIE ZONE FORMATION WHICH APPARENTLY INCREASES THE OBSERVED CRITICAL ENERGY RELEASE RATE BY SEVERAL HUNDRED J/m^2.

2. SLOW, STABLE DELAMINATION (MODE I, FIXED GRIPS, 8 PLIES):
   (I) THE ENERGY RELEASE RATE DURING SLOW GROWTH IS APPROXIMATELY 15% LESS THAN THE INITIATION VALUE G_c FOR THE BRITTLE SYSTEM (3502) AND 30% LESS THAN G_c FOR THE TOUGH SYSTEM (F185).
   (II) FOR THE BRITTLE SYSTEM, $G \simeq 204\dot{A}^{.0045}$ (DRY, RT) AND $G \simeq 195\dot{A}^{.0046}$ (WET, RT) WHERE G IS J/m^2 AND $\dot{A}$ IS CM/S.
   (III) FOR THE TOUGH SYSTEM, $G \simeq 1665\dot{A}^{.022}$.

3. DELAMINATION ENERGY RELEASE RATE ANALYSIS:
   (I) DEFORMATION OF THE LAMINATE AHEAD OF THE CRACK AFFECTS STIFFNESS AND ENERGY RELEASE RATE AS A FUNCTION OF DELAMINATION LENGTH.
   (II) FOR MODE I, A MODEL CONSISTING OF A BEAM SUPPORTED AT THE CRACK TIP BY A TORSION SPRING (WITH A LENGTH-INDEPENDENT SPRING CONSTANT) CAN BE USED TO PREDICT STIFFNESS AND ENERGY RELEASE RATE AS A FUNCTION OF LENGTH.
   (III) AN APPROXIMATE ANALYTICAL PREDICTION OF THE MODE I SPRING CONSTANT AGREES WELL WITH FINITE ELEMENT SOLUTIONS.
   (IV) IN A LIMITED ANALYTICAL STUDY OF MODE II DELAMINATION, THE CORRECTION FOR BEAM DEFORMATION NEAR THE CRACK TIP IS ESTIMATED TO BE 1/3 THAT FOR MODE I.

4. EFFECT OF MATRIX DEGRADATION:
   (I) FOR $[\pm 45/90_2]_s$ TENSILE COUPONS WITH A 20:1 WIDTH/THICKNESS RATIO, AXIAL STIFFNESS AND DELAMINATION ENERGY RELEASE RATE PREDICTIONS USING CLASSICAL LAMINATION THEORY ARE IN CONSIDERABLE ERROR WITH FATIGUE-INDUCED MATRIX DAMAGE.
   (II) WITH LOAD INPUT FIXED, THE DELAMINATION ENERGY RELEASE RATE INCREASES WITH MATRIX DAMAGE; BUT IT IS NOT HIGH ENOUGH TO CAUSE DELAMINATION UNTIL THERE IS CONSIDERABLE MATRIX DEGRADATION.

5. RESIDUAL STRESSES IN LAMINATES:
   (I) STUDIES TO DATE INDICATE VISCOELASTIC EFFECTS ARE OF SECONDARY IMPORTANCE DURING COOL-DOWN, BUT MAY BE SIGNIFICANT IN THE RESPONSE TO MOISTURE. MEASURED CURVATURES OF NON-SYMMETRIC LAMINATES DEVIATE NOTICEABLY FROM ELASTIC PREDICTIONS.
   (II) A CORRECT TIME-TEMPERATURE-MOISTURE ANALOGY SHOULD INCLUDE VERTICAL AS WELL AS HORIZONTAL SHIFT-FACTORS TO MAKE LONG-TIME PREDICTIONS FROM SHORT-TERM DATA.

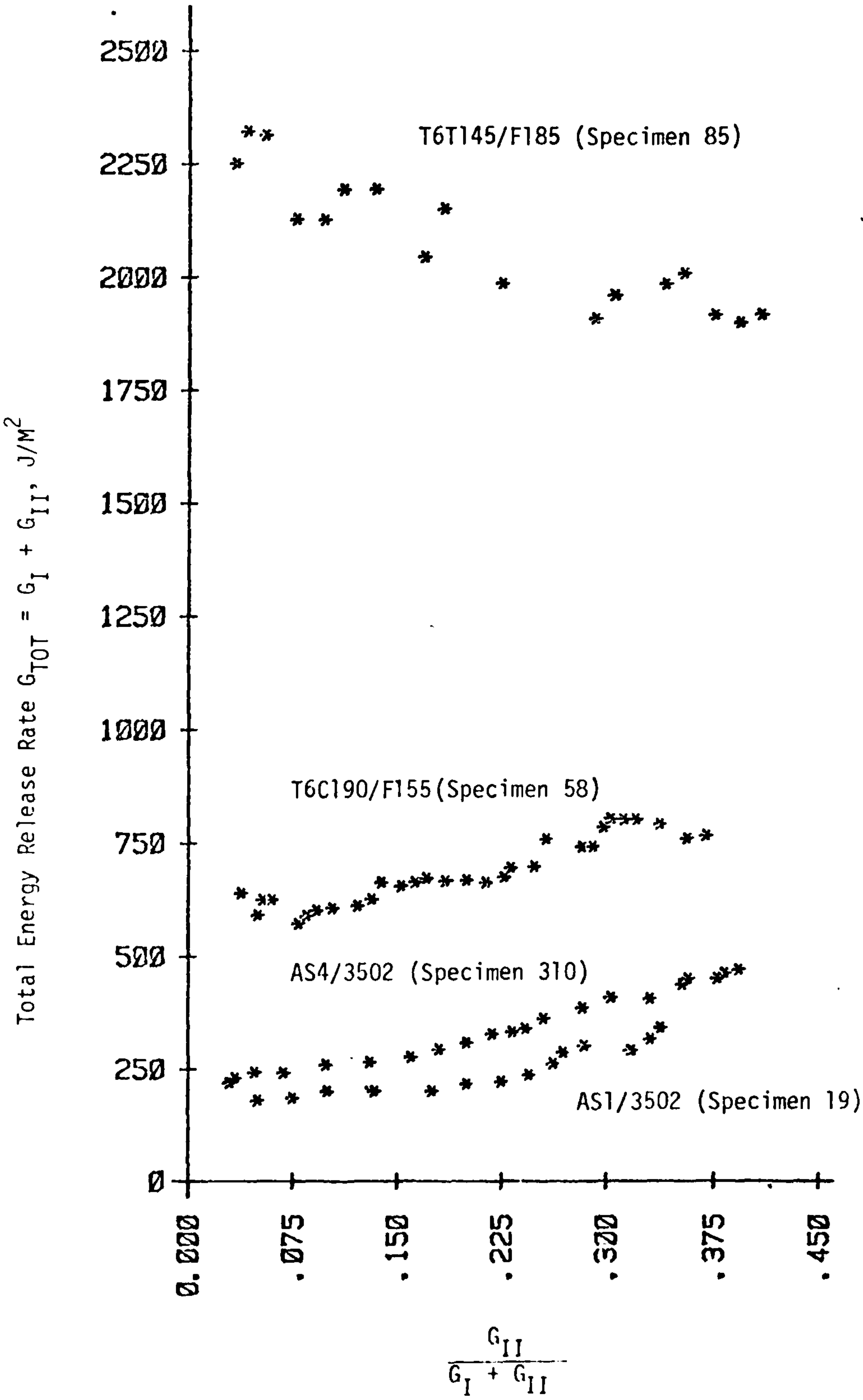
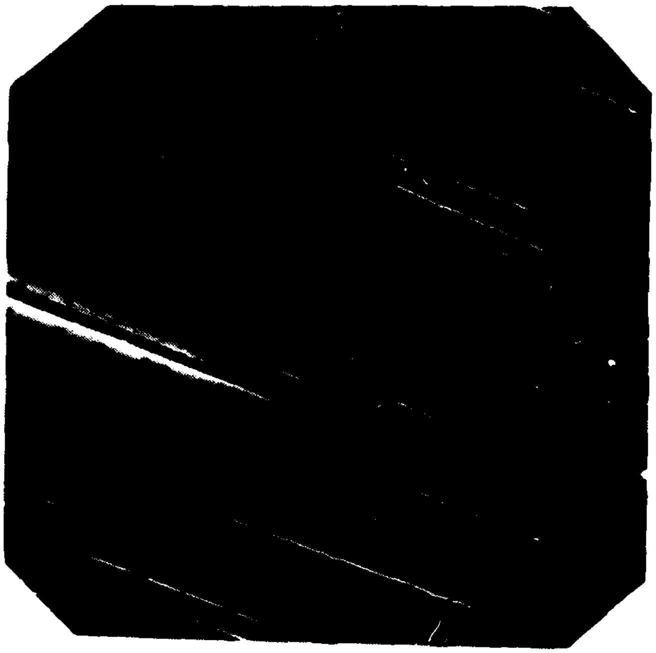
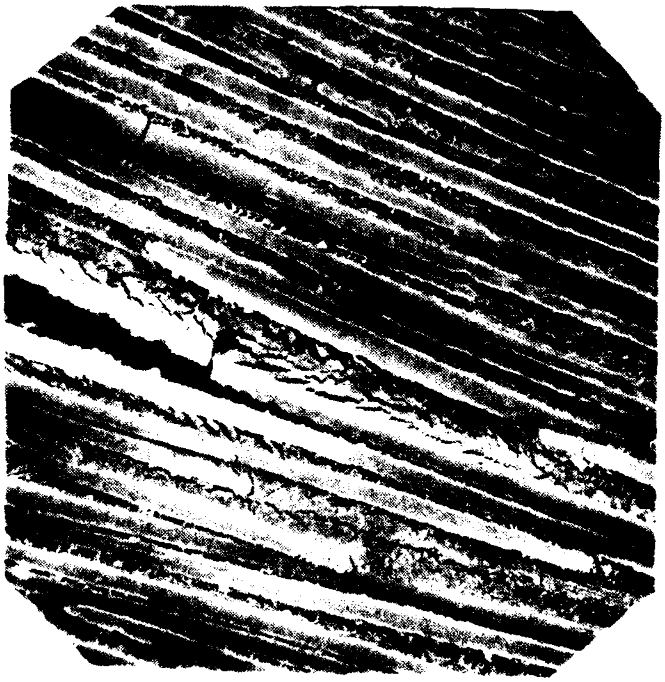


Fig. 1 Total critical energy release rate for mixed mode delamination for increasing fraction of Mode II.



(a)
AS4/3502
500X



(b)
T6T145/F185
500X

Fig. 2 Delamination crack tip under load
a) brittle system, b) tough system

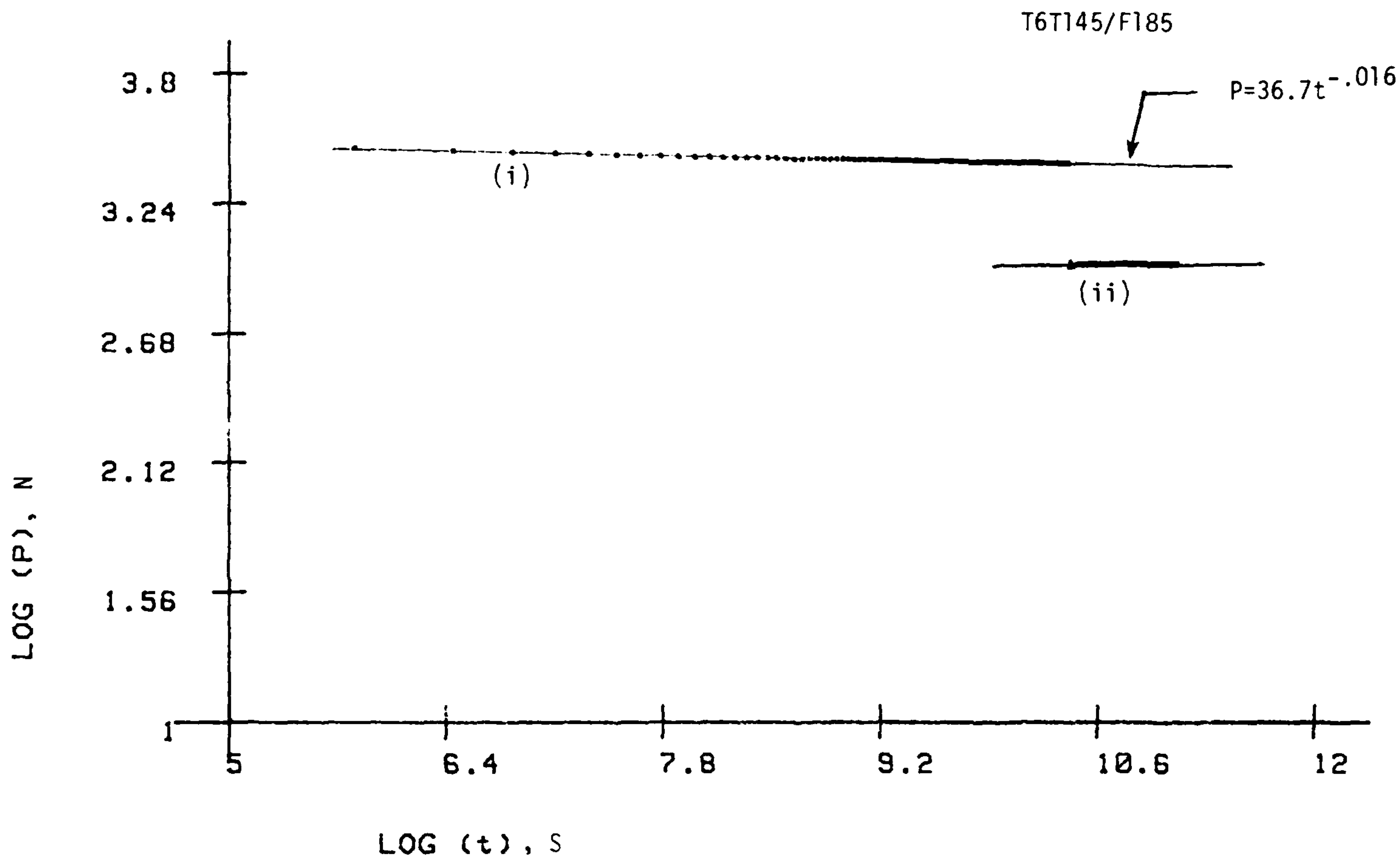
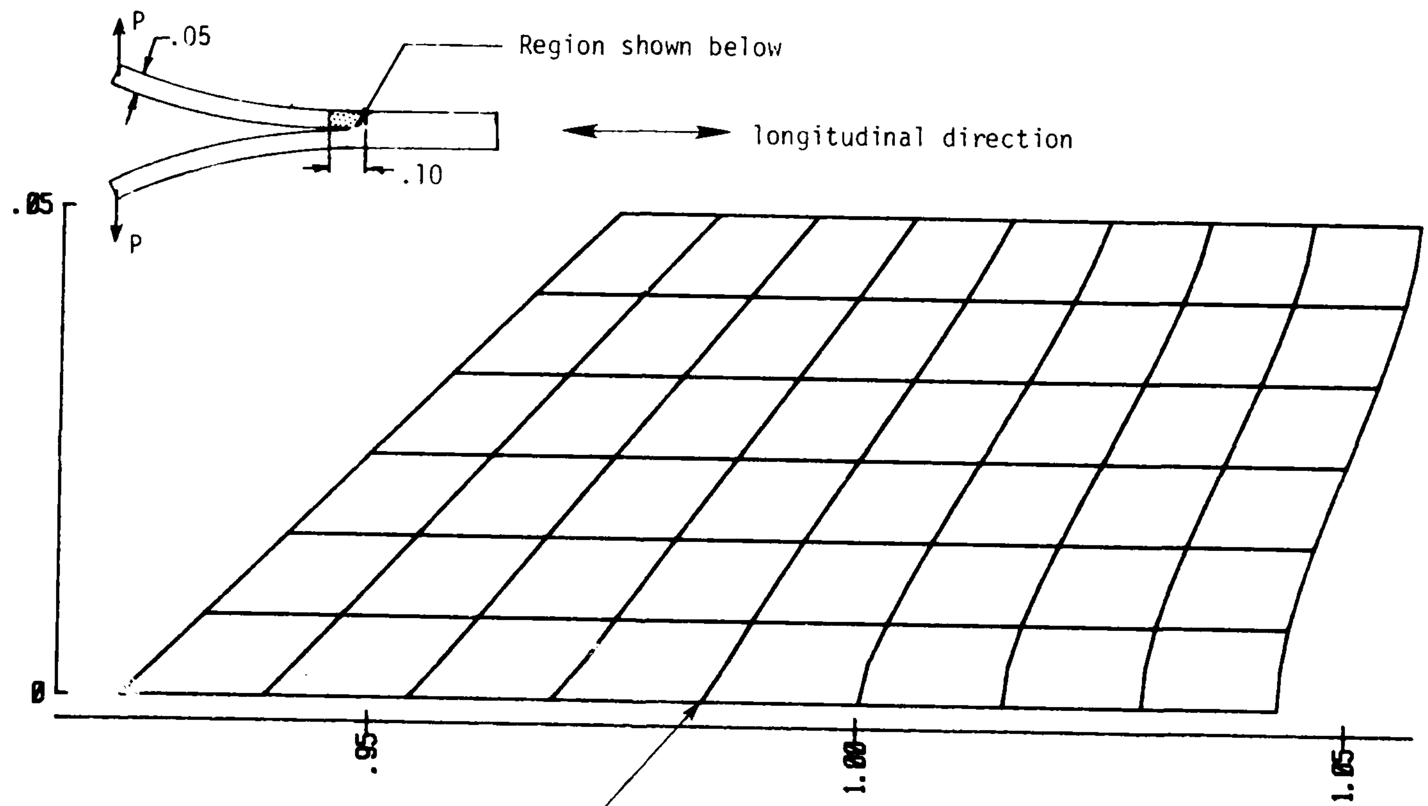


Fig. 3 Load Versus Time With Fixed Grips for Split Laminate During (i) Slow Crack Growth (ii) No Growth.



Method used: Finite element

Fig. 4 Longitudinal displacements near the crack tip in a graphite/epoxy split-laminate specimen. (0° Fiber Angle)

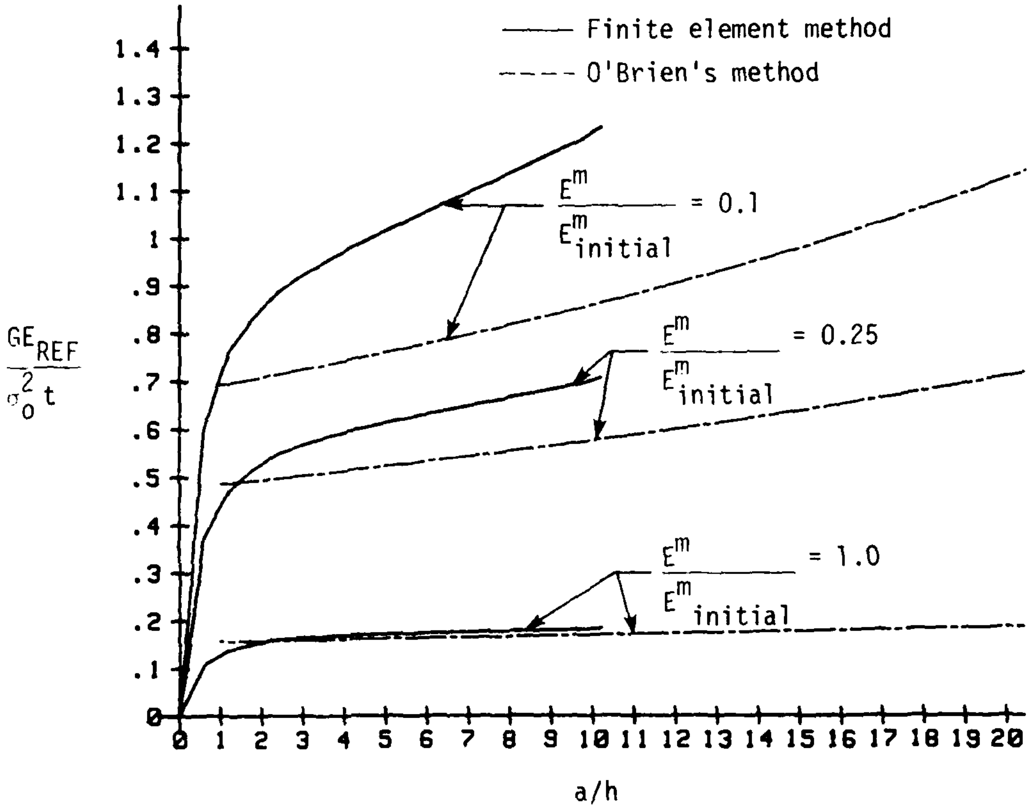


Fig. 5 Total normalized strain energy release rate as a function of delamination length/ply thickness for a constant stress condition. $[\pm 45/90_2]_s$ tensile coupon under axial stress σ_o. Total thickness is t and matrix modulus is E^m.

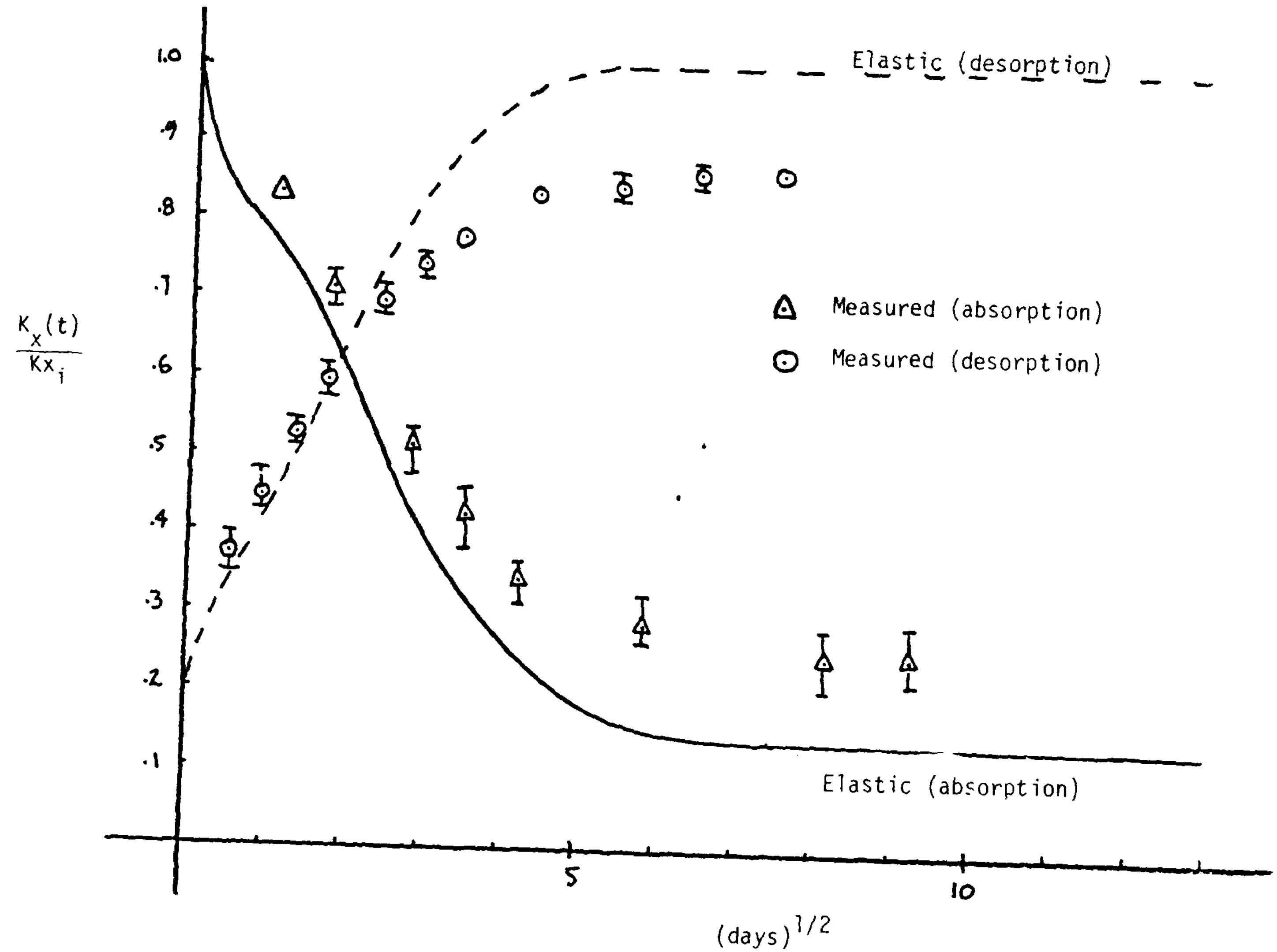


Fig. 6 Fraction of Dry curvature at Room Temperature Conditioned at 163°F, 95% RH. $[0/90/0_4/90_4/0/90]$, AS4/3502 Laminate

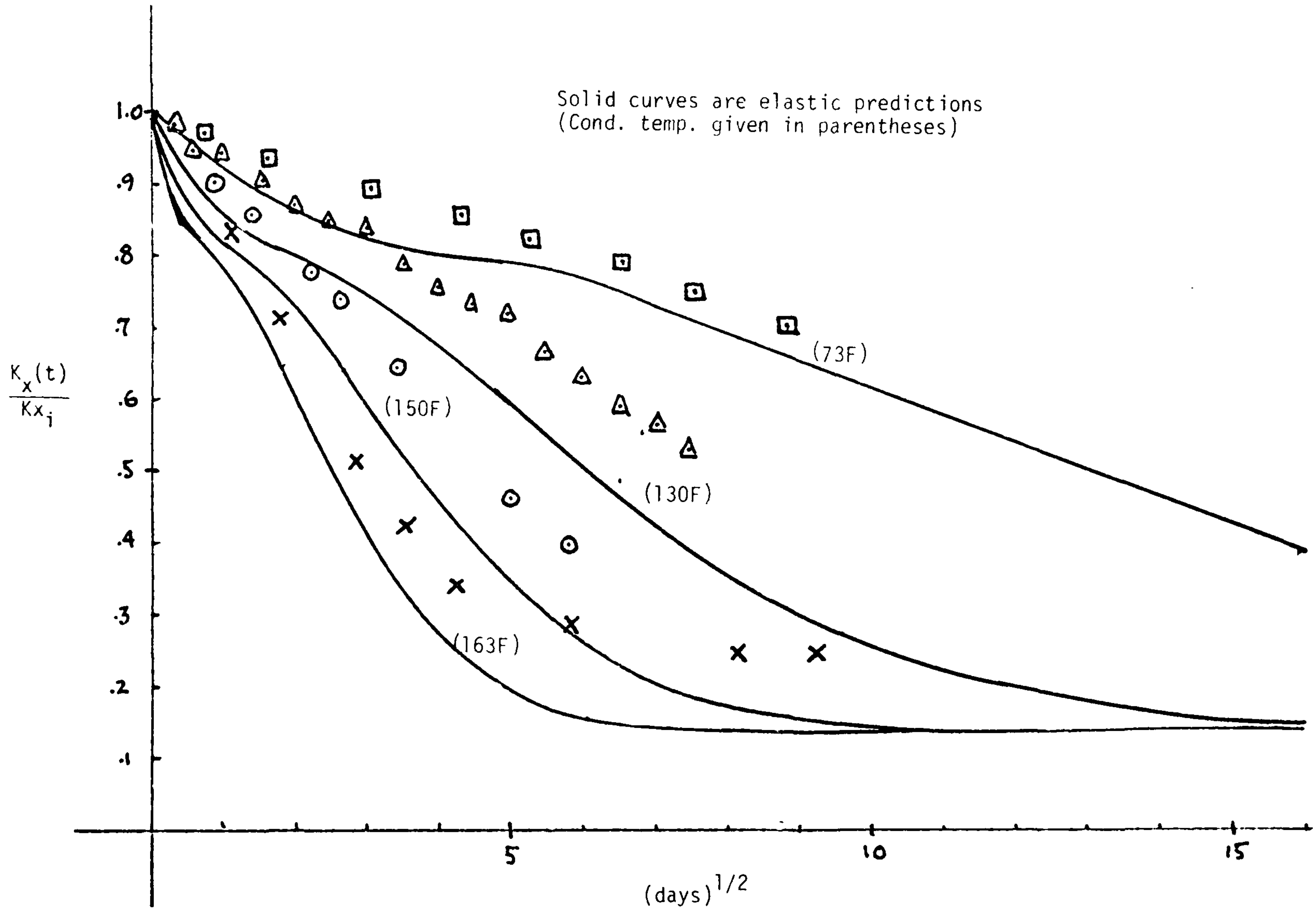


Fig. 7 Fraction of Dry Curvature at Room Temperature Conditioned at 95% RH and various temperatures. Same laminate as in Fig. 6.

AD-A130 750 PROCEEDINGS OF THE ANNUAL MECHANICS OF COMPOSITES REVIEW (8TH) HELD AT WR..(U) AIR FORCE WRIGHT AERONAUTICAL LABS WRIGHT-PATTERSON AFB OH L A WILSON 1/5

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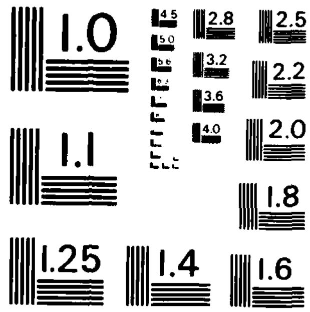

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

IMPERFECTION SENSITIVITY OF FIBER-REINFORCED COMPOSITE THIN CYLINDERS+

BY

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ATLANTA, GEORGIA

+ WORK SPONSORED BY AFOSR,
DIRECTORATE OF AEROSPACE SCIENCES

THE PRESENT WORK IS PART OF ONGOING ANALYTICAL RESEARCH DEALING WITH :

1. NONLINEAR RESPONSE

   (PRE- AND POST-LIMIT POINT BEHAVIOR)

2. OF GEOMETRICALLY IMPERFECT, CIRCULAR, CYLINDRICAL, THIN SHELLS OF VARIOUS CONSTRUCTIONS (METALLIC, WITH OR WITHOUT STIFFENERS, LAMINATED, ETC.)

3. SUPPORTED IN VARIOUS WAYS AT THE BOUNDARIES (CLAMPED, PINNED, ETC.)

4. AND SUBJECTED TO STATIC AND SUDDEN (STATIC AS WELL AS DYNAMIC) LOADS (PRESSURE, AXIAL COMPRESSION, TORSION) APPLIED INDIVIDUALLY OR IN COMBINATION.

THE GOVERNING EQUATIONS ARE BASED ON

1. NONLINEAR KINEMATIC RELATIONS (VON KARMAN-DONNELL TYPE)
2. SMEARED TECHNIQUE FOR THE ORTHOGONAL STIFFENERS
3. LINEARLY ELASTIC MATERIAL BEHAVIOR

AND DERIVED FROM THE PRINCIPLE OF THE STATIONARY VALUE OF THE TOTAL POTENTIAL (BY EMPLOYING THE USUAL LAMINATION THEORY - EACH LAMINA IS ORTHOTROPIC).

THE FIELD EQUATIONS CONSIST OF

(I) TRANSVERSE EQUILIBRIUM EQUATION

(II) IN-PLANE COMPATIBILITY EQUATIONS IN TERMS OF

$W(X,Y)$: TRANSVERSE DISPLACEMENT

$F(X,Y)$: AIRY STRESS (RESULTANT) FUNCTION DEFINED BY

$$N_{XX} = -\bar{\bar{N}}_{XX} + F_{,YY};\ N_{YY} = F_{,XX};\ N_{XY} = \bar{\bar{N}}_{XY} - F_{,XY}$$

BOUNDARY CONDITIONS $(x = 0.L)$

SS-i : $w = 0;\ M_{XX} = -\bar{\bar{E}}\bar{\bar{N}}_{XX}$

CC-i : $w = w_{,X} = 0$

FF-i : $Q^{*}_{X} = 0;\ M_{XX} = -\bar{\bar{E}}\bar{\bar{N}}_{XX}$

$i = 1,2,3,4$

1. $F_{,XY} = F_{,YY} = 0$
2. $F_{,XY} = 0;\ u = \text{CONST.}$
3. $F_{,YY} = 0;\ v = \text{CONST.}$
4. $u = \text{CONST.};\ v = \text{CONST.}$

STRUCTURAL GEOMETRY OF EXAMPLES

I. A FOUR-PLY BORON/EPOXY (AVCO 5505) LAMINATED CIRCULAR CYLINDRICAL SHELL IS USED.

$R = 7.5$ IN. ; $L = 15.0$ IN.

WITH ORTHOTROPIC PROPERTIES FOR EACH LAMINA

$E_{11} = 30 \times 10^6$ PSI; $E_{22} = 2.7 \times 10^6$ PSI

$\nu_{12} = 0.21$; $G_{12} = 0.65 \times 10^6$ PSI; $h_{ply} = 0.0053$ IN.

I-1: $45^\circ/-45^\circ/-45^\circ/45^\circ$ (OUTER TO INNER)

I-2: $45^\circ/-45^\circ/45^\circ/-45^\circ$; I-3: $-45^\circ/45^\circ/-45^\circ/45^\circ$;

I-4: $90^\circ/60^\circ/30^\circ/0^\circ$; I-5: $0^\circ/30^\circ/60^\circ/90^\circ$

II. METALLIC (ALUMINUM) ISOTROPIC CIRCULAR CYLINDRICAL SHELL

$E - 10.5 \times 10^6$ PSI ; $\nu = 0.3$; $R = L = 4$ IN. ; $H = 0.004$ IN.

III. ORTHOTROPIC (SINGLE LAYER BORON/EPOXY)

$E_{XX} = 30.5 \times 10^6$ PSI ; $E_{YY} = 2.7 \times 10^6$ PSI ; $\nu_{XY} = 0.21$;

$G_{XY} = 0.65 \times 10^6$ PSI ; $R = 7.5$ IN. ; $L = 15.0$ IN. ; $h = 0.0212$ IN.

INITIAL GEOMETRIC IMPERFECTIONS

(A) AXISYMMETRIC (ALMOST)

$$w^{0}(x,y) = \xi h\left(-\cos\frac{2\pi x}{L} + 0.1\sin\frac{\pi x}{L}\cos\frac{nY}{R}\right)$$

NOTE: ξ : IMPERFECTION AMPLITUDE PARAMETER AND

$$w^{0}_{MAX}/h = 1.1\,\xi$$

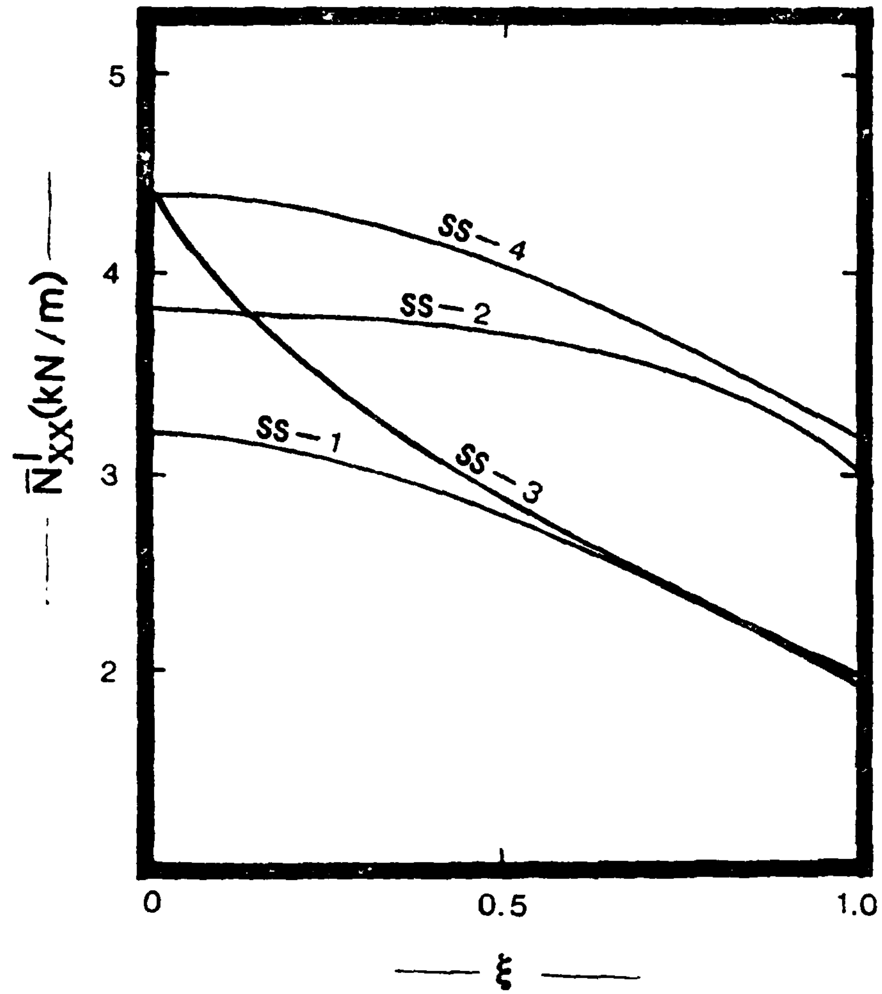
(B) SYMMETRIC

$$w^{0}(x,y) = \xi h \sin\frac{\pi x}{L}\cos\frac{nY}{R}$$

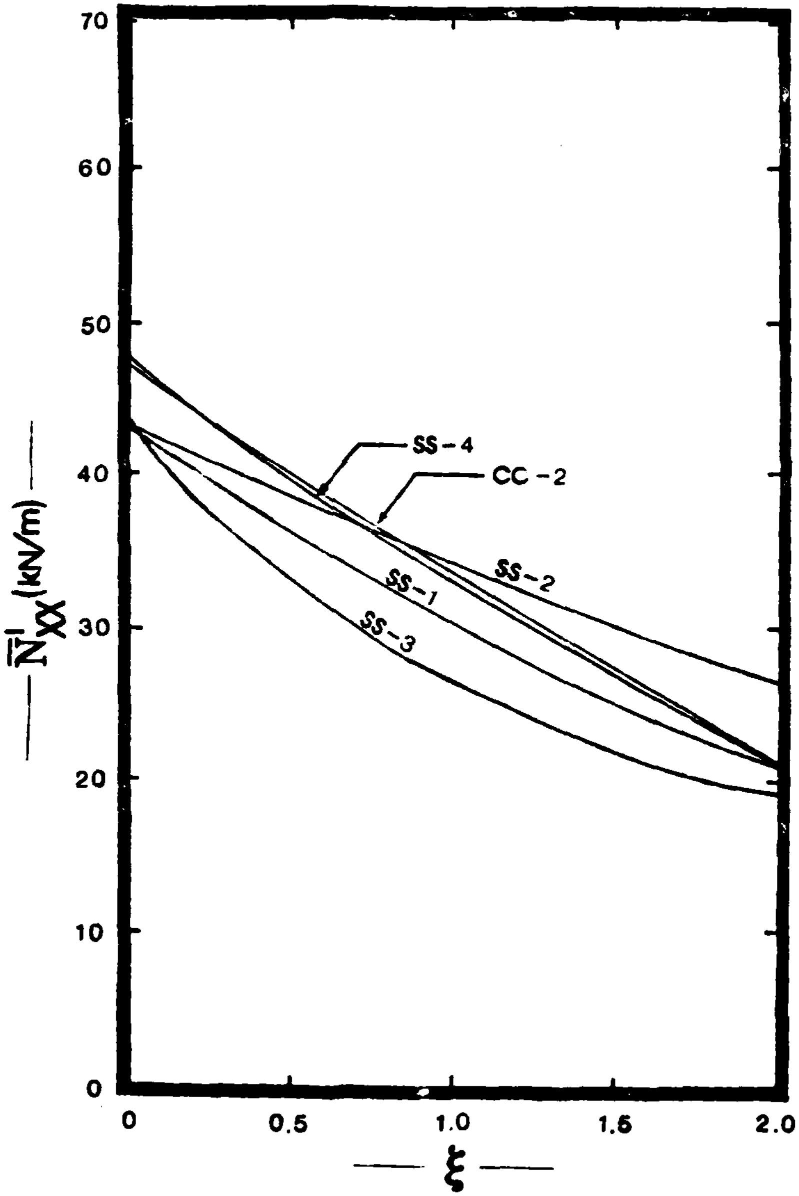
FOR THIS CASE $\quad w^{0}_{MAX.}/h = \xi$

<u>RESULTS</u> ARE GENERATED FOR

1. TESTING THE SOLUTION SCHEME (BENCH MARKS)
2. STUDYING THE IMPERFECTION SENSITIVITY OF CERTAIN GEOMETRIES.
3. ASSESSING THE EFFECT OF BOUNDARY CONDITIONS (BOTH TRANSVERSE AND IN-PLANE, SS-i AND CC-i), AND
4. ASSESSING THE EFFECT OF APPLYING THE UNIFORM COMPRESSION ECCENTRICALLY.



ISOTROPIC GEOMETRY
AXISYM. IMPERFECTION



LAMINATED GEOMETRY I-5

GENERAL CONCLUSIONS

1. ISOTROPIC GEOMETRY AND SS-BOUNDARIES

   (A) FOR SMALL ξ-VALUES (INCLUDING $\xi = 0$)

   SS-3 AND SS-4 (v = CONST.) CONDITIONS YIELD A STRONGER CONFIGURATION THAN SS-1 AND SS-2 ($N_{XY} = - F_{,XY} = 0$)

   (B) FOR LARGER ξ-VALUES ($\xi \gtrsim 0.2$)

   SS-2 AND SS-4 (u = CONST.) CONDITIONS YIELD A STRONGER CONFIGURATION THAN SS-1 AND SS-3 ($N_{XX} = - \overline{N}_{XX}$)

2. LAMINATED GEOMETRY. FOR ALL -VALUES CONSIDERED SS-2 AND SS-4 (u = CONST.) CONDITIONS YIELD A STRONGER CONFIGURATION THAN SS-1 AND SS-3. THIS IS ALSO TRUE FOR CLAMPED BOUNDARIES.

3. FOR ALL GEOMETRIES STUDIED, NO GENERAL CONCLUSION CAN BE DRAWN WITH REGARD TO THE STABILIZING OR DESTABILIZING EFFECT OF LOAD ECCENTRICITY. IT APPEARS, THOUGH THAT

   (A) THE EFFECT DIMINISHES WITH INCREASING ξ -VALUE, AND

   (B) THE EFFECT IS DEPENDENT ON THE VALUE OF A_{12}.

RESEARCH INTO THE DESIGN TECHNOLOGY OF ADVANCED COMPOSITES

A) DAMAGE TOLERANCE

B) AEROELASTICITY

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OBJECTIVES:

A) DAMAGE TOLERANCE: —

- TO EXAMINE THE EFFECT OF FLAW SIZE AND STACKING SEQUENCE ON FRACTURE STRESS (PRESSURE) AND MODE FOR GRAPHITE/EPOXY CYLINDERS AND TO PREDICT THESE FRACTURE STRESSES USING COUPON (FLAT PLATE) DATA
- TO EXPLORE POSSIBLE MEANS OF SELECTIVELY REINFORCING COMPOSITE CYLINDERS IN ORDER TO ARREST CRACK PROPAGATION

B) AEROELASTICITY: —

- TO STUDY THE EFFECT OF BENDING-TORSION STIFFNESS COUPLING IN ADVANCED COMPOSITES, ON THE AEROELASTIC FLUTTER AND DIVERGENCE BEHAVIOR OF UNSWEPT AND FORWARD SWEPT WINGS
- TO EXPLORE AEROLEASTIC BEHAVIOR OF WINGS AT HIGH AS WELL AS LOW ANGLES OF ATTACK

CONCLUSIONS TO DATE

A) DAMAGE TOLERANCE:

- FRACTURE DATA OBTAINED FROM COUPON TESTS CAN BE USED TO PREDICT THE FAILURE OF GRAPHITE/EPOXY CYLINDERS BY CONSIDERING THE LOCAL STRESS INTENSIFICATION NEAR THE FLAW DUE TO THE EFFECTS OF CURVATURE.

- STACKING SEQUENCE HAS AN IMPORTANT EFFECT ON THE FAILURE MODE OF GRAPHITE/EPOXY CYLINDERS WHILE FLAW SIZE DOES NOT.

- SELECTIVE REINFORCEMENTS OF THESE CYLINDERS DID NOT PREVENT CATASTROPHIC FAILURE BUT CAN CHANGE THE PATH OF DAMAGE PROPAGATION.

- THE PATH OF DAMAGE PROPAGATION IN THESE SELECTIVELY REINFORCED CYLINDERS IS DEPENDENT UPON THE STACKING SEQUENCE OF THE REINFORCEMENT PLIES.

B) AEROELASTICITY:

- WIND TUNNEL TESTS SHOWED DIVERGENCE AND B-T FLUTTER AT LOW ANGLES OF ATTACK; T-STALL FLUTTER AND B-STALL FLUTTER AT HIGH ANGLES OF ATTACK.

- BENDING-TORSION STIFFNESS COUPLING CAN BE BENEFICIAL IN DELAYING OR ELIMINATING DIVERGENCE AND FLUTTER IN FORWARD SWEPT WINGS.

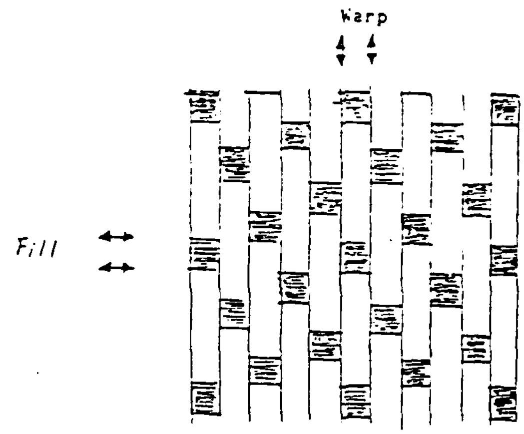
- THERE WAS REASONABLE AGREEMENT BETWEEN TEST AND THEORY AT LOW ANGLES OF ATTACK FOR DIVERGENCE AND FLUTTER.

CHARACTERISTICS OF TEST SPECIMENS

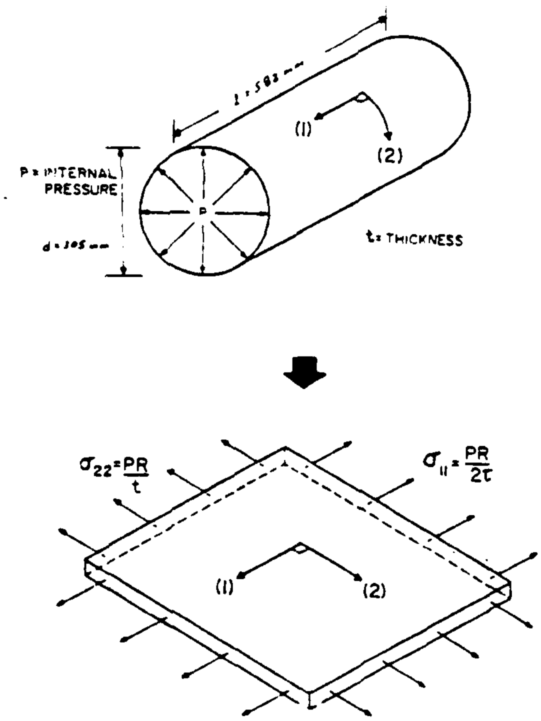
Hercules A370-5H/3501-6 Graphite/Epoxy Cloth

NOMINAL PROPERTIES:

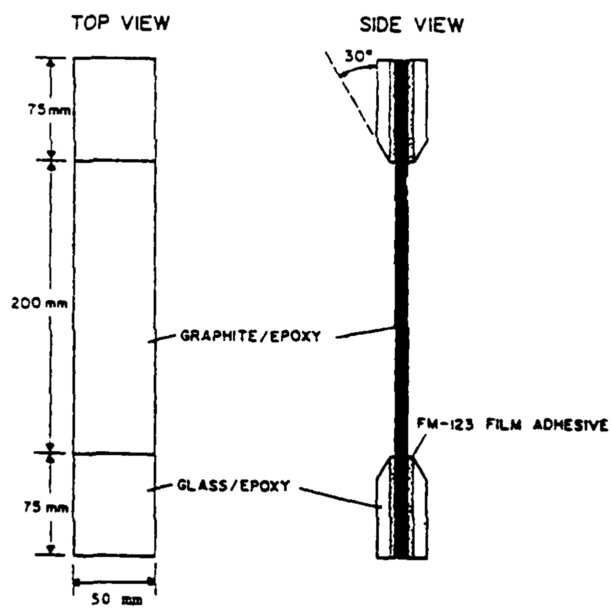
CURED PER PLY THICKNESS	0.35	MM
WARP MODULUS, E_L	72	GPA
FILL MODULUS, E_T	72	GPA
WARP POISSON RATIO, ν_{LT}	0.06	
FILL POISSON RATIO, ν_{LT}	0.06	
SHEAR MODULUS, G_{LT}	4.5	GPA
TENSILE STRENGTH		
WARP, σ_L	799	MPA
FILL, σ_T	712	MPA



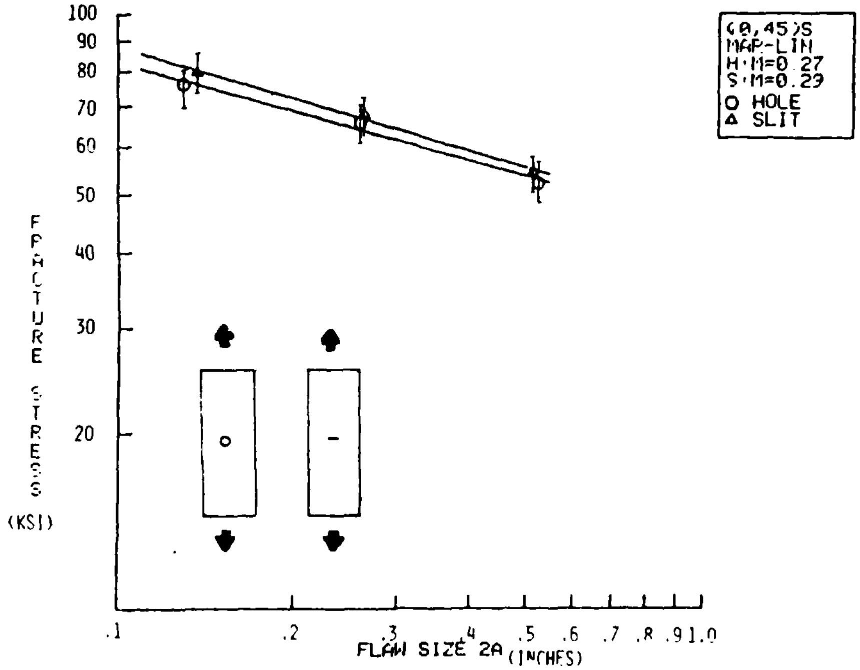
CYLINDER SPECIMEN



COUPON SPECIMEN



CORRELATION OF COUPON DATA

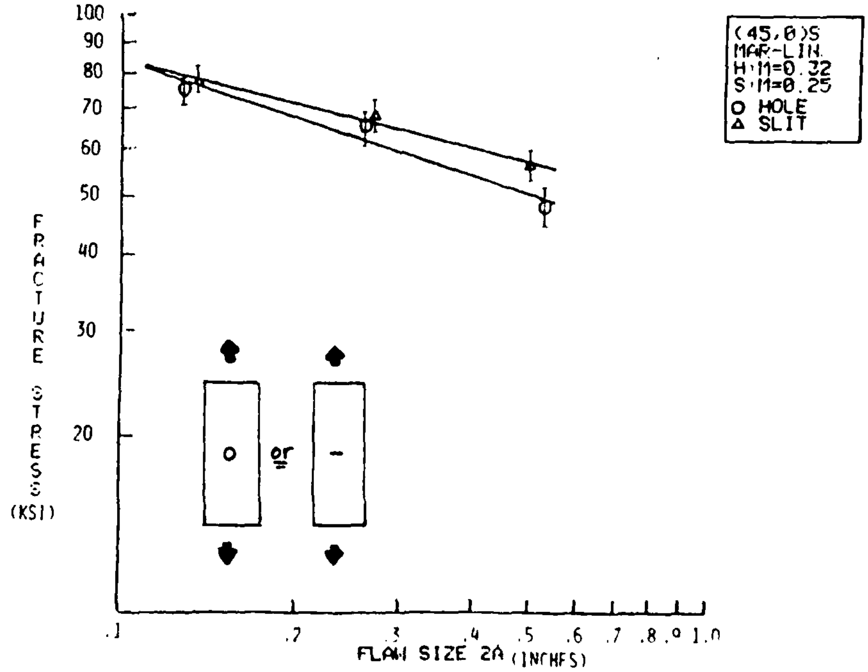


$$\sigma_f = H_c(2a)^{-m}$$

2a = flaw size perpendicular to load

m = .28 (theoretical)

H_c = Composite Fracture Toughness



CORRELATION OF PRESSURIZED CYLINDER DATA

CORRECTION:

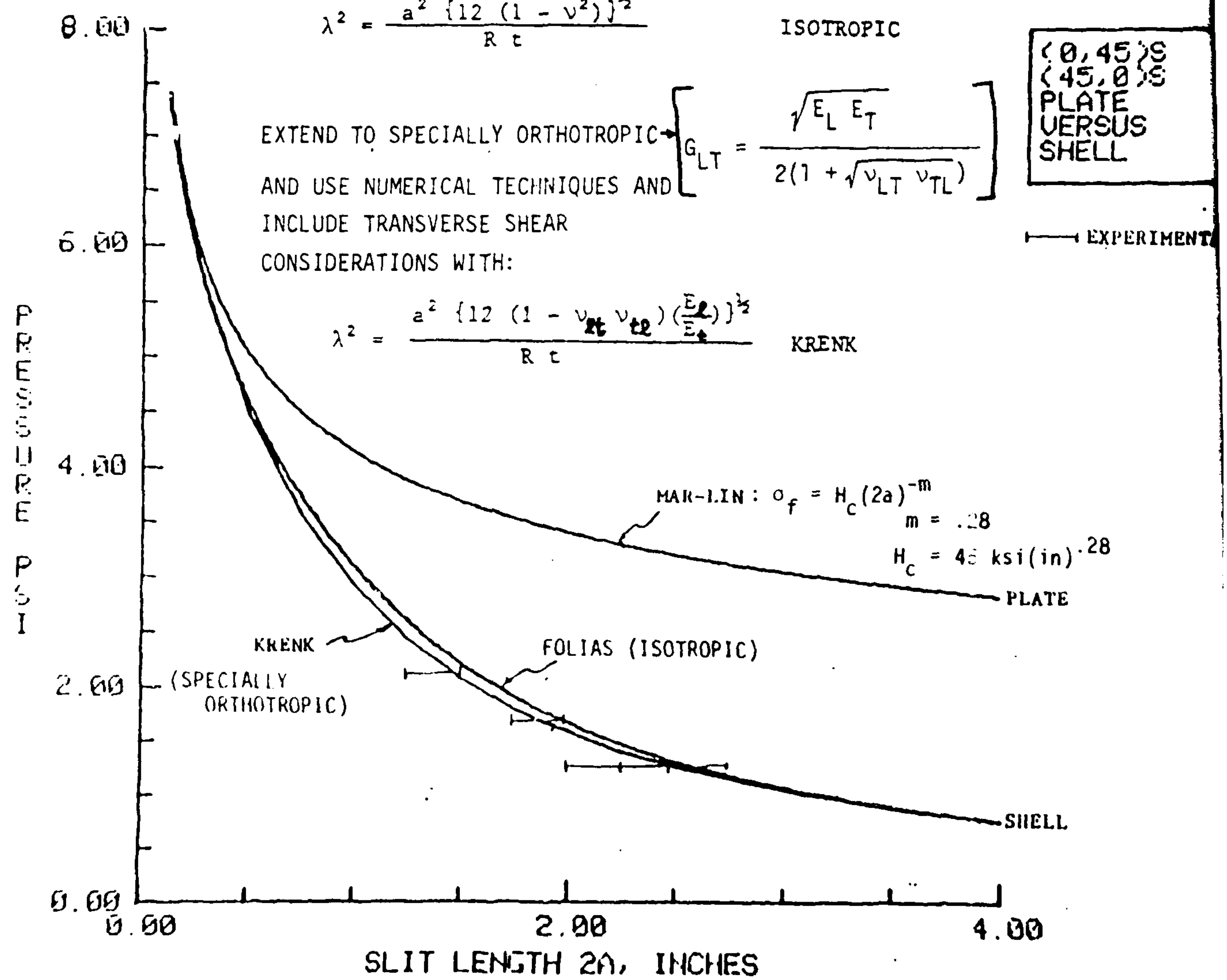
$$\frac{\sigma_{hoop}}{\sigma_{plate}} = \frac{1}{(1 + 0.317\,\lambda^2)^{\frac{1}{2}}} \qquad \text{FOLIAS}$$

$$\lambda^2 = \frac{a^2\,\{12\,(1 - \nu^2)\}^{\frac{1}{2}}}{R\,t} \qquad \text{ISOTROPIC}$$

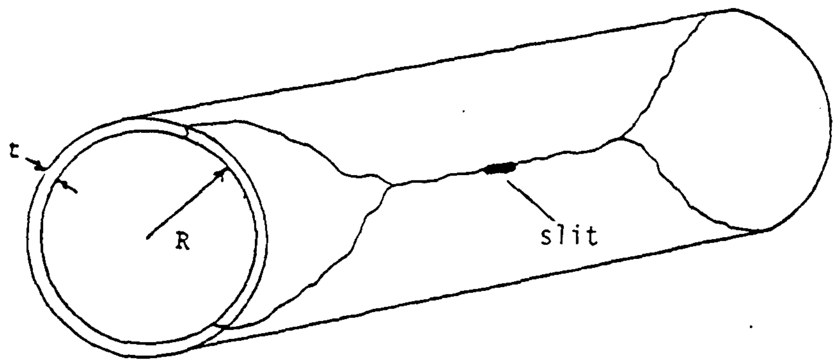
EXTEND TO SPECIALLY ORTHOTROPIC AND USE NUMERICAL TECHNIQUES AND INCLUDE TRANSVERSE SHEAR CONSIDERATIONS WITH:

$$\left[G_{LT} = \frac{\sqrt{E_L\,E_T}}{2(1 + \sqrt{\nu_{LT}\,\nu_{TL}})} \right]$$

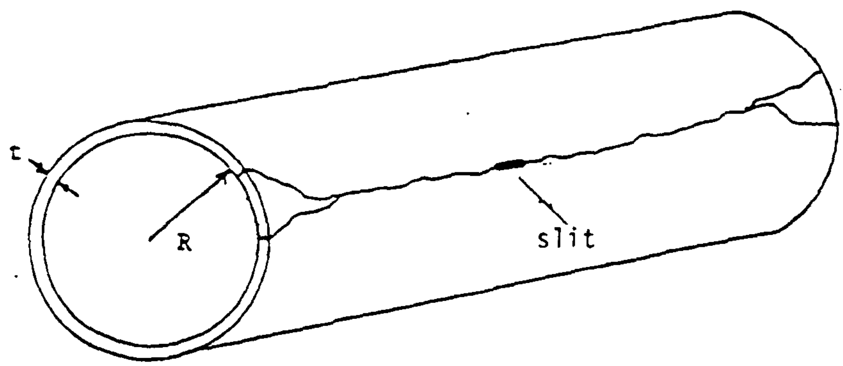
$$\lambda^2 = \frac{a^2\,\{12\,(1 - \nu_{\ell t}\,\nu_{t\ell})(\frac{E_\ell}{E_t})\}^{\frac{1}{2}}}{R\,t} \qquad \text{KRENK}$$



FAILURE MODES OF UNREINFORCED CYLINDERS

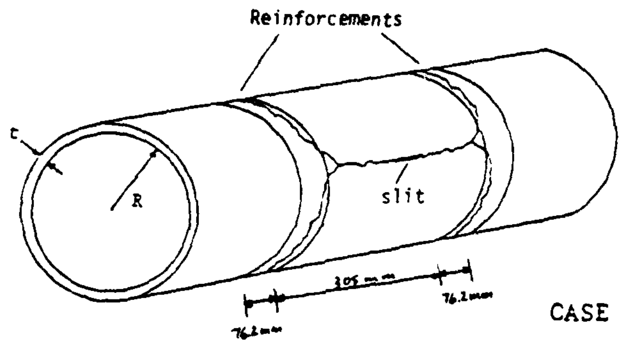


FAILURE MODE -- $(0,45)_s$ LAMINATE



FAILURE MODE -- $(45,0)_s$ LAMINATE

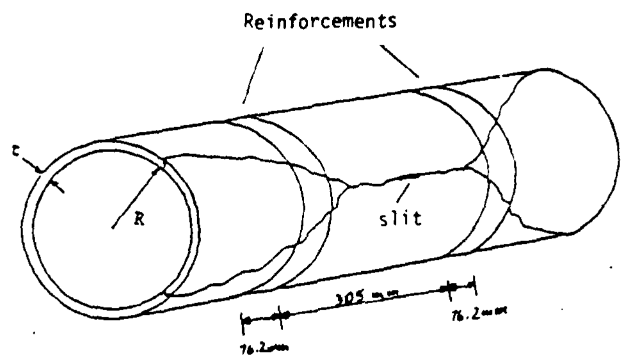
FAILURE MODES OF REINFORCED CYLINDERS



FAILURE MODE -- $(0,45)_s$ REINFORCED

CASE 1 FOUR PLIES OUTER SURFACE ONLY

CASE 2 TWO PLIES INNER SURFACE; TWO PLIES OUTER SURFACE

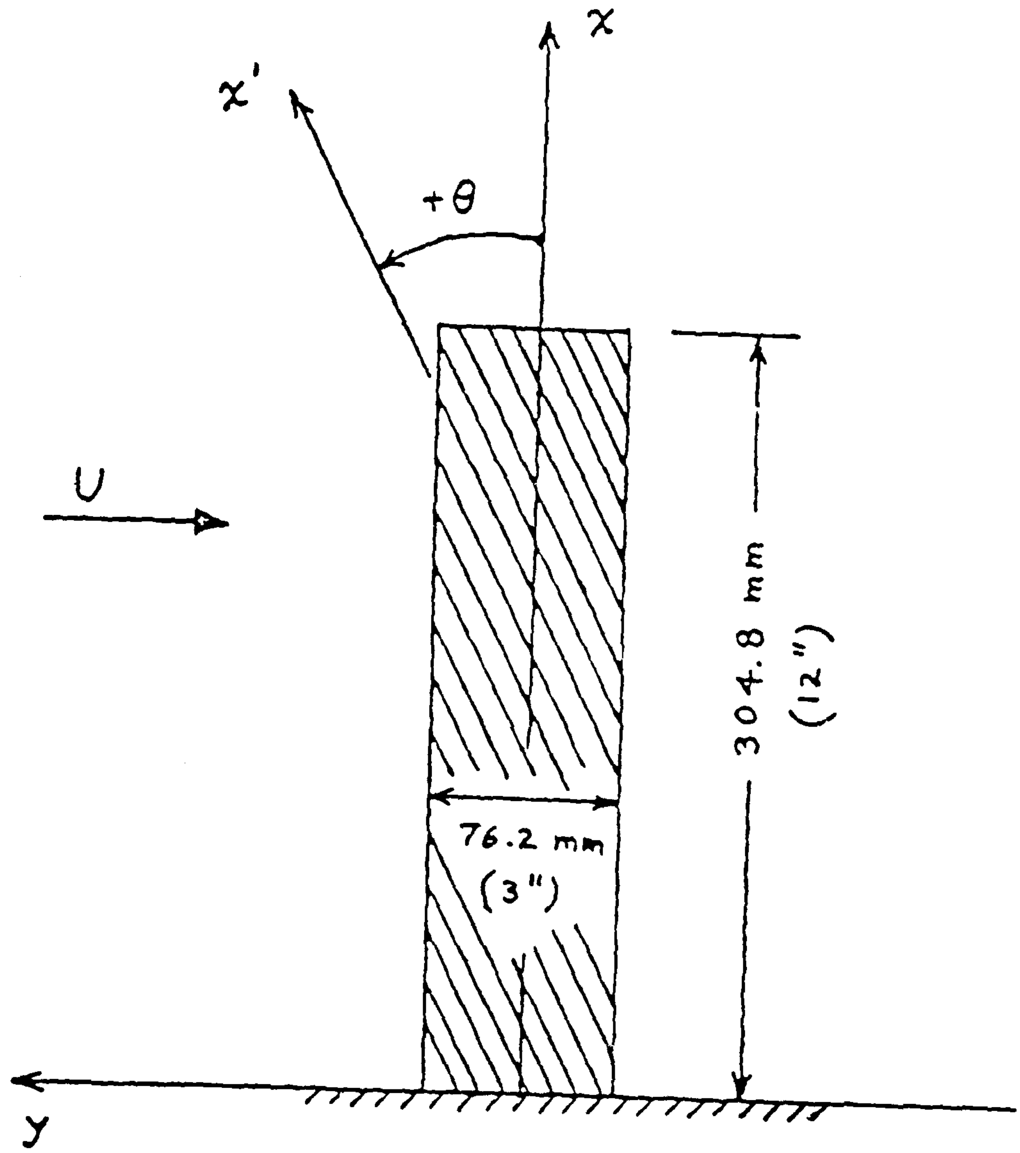


FAILURE MODE -- $(0,45)_s$ REINFORCED

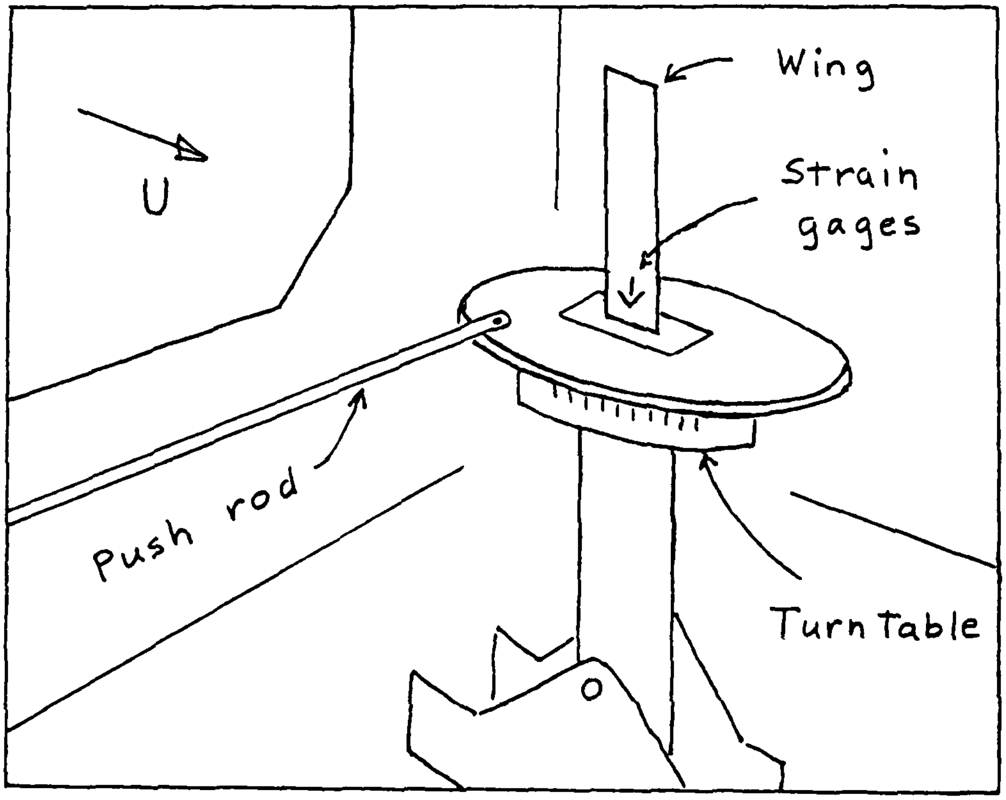
CASE 3 INTERLEAVED:

CLOTH
--TAPE
CLOTH
--TAPE
--TAPE
CLOTH
--TAPE
CLOTH

PLATE LAYOUT AND SIGN CONVENTIONS



WING MODEL IN WIND TUNNEL



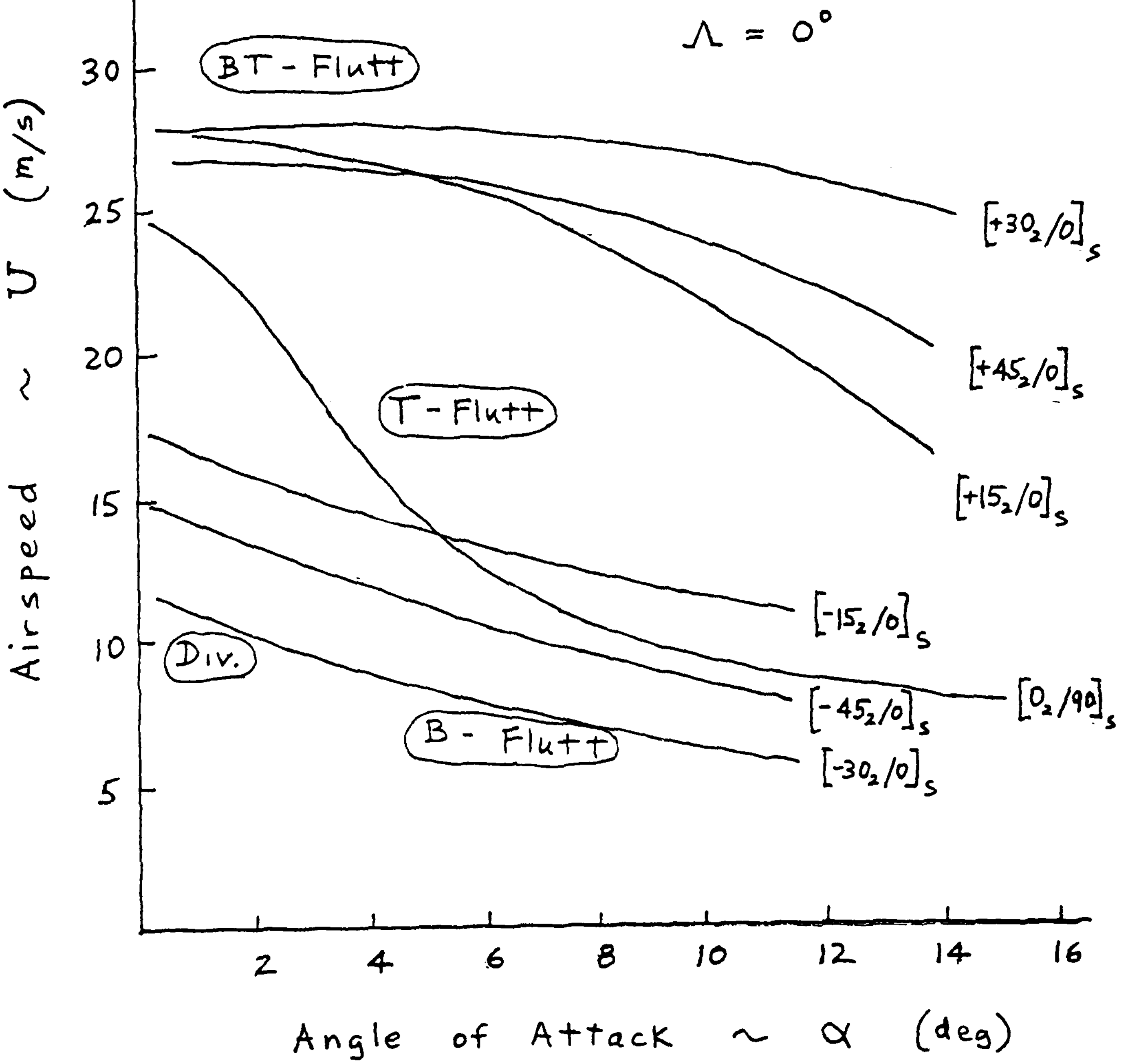
M.I.T. Acoustic Wind Tunnel

1.5 m x 2.3 m (5' x 7.5')

U = 0 - 32 m/s (0 - 105 '/s)

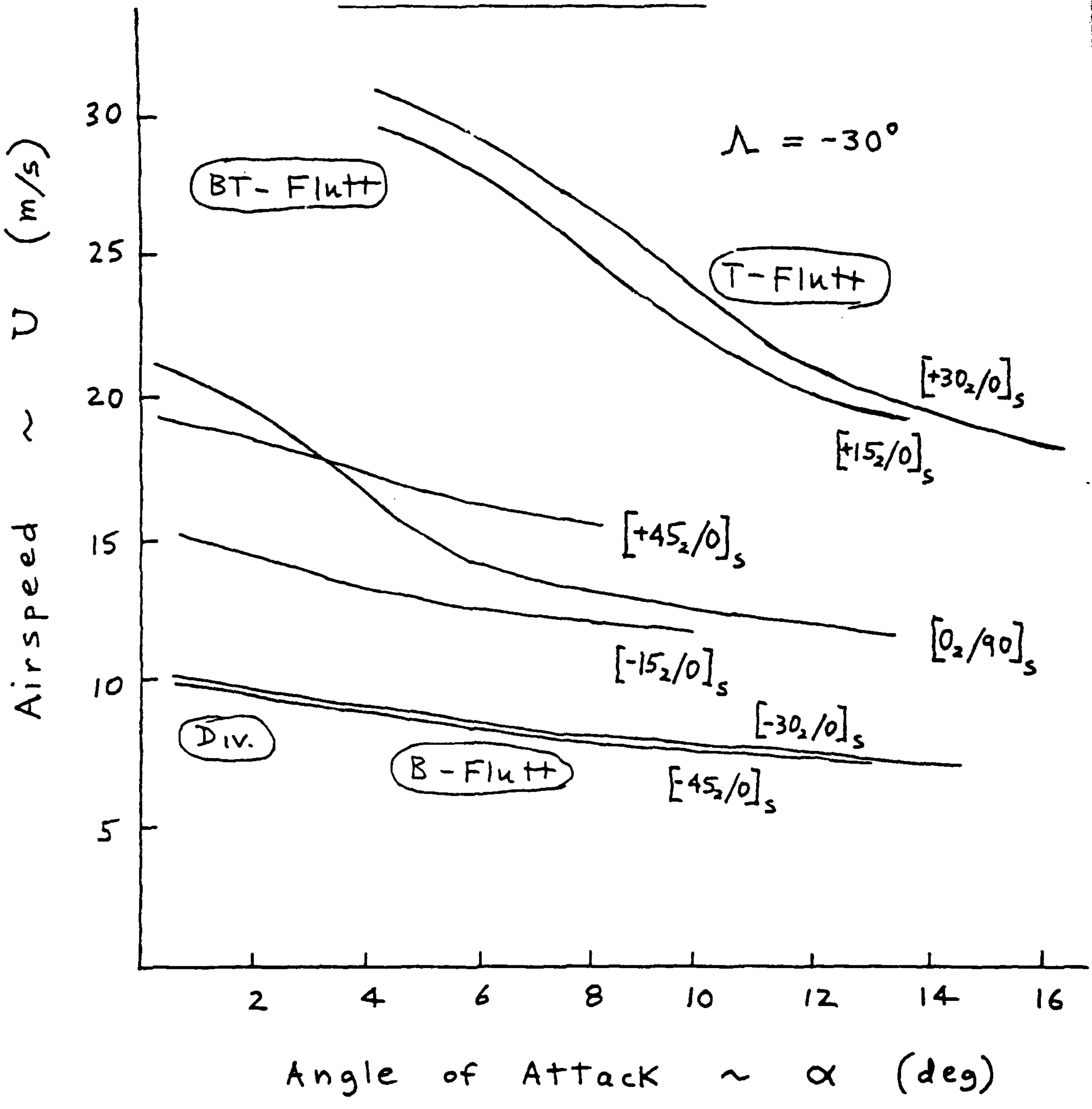
FLUTTER AND DIVERGENCE SPEEDS

FOR STRAIGHT WINGS (Λ=0)



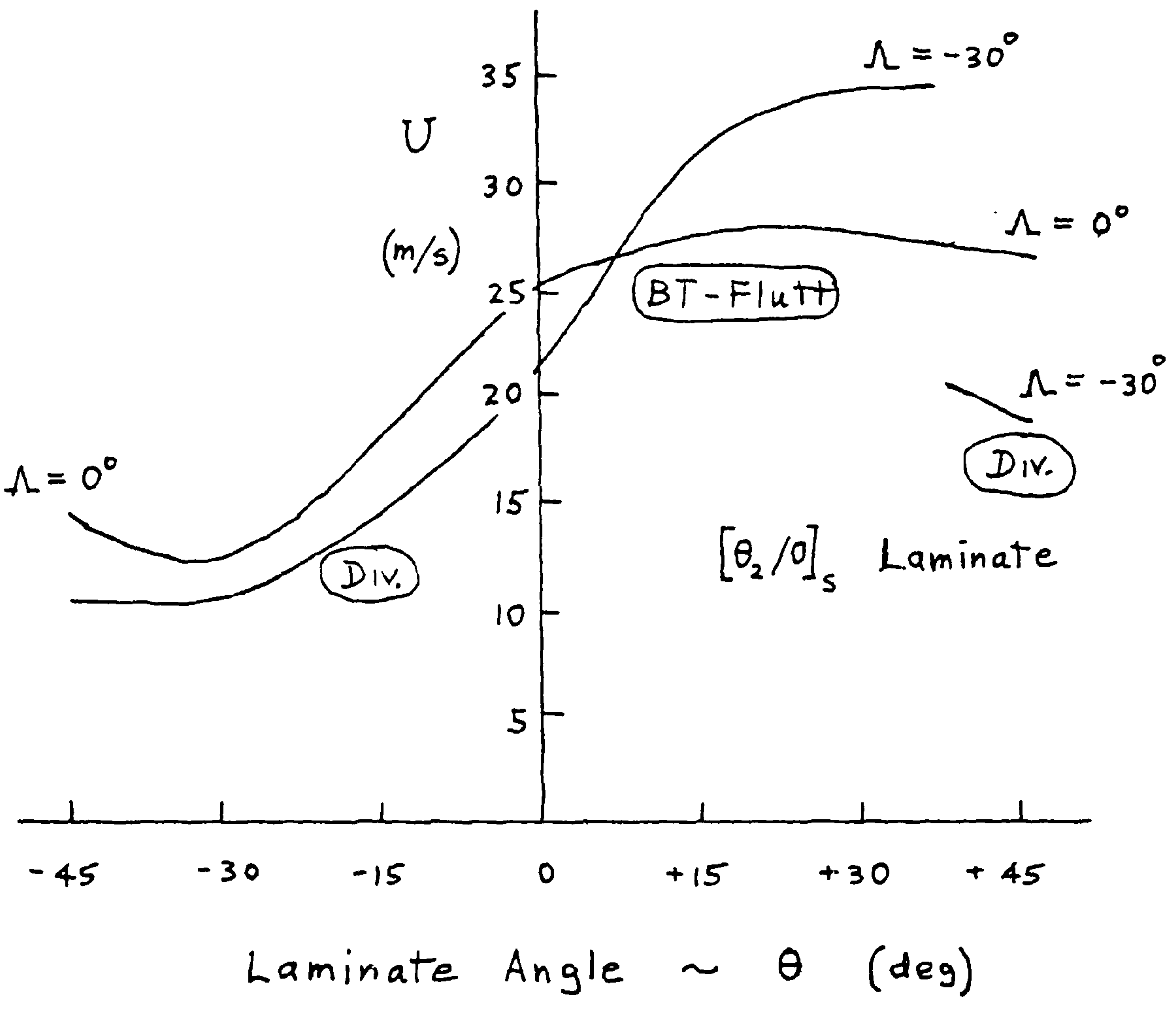
FLUTTER AND DIVERGENCE SPEEDS

FOR FORWARD SWEEP (Λ=-30°)



FLUTTER AND DIVERGENCE SPEEDS

AT LOW ANGLES OF ATTACK



FLUTTER AND DIVERGENCE SPEEDS AT LOW ANGLES OF ATTACK ($\Lambda = 0°$)

DIVERGENCE VELOCITIES

Plate	U_D (m/s) Theo.	Exp.
$[0_2/90]_s$	25.3	—
$[\pm 45/0]_s$	∞	> 32
$[-45_2/0]_s$	11.3	12.5
$[-30_2/0]_s$	11.6	11.9

FLUTTER VELOCITIES & FREQUENCIES

Plate	U_F (m/s) Theo.	Exp.	ω_F (Hz) Theo.	Exp.
$[0_2/90]_s$	20.7	25	25	29
$[\pm 45/0]_s$	38.7	> 32	39	—
$[+45_2/0]_s$	27.7	28	28	24
$[+30_2/0]_s$	27.7	27	31	28

NONLINEAR TRANSIENT ANALYSIS OF COMPOSITE PLATES

J. N. REDDY

Professor

Department of Engineering Science and Mechanics

Virginia Polytechnic Institute

Sponsored by

THE AIR FORCE OFFICE OF SCIENTIFIC RESEARCH

OBJECTIVES

TO DEVELOP:

- A FINITE ELEMENT FOR TRANSIENT ANALYSIS –

  shear-deformation theory

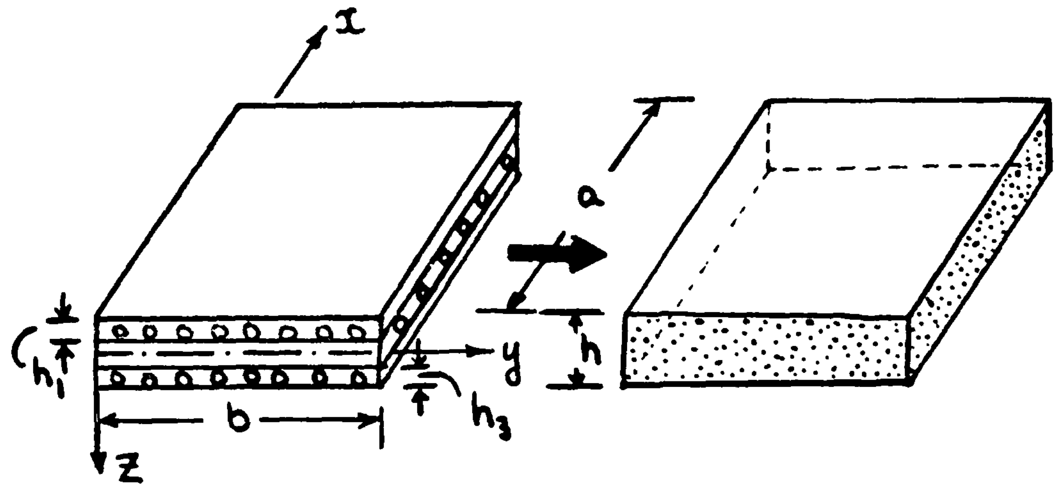
  geometric nonlinearity (in the von Karman sense)

- ANALYTICAL SOLUTIONS FOR VERIFICATION

CONCLUSIONS

- global response is accurate
- transverse shear effect is significant
- coupling effect is significant
- local response is not good
- stress waves due to impact cannot be predicted

LAMINATED-PLATE/SHELL THEORY



Laminated plate constructed of homogeneous, orthotropic layers of uniform thickness $\Rightarrow$ Homogeneous, anisotropic plate

$$\{\sigma\}_{layer} = [Q_{ij}]_{layer}\{\varepsilon\}_{layer} \Longrightarrow \left\{\begin{array}{c} N \\ \hline M \end{array}\right\}_{plate} = \left[\begin{array}{c|c} A & B \\ \hline B & D \end{array}\right]_{plate} \left\{\begin{array}{c} \varepsilon \\ \hline \chi \end{array}\right\}_{plate}$$

(in plate coordinates)

VARIATIONAL FORMULATION OF THE GOVERNING EQUATIONS

Governing equations:

$$N_{1,x} + N_{6,y} = \underline{P u_{,tt}} + \underline{R \psi_{x,tt}}$$
$$N_{6,x} + N_{2,y} = \underline{P v_{,tt}} + \underline{R \psi_{y,tt}}$$
$$Q_{1,x} + Q_{2,y} + \mathcal{N}(N_i, w) = P w_{,tt}$$
$$M_{1,x} + M_{6,y} - Q_1 = \underline{I \psi_{x,tt}} + \underline{R u_{,tt}}$$
$$M_{6,x} + M_{2,y} - Q_2 = \underline{I \psi_{y,tt}} + \underline{R v_{,tt}}$$

$$(P, R, I) = \int_{-h/2}^{h/2} (1, z, z^2) \rho \, dz$$

$$\mathcal{N}(w, N_i) = \frac{\partial}{\partial x}\left(N_1 \frac{\partial w}{\partial x}\right) + \frac{\partial}{\partial y}\left(N_6 \frac{\partial w}{\partial x}\right) + \frac{\partial}{\partial x}\left(N_6 \frac{\partial w}{\partial y}\right) + \frac{\partial}{\partial y}\left(N_2 \frac{\partial w}{\partial y}\right)$$

Variational Formulation

$$\begin{aligned} 0 = \int_{R^e} \Big\{ & \delta u \left(P u_{,tt} + R \psi_{x,tt}\right) + \delta u_{,x} N_1 + \delta u_{,y} N_6 + \cdots \\ & + \frac{\partial \delta w}{\partial x} \frac{\partial w}{\partial x} N_1 + \frac{\partial \delta w}{\partial y} \frac{\partial w}{\partial x} N_6 + \frac{\partial \delta w}{\partial x} \frac{\partial w}{\partial y} N_6 + \frac{\partial \delta w}{\partial y} \frac{\partial w}{\partial y} N_2 \\ & + \delta \psi_x \left(I \psi_{x,tt} + R u_{,tt}\right) + \delta \psi_{x,x} M_1 + \delta \psi_{x,y} M_6 \\ & + \delta \psi_x Q_1 + \cdots \Big\} dx\, dy \\ & + \int_{C_n} (\delta u_n N_n + \delta u_s N_{ns}) \, ds + \int_{C_q} \delta w \, q \, ds + \int_{C_m} (\delta \psi_n M_n + \delta \psi_s M_{ns}) \, ds \end{aligned}$$

FINITE ELEMENT MODEL

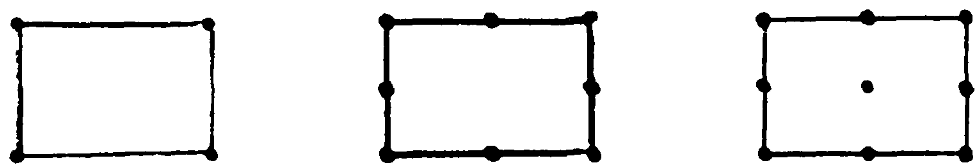
$$u=\sum_i^r u_i\phi_i^1\,,\quad v=\sum_i^r v_i\phi_i^1\,,\quad w=\sum_i^s w_i\phi_i^2$$

$$\psi_x=\sum_i^p \psi_{xi}\phi_i^3\,,\quad \psi_y=\sum_i^p \psi_{yi}\phi_i^3$$

$$[M]\{\ddot{\Delta}\}+[K]\{\Delta\}=\{F\}$$

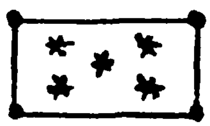
$$\phi_i^1=\phi_i^2=\phi_i^3$$

Linear	Serendipity	Lagrange



20X20	40X40	45X45

- Newmark's direct integration scheme ($\alpha=1/2$, $\beta=1/4$)
- Reduced Integration:



linear

* full integration
* reduced integration

TIME INTEGRATION SCHEME

The Newmark method:

$$\{\ddot{\Delta}\}_{n+1} = a_0\left(\{\Delta\}_{n+1} - \{\Delta\}_n\right) - a_1\{\dot{\Delta}\}_n - a_2\{\ddot{\Delta}\}_n$$

$$\{\dot{\Delta}\}_{n+1} = \{\dot{\Delta}\}_n + a_3\{\ddot{\Delta}\}_n + a_4\{\ddot{\Delta}\}_{n+1}$$

$$a_0 = \frac{1}{\beta(\Delta t)^2}, \quad a_1 = a_0\Delta t, \quad a_2 = \frac{1}{2\beta} - 1$$

$$a_3 = (1-\alpha)\Delta t, \quad a_4 = \alpha\Delta t$$

$\Rightarrow$ $\alpha = 0.5$, $\beta = 0.25$ (constant-average-acceleration method)

$\downarrow$

Numerical scheme $\rightarrow$

$$[K]\{\Delta\} + [M]\{\ddot{\Delta}\} = \{F\}$$

$\downarrow$

$$[\bar{K}]\{\Delta\}_{n+1} = \{\bar{F}\}_{n,n+1}$$

$$[\bar{K}] = [K] + a_0[M]$$

$$\{\bar{F}\} = \{F\}_{n+1} + [M]\left(a_0\{\Delta\}_n + a_1\{\dot{\Delta}\}_n + a_2\{\ddot{\Delta}\}_n\right)$$

$\downarrow$

Computational scheme $\rightarrow$
(for nonlinear problems)

$$[K(\{\Delta\}^r_{n+1})]\{\Delta\}^{r+1}_{n+1} = \{\bar{F}\}_{n,n+1}$$

r – iteration no.
n – time step no.

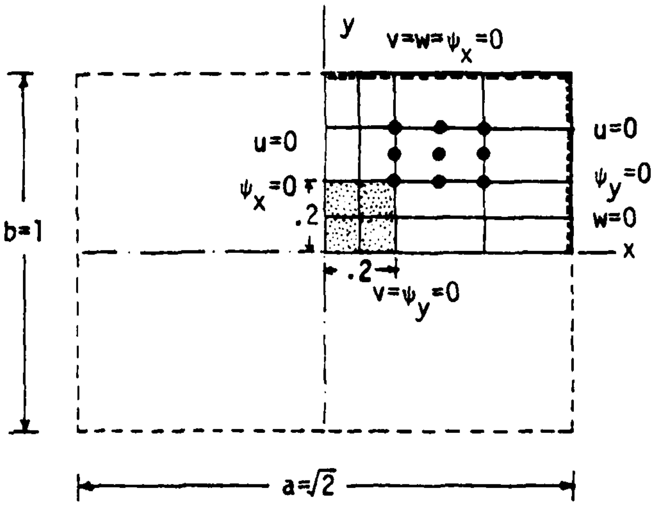
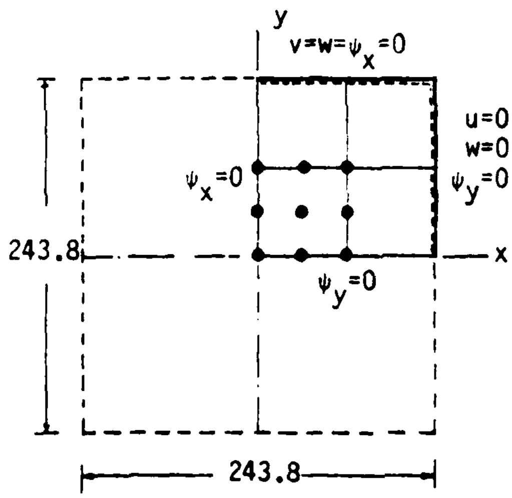


Figure 1. Finite-element mesh and boundary conditions for isotropic rectangular plates under suddenly applied pressure loading at the center.



BC1: $u(0,y)=v(x,0)=0$

BC2: $u(y,0)=v(0,y)=0$

Figure 2. Finite-element mesh and boundary conditions for isotropic square plates under suddenly applied pressure loading.

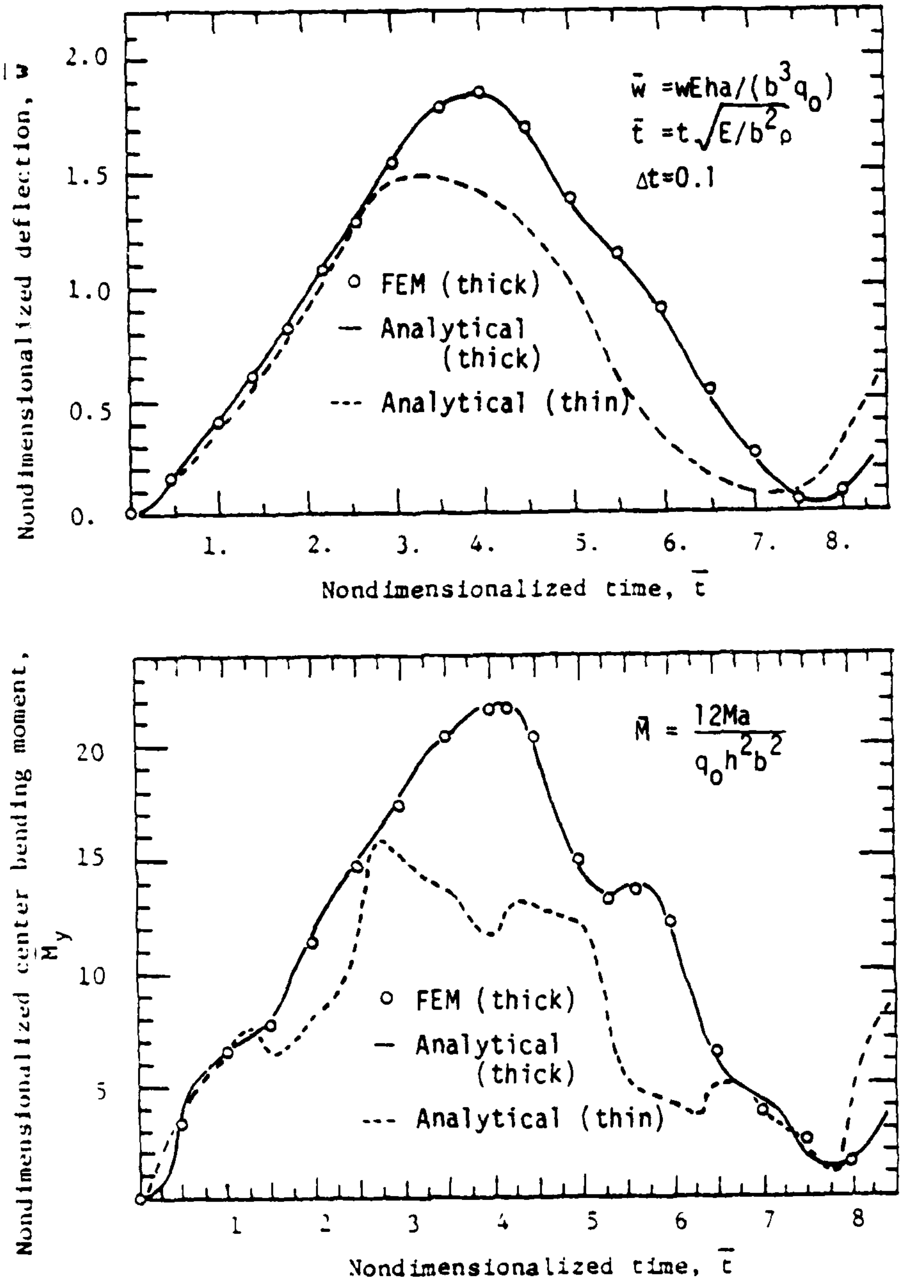


Figure 3. Nondimensionalized center deflection and bending moment versus nondimensionalized time for simply supported rectangular plates ($\nu = 0.3$) under suddenly applied pressure loading at the center square area (4x4 mesh).

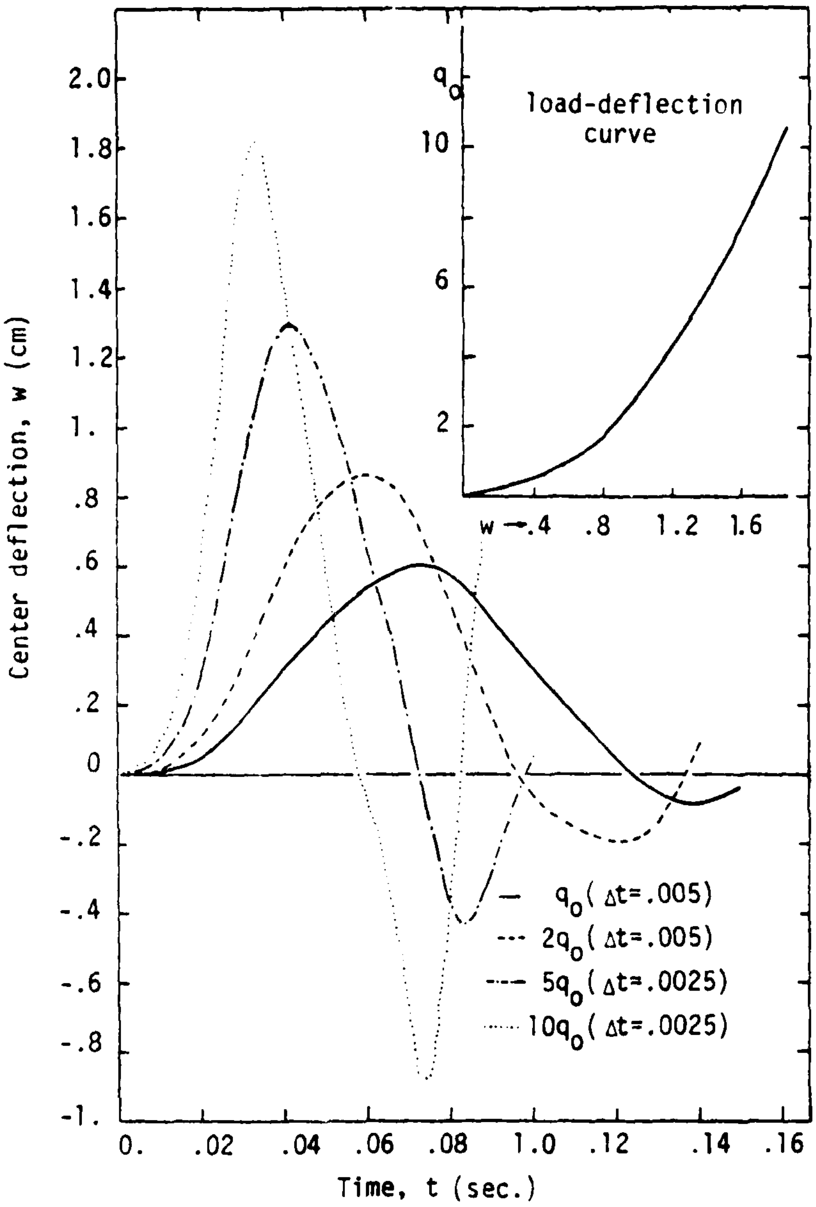


Figure 4. Nonlinear transient response of isotropic plates (see Figure 2 for the data and finite element mesh and boundary conditions)under suddenly applied uniformly distributed loads.

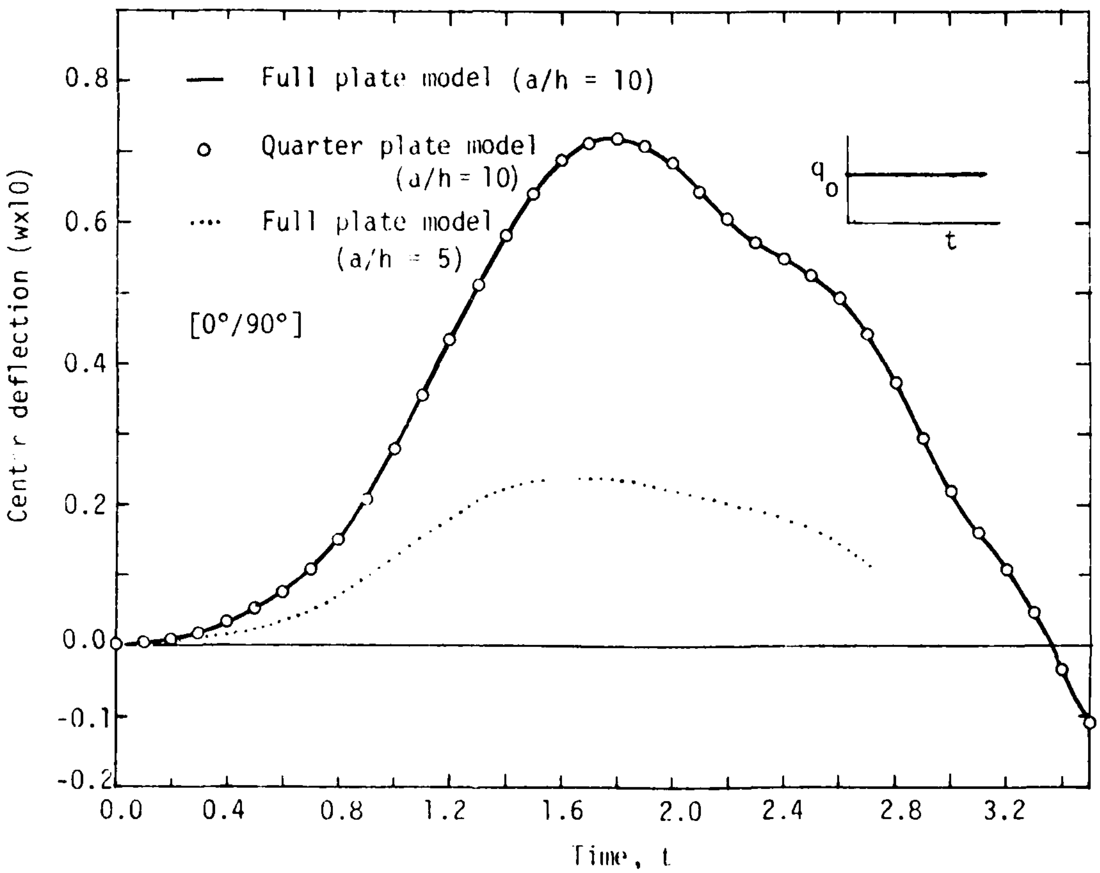


Figure 5. The effect of plate thickness and finite-element models (used to analyze the transient response) on the center deflection of the plate.

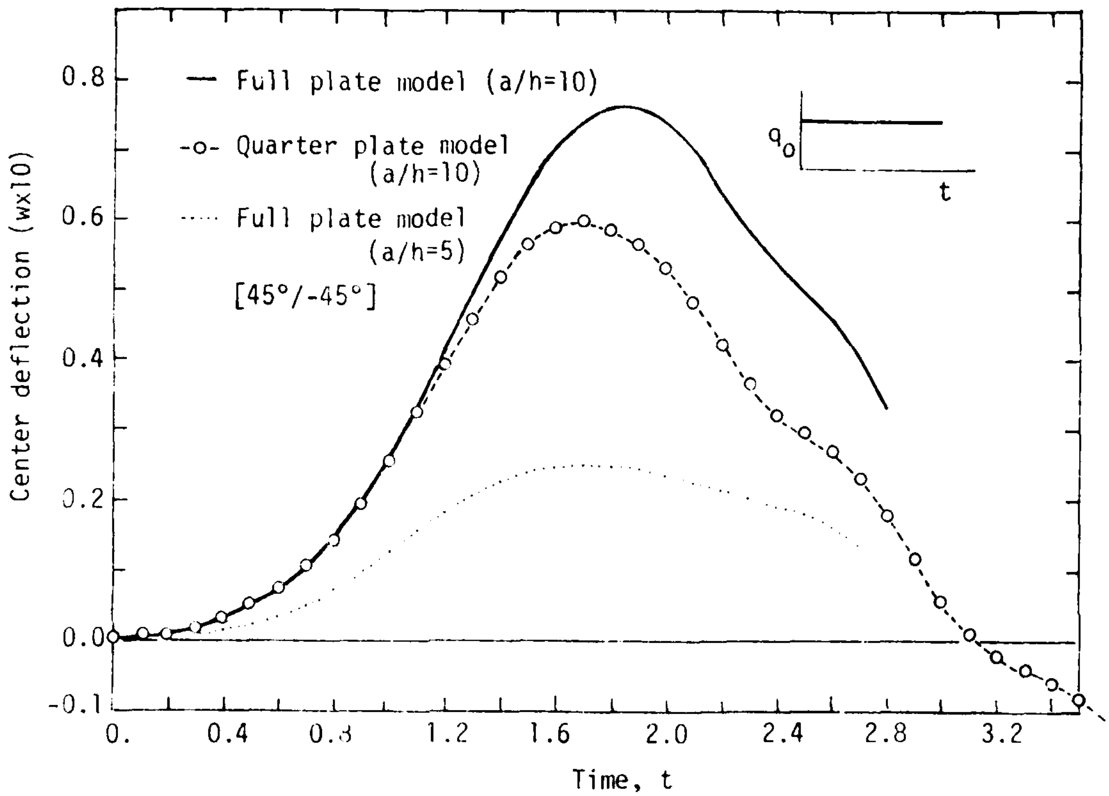


Figure 6 . The effect of plate thickness and finite-element models (used to analyze the transient response)on the center deflection

Materials Sciences Corporation

IMPROVED CERAMIC FRACTURE BEHAVIOR FOR HIGH TEMPERATURE TURBINE APPLICATIONS

AFOSR CONTRACT F44620-82-C-0041

OBJECTIVES

- DETERMINE FEASIBILITY OF DEVELOPING AN ANALYTICAL MODEL TO INVESTIGATE FRACTURE BEHAVIOR AND STRENGTH OF CERAMIC COMPOSITES
  - IDENTIFY ALTERNATE ANALYTICAL APPROACHES FOR MATERIAL MODEL
  - FABRICATE WHISKER REINFORCED CERAMICS AND EXPERIMENTALLY MEASURE FRACTURE TOUGHNESS AND TENSILE STRENGTH
  - COMPARE EXPERIMENTAL FAILURE MECHANISMS TO ANALYTICAL METHODS TO CHOOSE THEORY MOST REPRESENTATIVE OF MATERIAL BEHAVIOR

CURRENT STATUS

- THREE POTENTIAL ANALYTICAL APPROACHES HAVE BEEN DEFINED FOR DATA COMPARISON

  (1) TORTOUS PATH CRACK PROPAGATION

  (2) STRENGTH VS. SIZE

  (3) CRITICAL MATRIX STRESS

- MATERIAL IS PRESENTLY BEING FABRICATED

- EXPERIMENTAL MATRIX IS DEFINED

TORTUOUS PATH APPROACH

- HYPOTHESIS

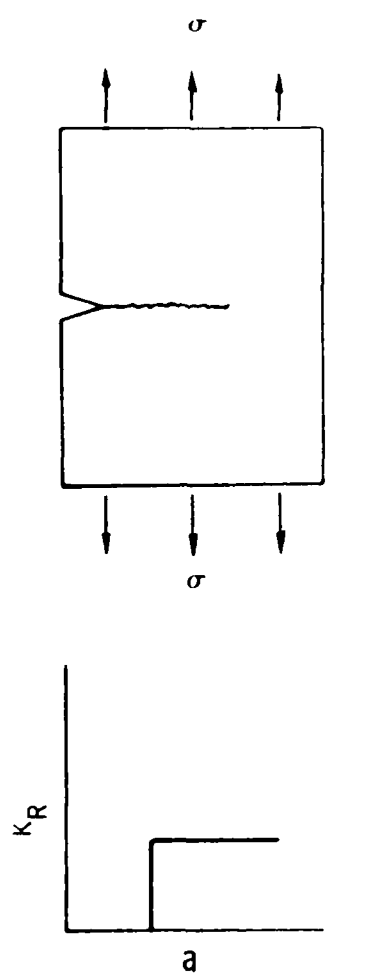
  - IN A WHISKER REINFORCED MATERIAL, A CRACK WILL NOT GROW IN A STRAIGHT LINE BUT WILL BE FORCED TO CHANGE DIRECTIONS BECAUSE OF THE PRESENCE OF THE WHISKERS

  - AS THE CRACK PATH CHANGES DIRECTION, IT WILL REQUIRE A HIGHER ENERGY LEVEL TO PROPAGATE

  - THEREFORE, A WHISKER REINFORCED MATERIAL WILL POSSESS A HIGHER FRACTURE TOUGHNESS THAN A BULK MATERIAL

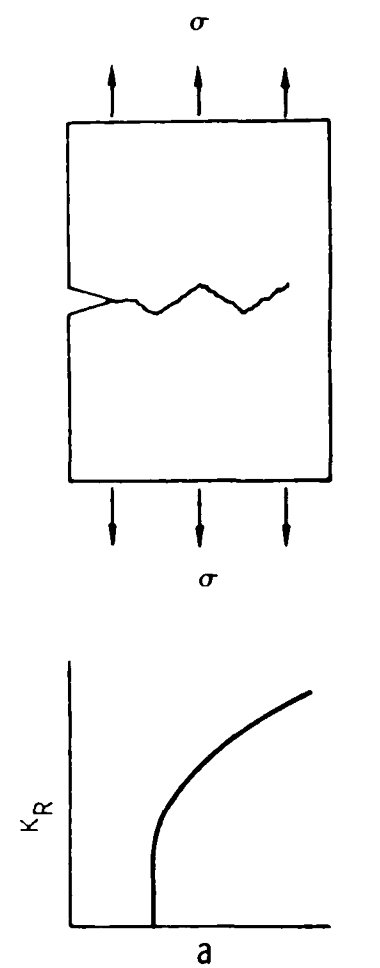
TORTUOUS PATH APPROACH

HOMOGENOUS MATERIAL

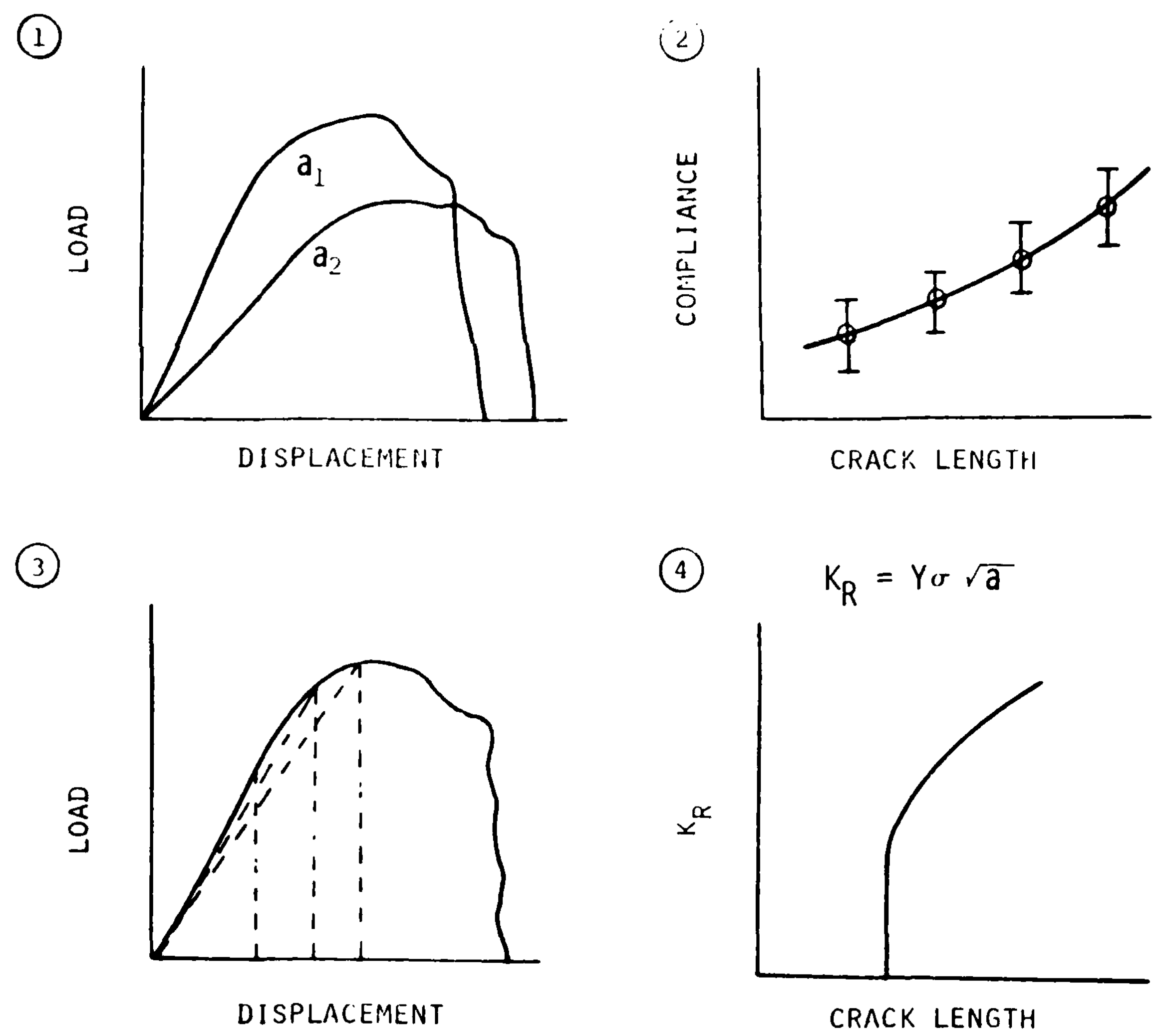


$$K_R = Y\sigma\sqrt{a}$$

REINFORCED MATERIAL



CRACK GROWTH RESISTANCE CURVES



STRENGTH VS. SIZE APPROACH

- HYPOTHESIS

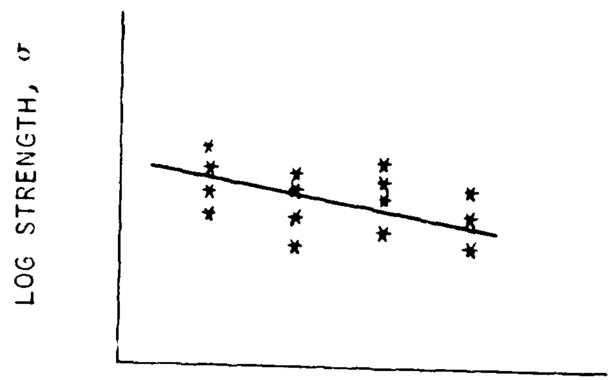
  - THE STRENGTH OF A BRITTLE MATERIAL (CERAMIC) IS GOVERNED BY THE PRESENCE OF FLAWS. THUS, THE MEASURED STRENGTH DEPENDS UPON THE PROBABILITY OF THE EXISTENCE OF A CRITICAL FLAW (SPECIMEN SIZE)

  - THE PRESENCE OF THE WHISKERS EFFECTIVELY CHANGES THE MATERIAL FROM A LARGE HOMOGENEOUS MEDIUM TO AN ASSEMBLAGE OF SMALL PARTICLES

  - THE STRENGTH OF THE REINFORCED MATERIAL WILL BE RELATED TO THE STRENGTH DISTRIBUTION OF SPECIMENS OF SIZE EQUAL TO THE PARTICLE SIZE

WEIBULL STRENGTH DISTRIBUTION

$$\sigma = (\alpha \beta L e)^{-1/\beta}$$



- DETERMINE WEIBULL PARAMETERS FROM HOMOGENOUS CERAMIC TESTS
- DEFINE PARTICLE SIZE FOR REINFORCED MATERIAL FROM LOCAL GEOMETRY
- CALCULATE REINFORCED STRENGTH BASED UPON ASSEMBLAGE MODEL AND COMPARE TO EXPERIMENTS

CRITICAL MATRIX STRESS APPROACH

- HYPOTHESIS

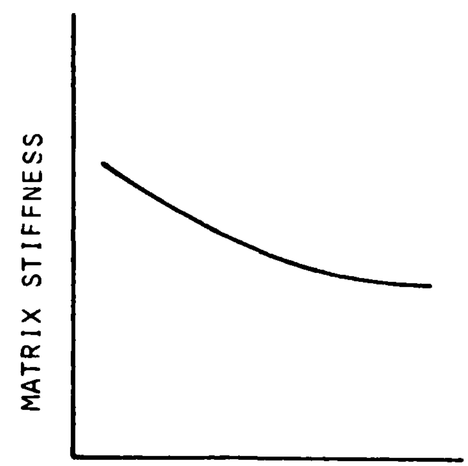
  - THE FAILURE OF A WHISKER REINFORCED CERAMIC IS GOVERNED BY DAMAGE ACCUMULATION WITHIN THE MATRIX PHASE

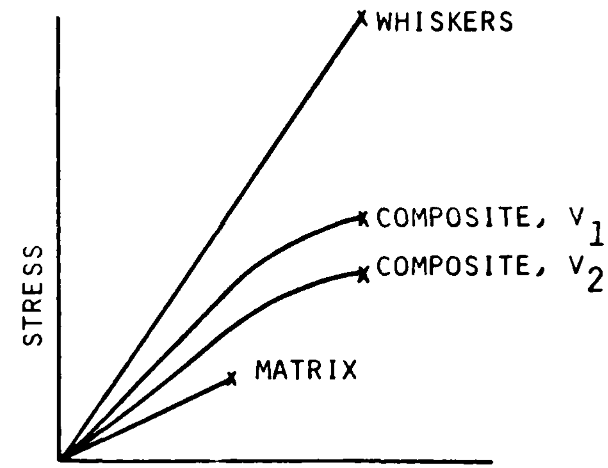
  - FAILURE WILL OCCUR WHEN THE AVERAGE MATRIX MODULUS REACHES SOME CRITICAL VALUE

  - BY ADDING WHISKERS, DAMAGE ACCUMULATION IN THE MATRIX WILL BE REDUCED LEADING TO IMPROVED COMPOSITE STRENGTH

  - THUS, COMPOSITES FABRICATED WITH TWO DIFFERENT VOLUME FRACTIONS WILL HAVE DIFFERENT COMPOSITE STRENGTHS BUT THE SAME MATRIX STRESS-STRAIN CURVE

CRITICAL MATRIX STRESS APPROACH





- MATRIX STIFFNESS WILL DECREASE AS CRACK DENSITY INCREASES

- WHISKERS WILL ACT AS CRACK ARRESTORS, REDUCING CRACK DENSITY

- RESULTING COMPOSITE WILL SHOW NON-LINEAR STRESS-STRAIN RESPONSE THAT IS A FUNCTION OF THE WHISKER VOLUME FRACTION

REQUIRED EXPERIMENTS

LOAD-DISPLACEMENT CURVES

TORTUOUS PATH	STRENGTH VS. SIZE	CRITICAL MATRIX σ	GENERAL
o HOMOGENOUS MATERIAL	o HOMOGENOUS MATERIAL	o REINFORCED MATERIAL	o VOLUME FRACTION
SEVERAL a_0'S	UNCRACKED	V_1	o ASPECT RATIO
o REINFORCED MATERIAL	o REINFORCED MATERIAL	o REINFORCED MATERIAL	o SEM'S
SEVERAL a_0'S	UNCRACKED	V_2	o PHOTO MICROGRAPHS
			o ACOUSTIC EMISSIONS

TEST MATRIX

MATERIAL 1 HOMOGENEOUS	MATERIAL 2 REINFORCED, V_1	MATERIAL 3 REINFORCED, V_2
NO CRACK (10)*; AE**	NO CRACK (5); AE	NO CRACK (5); AE
a_1 (5)	a_1 (5)	a_1 (5)
a_2 (5); AE	a_2 (5); AE	a_2 (5); AE
a_3 (5)	a_3 (5)	a_3 (5)

* NUMBERS IN PARENTHESES ARE NUMBERS OF SPECIMENS

** AE - ACOUSTIC EMISSIONS

APPENDIX A

ABSTRACTS

TITLE: POSTBUCKLING BEHAVIOR OF GRAPHITE-EPOXY PANELS LOADED IN COMPRESSION
J. H. Starnes, Jr., M. Rouse, M. Stein, and N. Knight, Jr., NASA Langley Research Center

A summary of recent NASA Langley Research Center studies of the post-buckling behavior of graphite-epoxy panels loaded in compression is presented. Results for both unstiffened and stiffened flat panels are described. Typical experimental and analytical results are compared. The parameters that govern postbuckling response are identified. In addition to the orthotropic plate buckling parameters, only one new parameter is needed to describe the post-buckling response of an orthotropic plate loaded in compression. The failure modes that limit postbuckling strength are described. Failures occur due to high transverse shearing forces or skin-stiffener separation. The effects of circular holes and low-speed impact damage on postbuckling strength are described.

TITLE: SPECTRUM FATIGUE BAHAVIOR OF POSTBUCKLED SHEAR PANELS
B. L. Agerwall, Northorp Corporation

An experimental program was conducted to examine the room temperature, dry postbuckling behavior of composite shear panels under typical V/STOL spectrum fatigue loading. Tests were also conducted to supplement previously obtained constant amplitude test data to define a fatigue life curve. The constant amplitude fatigue life curves was used to predict fatigue life under spectrum loading.

The panels tested under spectrum loading demonstrated excellent fatigue life. The panels subjected to maximum spectrum loads of 80 percent of the average static strength, survived two lifetimes without any loss in stiffeners or strength. These results indicate that fatigue loading is not likely to be a significant factor in the design of postbuckled composite shear panels, especially under room temperature dry conditions.

Three constant amplitude fatigue tests, in addition to twelve panel tests conducted under a previous Navy program were used to obtain the fatigue life curve for panels subjected to constant amplitude fatigue loading. The fatigue life curve was used in conjunction with Miner's rule to predict the fatigue life of panels subjected to spectrum loading. Good correlation between predicted and experimentally obtained fatigue lives was obtained.

TITLE: DEVELOPMENT OF ANALYSIS FOR PREDICTING COMPRESSION FATIGUE LIFE AND RESIDUAL STRENGTH IN COMPOSITES
M. Ratwani and H. Kan, Northrop Corporation

A generalized macromechanics model, based on the delamination propagation between the plies of a composite laminate, has been developed for predicting compression fatigue life and residual strength in composites. The model is applicable to laminates in which significant interlaminar shear and normal stresses are present.

Test data have been generated under both constant amplitude and flight-by-flight spectrum loading on two different AS/3501-6 graphite/epoxy laminates. Good correlation between experimentally observed constant amplitude fatigue data and analytical predictions has been obtained. Residual strength test data also show good agreement with predictions.

TITLE: MICROBUCKLING INITIATED FAILURE IN TOUGH RESIN LAMINATES
J. Williams, NASA Langley Research Center

Local discontinuities such as holes and damage resulting from impact can cause significant reductions in the strength of compression loaded graphite-epoxy laminates and structures. Recent studies indicate delamination-resistant resins can improve the residual strength of graphite-epoxy laminates damaged by impact, however, the compressive strength of delamination-resistant laminates with holes is not increased. To understand this paradox, the mechanisms of failure are examined and an explanation for the different effects is proposed based on experimental evidence. The residual strength of two stiffened panels (one constructed using a delamination-prone resin and the other a delamination-resistant resin) following the local failure of one of the stiffeners is compared and discussed relative to compression failure mechanisms.

TITLE: SUMMARY OF IMPACT WORK IN THE FATIGUE AND FRACTURE BRANCH
W. Illg, NASA Langley

Progress was made in five areas of impact research and phenomena: (1) the effects of a polysulfone matrix versus epoxy, (2) an analysis to predict progressive impact damage, (3) the effects of deliberate partial bonding on impact resistance under axial preload, (4) ultrasonic evaluation of impact damage, and (5) a semi-empirical technique to predict the residual tensile strength after impact. In all of these efforts, quasi-isotropic graphite-reinforced polymers were used. Using quasi-static transverse loading, it was shown that in small circular plates significant delamination preceded fiber failure, whereas the reverse was true in large plates. A laminate with a ductile matrix reached the same maximum load as epoxy, but with much less delamination. A potential energy model of the circular plates predicted damage accumulation in the correct sequence. Partial bonding, although effectively increasing fracture strength, drastically reduced the compressive prestrain that could be sustained under impact. A new frequency-sweep C-scan showed promise to differentiate modes of impact damage. A fracture mechanics correlation was used to conservatively predict residual tensile strengths.

TITLE: CHARACTERIZATION OF INTERLAMINAR FRACTURE TOUGHNESS IN COMPOSITE MATERIALS
James M. Whitney, AFWAL/Materials Laboratory

The double cantilever beam test is examined as a candidate for measuring interlaminar fracture resistance as related to normal stress induced delamination. A number of approaches to data reduction schemes used in conjunction with this test method for determining critical strain energy release rate are discussed. Experimental data on unidirectional tape graphite fiber reinforced polymeric matrix composites are compared to assess the potential of the double cantilever beam test as a materials screening tool. Matrix materials of varying inherent toughness were chosen for comparative purposes. Center notch data for 90-degree unidirectional graphite-epoxy composites were also obtained as a basis for test method comparison.

TITLE: COMPOSITE DEFECT SIGNIFICANCE
S. Chatterjee, Materials Sciences Corporation and R. B. Pipes, University of Delaware

The state of the art of nondestructive and analytical evaluation of criticality of defects in composite laminates is reviewed. Methods based on concepts of Fracture Mechanics for modeling growth is isolated disbonds in laminated beams and plates under transverse shear are extended to consider the effects of multiple disbonds. Results from tests on disbonded laminates under transverse and inplane loads are correlated with analytical results. Disbond growth in presence of geometric discontinuities are also examined.

TITLE: SUPERPOSITION METHOD FOR ANALYSIS OF FREE EDGE STRESSES
J. Whitcomb and I. Raju, NASA Langley Research Center

Superposition principles were used to simplify free edge stress analysis by eliminating loads not contributing to the edge stresses. The technique is applicable to mechanical, thermal, and hygroscopic loads. A two-dimensional analysis is described for calculation of interlaminar normal stresses in quasi-isotropic laminates.

TITLE: MECHANICS OF DELAMINATION UNDER COMPRESSIVE LOADS
A. S. D. Wang, Drexel University

The delamination mechanisms in graphite-epoxy laminates subjected to compressive loads are studied. Experiments using AS-3501-06 material systems have been conducted under both static and fatigue compressive loadings. Delaminations caused by free edge interlaminar stress and by inplanted interlaminar flaws are monitored during the course of growth. An analytical fracture mechanics model is used to predict the initiation of delamination under static loading; and a fatigue growth model based on the same concept is also suggested.

TITLE: A CUMULATIVE DAMAGE MODEL FOR ADVANCED COMPOSITE MATERIALS

P. Chou, Dyna East Corporation

A mechanics model is proposed to explain the formation of transverse cracks in the 90-degree plies and the free edge delamination between plies in laminated composite materials under tension fatigue loading. The model is based on the assumption that the length of the inherent microcracks in the material has a random distribution. Under a given load, each initial crack is associated with a critical fatigue cycle that caused it to propagate. This critical cycle is calculated by fracture mechanics principles, using a finite-element program and the energy release rate approach. Results from the model are compared with experimental data. The calculated crack density (number of cracks per unit length) as a function of the fatigue cycle compares favorably with experimental measurement.

TITLE: PROPERTY DEGRADATION APPROACH TO CUMULATIVE DAMAGE MODELING OF ADVANCED COMPOSITES

W. Stinchcomb, K. Reifsnider, Virginia Polytechnic Institute and State University, and D. Ulman, General Dynamics/Fort Worth Division

A mechanistic cumulative damage model based upon experimentally observed damage modes was developed for advanced composite materials. The model uses the degradation of material properties of the individual lamina as well as the laminate with damage development. Principals of mechanics are applied to determine the stress redistribution associated with a given damage state. Stiffness change is used as a direct measurement of laminate property degradation and as a verification of the effects of the redistributed stress state. The residual strength of the laminate associated with a damage state can then be computed as a function of such variables as the redistributed 0-degree ply stress and interlaminar integrity (delamination). Stiffness retention and damage accumulation curves versus load history are used to generate residual strength and lifetime predictions.

TITLE: FATIGUE DAMAGE-STRENGTH RELATIONSHIPS IN COMPOSITE LAMINATES

K. Reifsnider, W. Stinchcomb, E. Henneke, II, J. Duke, Jr., and R. Jamison, Virginia Polytechnic Institute and State University

It is the general objective of this research program to determine the precise nature of the micro-events that degrade the integrity of composite laminates during cyclic loading, especially those events which are ultimately and directly associated with changes in the strength of the laminates. Long-term behavior has been the principal focus of our investigation since such situations correspond to cyclic load levels which are common to engineering applications, and strength reductions for finite life at such load amplitudes is very large, commonly 30-50 percent. Investigations to date have established the nature of the micro-events collectively called damage, and the states of damage that immediately precede the fracture event. Salient results include a surprising role played by internal stress redistribution due to local damage events and a history of fiber fracture events that is quite unexpectedly dependent on laminate type. The actual fracture event itself is also discussed.

TITLE: LAYUP AND FREQUENCY EFFECTS ON FATIGUE LIFE OF COMPOSITES
C. Saff, McDonnell Aircraft

One of the advantages of graphite/epoxy composites in fighter aircraft structures is its very long fatigue life. At current design strain levels, element fatigue test lives exceed design requirements by orders of magnitude. Because lives are so long, element tests are generally run at frequencies of about 10 Hz. These frequencies are significantly higher than those experienced in fighter aircraft structure, since maximum wing loads occur during high g maneuvers generally applied at frequencies of nearly 0.1 Hz. Tests of some graphite-epoxy laminates show that fatigue life can be proportional to cyclic frequency. In these cases, data from high frequency tests would not provide accurate assessment of life under service loadings.

The objectives of this program were to determine the load frequency sensitivity of layups typical of those used in aircraft structure and to develop analysis techniques which could be used to identify particularly sensitive layups so that their behavior could be accounted for in structural design and analysis.

Test results from this program show that load frequencies below 1 Hz produce a measurable reduction in graphite/epoxy fatigue life. This reduction is greatest in matrix dominated layups (less than 25 percent 0-degree fibers), but is still significant in fiber dominated layups.

The effect of load frequency on life under reversed load (R=1) constant amplitude fatigue is related to time at load. Trapezoidal waves, having long times at peak load, produce greater reduction in life than do sine wave loadings. Also, the effect of load frequency of life is greater at high stress levels.

At high frequencies, temperatures in graphite/epoxy laminates increase rapidly at holes. If measures are not taken to cool the specimens, fatigue life is adversely affected. This effect was alleviated in this program by using a cooling coil and vortex splitter refrigeration device controlled by a thermocouple located near the hole.

A method was developed for predicting the effect of load frequency, layup, load wave shape, and stress level on fatigue life of graphite/epoxy composites under reversed loadings. This method is based on the energy absorbed by the laminate per cycle and accounts for a time dependent change in gross strain found to occur under sustained loads.

TITLE: EFFECT OF STRESS RATIO ON FATIGUE LIFE OF COMPOSITES

G. P. Sendeckyj, AFWAL/Flight Dynamics Laboratory

A necessary step in the formulation of life analysis methods for composite materials is the development of a fatigue data representation that properly accounts for the stress ratio dependence. While many representations are possible, the best can only be developed based on analysis of experimental data. With this in mind, we conducted an experimental program to generate constant amplitude, tension-tension, fatigue data at different stress ratios for glass-epoxy and graphite-epoxy laminates.

General forms of the residual strength fatigue model were fitted to the data with the following results:

(a) The three-parameter residual strength model proposed by Yang, etc. does not provide a good representation of the data.

(b) A new four-parameter model provides a good representation of the data, but exhibits local maximum when using the maximum shape parameter model fitting procedure.

The derivation of the general residual strength degradation model will be presented. The procedure for testing various forms of the model against experimental data will be shown in detail.

TITLE: HIGH-LOAD TRANSFER JOINTS FOR WING STRUCTURES

S. Garbo and D. Buchanan, McDonnell Aircraft

The objective of this program was to demonstrate, through analysis and test, the potential of bolted metal-to-composite joints for high-load-transfer applications (20,000 to 30,000 lb/inch). These joints were designed to the structural requirements of the adhesive bonded step-lap joint currently found in the McDonnell Aircraft (MCAIR) F/A-18 Hornet wing root. Parametric studies were performed, evaluating such design variables as the thickness and width of the member, the number of fasteners and fastener spring rates, layup variations, local laminate softening, and hole tolerance. The effects of bypass and bearing load interactions were also considered.

Two designs were selected as a result of these trade studies. Both incorporated spanwise geometric tailoring of both composite and metal members, but one also used laminate "softening" strips in which 0-degree plies local to the fastener hole were replaced by ±45-degree plies.

Specimens of both designs were tested under tension-dominated static loads and spectrum fatigue cycling in a dry, room temperature environment. Both joints survived two lifetimes fatigue exposure with no change in mechanical properties. Predicted static strengths were within 5% of test data, and residual strength data were equal to or greater than static strength data.

The structural efficiency and manufacturing costs of both of these bolted designs were evaluated and compared to the current adhesive bonded step-lap joint. Weight studies indicated that step-lap bonded joints were structurally more efficient, but differences in relative cost were insignificant. Additional design options were also identified which could potentially reduce cost and weight.

Results were encouraging enough that a follow-on program was initiated to further develop high-load-transfer bolted composite joints for tension as well as compression loadings, with expanded structural verification, including full scale testing.

The completed work was conducted under Navy Contract N62269-80-C-0285 and contract activities are detailed in Navy Report NADC-81194-60. The current work is being conducted under Navy Contract N62269-82-C-0238.

TITLE: DESIGN METHODOLOGY FOR BONDED-BOLTED COMPOSITE JOINTS
J. Hart-Smith, McDonnell Douglas Corporation

This report contains recent developments in three aspects of joints in advanced fibrous composite structures: (1) nonlinear analysis of adhesively-bonded stepped-lap joints and doublers, (2) multirow mechanically fastened joints in aerospace structures, and (3) nonlinear analysis of combined bonded and bolted joints. The methods developed include nonlinearities needed for metal structures as well as those for composites. The analyses are based on continum mechanics techniques and have been coded into three Fortran IV digital computer programs A4EI, A4EJ, and A4EK respectively. The report contains explanations of the derivations of the solutions as well as sample worked problems to illustrate both the capabilities of the programs and the characteristic behavior of the real structures.

TITLE: INTERPLY LAYER PROGRESSIVE WEAKENING EFFECTS ON COMPOSITE STRUCTURAL RESPONSE
C. Chamis, NASA Lewis Research Center

The presentation summarizes recent research activities at Lewis to computationally determine and assess the effects of interply layer progressive weakening (degradation) on the structural response of a layered composite beam. The structural response of interest includes: (1) bending; (2) buckling; (3) free vibrations; (4) periodic excitation; and (5) impact. Finite elements analysis was used for the computational method. The interply layer degradation effects on the various structural responses were determined and assessed as a function of the interply layer modulus varying from one million psi down to 1000 psi and even lower for some limiting cases. The results obtained show that the interply layer degradation has negligible effect on composite structural integrity for interply layer modulus greater than about 10,000 psi.

TITLE: RESEARCH ON COMPOSITE MATERIALS FOR STRUCTURAL DESIGN
R. Schapery, Texas A&M University

Research on composites at Texas A&M University sponsored by the Air Force Office of Scientific Research is reviewed. Much of the effort is concerned with several student/faculty research projects. The following five studies are nearing completion or are completed, and they are reviewed: "Effect of Resin Toughness on Fracture Behavior of Graphite/Epoxy Composites", (Cohen/Bradley): "Slow, Stable Delamination in Graphite/Epoxy Composites", (Razi/Schapery); "Evaluation of Energy Release Rates in Unidirectional Split Laminate Specimens", (Weatherby/Schapery); "Analysis of the Effect of Matrix Degradation on Fatigue Behavior of a Graphite/Epoxy Laminate", (Arenburg/Schapery); "On the Effects of Post Cure Cool Down and Environmental Conditioning on Residual Stresses in Composite Laminates", (Harper/Weitsman).

TITLE: IMPERFECTION SENSITIVITY OF FIBER-REINFORCED, COMPOSITE, THIN CYLINDERS
G. Simitses, D. Shaw and I. Sheinman, Georgia Institute of Technology

The imperfection sensitivity of thin cylindrical shells made out of fiber-reinforced composite material and subjected to uniform axial compression is investigated. The methodology is based on linear constitutive relations, nonlinear kinematic shell equations (Donnell-type) and the usual laminated theory. The laminate consists of orthotropic laminae, stacked in a general manner (asymmetric laminate). The uniform axial compression is applied eccentrically, and the geometrically imperfect cylindrical shell can be supported in various ways at the boundaries. In this investigation a number of parametric studies are performed. The scope of these studies is to establish the effect of (a) in-plane and transverse boundary conditions and (b) load eccentricity on the critical load of a typical Boron/Epoxy laminate with various stacking sequences of laminae.

TITLE: RESEARCH INTO THE DESIGN TECHNOLOGY OF ADVANCED COMPOSITES
P. Lagace and J. Dugundji, Massachusetts Institute of Technology

Two areas of current research will be described: (a) Damage Tolerance: The fracture behavior of pressurized graphite/epoxy cylinders was investigated. The cylinders were 582 mm long and 305 mm in diameter and were manufactured from Hercules A370-5H/3501-6 prepreg cloth in 4-ply configurations: $(0,45)_s$ and $(45,0)_s$. The fracture point of the cylinders is well predicted from flat coupon data corrected for the effects of curvature. In addition, circumferentially wrapped unidirectional plies of Hercules AS1/3501-6 tape of various stacking sequences were used as selective reinforcement on several $(0,45)_s$ cylinders. These reinforcing plies did change the path of damage propagation but did not prevent catastrophic failure. (b) Aeroelasticity: Aeroelastic flutter and divergence of graphite/epoxy, cantilevered plates with various amounts of bending-torsion stiffness coupling was investigated for incompressible flow, at both 0^o and 30^o forward sweep. Wind tunnel tests indicated divergence, bending-torsion flutter, and stall flutter. Tests agreed reasonably with theory at low angels of attack.

TITLE: NONLINEAR TRANSIENT ANALYSIS OF COMPOSITE PLATES:
J. Reddy, Virginia Polytechnic Institute and State University

Forced motions of laminated, anisotropic composite plates are investigated using laminated plate theory that account for geometric nonlinearity in the von Karman sense. The finite element method is used to solve the equations numerically. For two different lamination schemes, under appropriate edge conditions and sinusiodal distribution of the transverse load, the exact form of the spatial variation of the linear solution is obtained to validate the finite element analysis. The effect of edge conditions, lamination schemes, and plate side-to-thickness ratio on the global response of composite plates is investigated. The finite element results are found to agree very closely with the analytical results. It is also found that the transient response of composite plates is very sensitive to edge conditions, plate side-to-thickness ratio, lamination schemes, and geometric nonlinearity. Current research in the transient analysis of composite plates and shells is also reviewed.

TITLE: IMPROVED CERAMIC FRACTURE BEHAVIOR FOR HIGH TEMPERATURE TURBINE APPLICATIONS
K. Buesking, S. Chatterjee, B. Rosen, Materials Sciences Corporation

Several analytical models for assessing the fracture toughness and/or tensile strength of a whisker reinforced ceramic are identified. One model, referred to as the tortuous path approach, relates the material fracture toughness to the increased energy required for a crack to change directions as it grows around whiskers. Another model, identified as strength versus size, relates increases in tensile strength to statistical parameters which describe the size of local unreinforced matrix elements. The third approach, known as the critical matrix stress, bases tensile failure on the average stress state in the matrix phase of the composite.

The models will be tested for feasibility by comparing theoretical predictions to measured data from mechanical tests of whisker reinforced ceramics. Based upon the comparisons, a particular theory will be chosen for development into a meterial design aid.

APPENDIX B

PROGRAM LISTINGS

AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
MATERIALS LABORATORY

INHOUSE

ADVANCED COMPOSITES
WORK UNIT DIRECTIVE (WUD) NUMBER 45
77 April - 84 April

WUD Leader: James M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Objective: The objective of the current thrust under this work is to develop and demonstrate concepts of damage resistance as applied to fiber reinforced composite laminates. Short term objectives (1-3 yrs) include the following:

(a) Development of failure mode models with emphasis on delamination and matrix cracking.
(b) Assess the role of matrix toughness in composite failure processes.
(c) To develop concepts of interface/interphase strengthening.

CONTRACTS

IMPROVED MATERIALS FOR COMPOSITES AND ADHESIVE JOINTS
F33615-81-C-5056
1 Sept 81 to 31 Aug 84

Project Engineer: James M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Principal Investigator: Ran Y. Kim
University of Dayton Research Institute
300 College Park Avenue
Dayton, Ohio 45469

Objective: To investigate from both an experimental and analytical standpoint the potential of new and/or modifications of existing materials and reinforcement for use in advanced composite materials and adhesive bonded joints. Such materials are subsequent candidates for use in advanced aircraft and aerospace structural applications.

DAMAGE TOLERANT COMPOSITE LAMINATES
F33615-81-C-5050
81 June - 85 September

Project Engineer: Stephen W. Tsai
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-3068 Autovon: 785-3068

Principal Investigator: George S. Springer
Department of Mechanical Engineering
University of Michigan
550 E. University
Ann Arbor, MI 48109

Objective: To develop design and processing techniques in order to enhance damage tolerance of composite laminates.

CUMULATIVE DAMAGE MODEL FOR COMPOSITE MATERIALS
F33615-80-C-5039
81 Feb 23 - 82 Aug 24

Project Engineer: Marvin Knight
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-7131 Autovon: 785-7131

Principal Investigator: P. C. Chou
Dyna East Corporation
227 Hemlock Road
Wynnewood, PA 19096
(215) 895-2288

Objective: This program will develop a methodology, including analytical modeling, for predicting and experimentally characterizing advanced composite materials' mechanical responses to defined load histories. A cumulative damage model is the ultimate goal.

CUMULATIVE DAMAGE MODEL FOR COMPOSITE MATERIALS
F33615-81-C-5049
81 Feb 23 - 82 Aug 24

Project Engineer: Marvin Knight
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-7131 Autovon: 785-7131

Principal Investigator: J. Masters
General Dynamics Corporation
Fort Worth Division
P.O. Box 748
Fort Worth, TX 76101
(817) 732-4811 Ext 5375

Objective: This program will develop a methodology, including analytical modeling, for predicting and experimentally characterizing advanced composite materials' mechanical responses to defined load histories. A cumulative damage model is the ultimate goal.

FLIGHT DYNAMICS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES

IN-HOUSE

STRUCTURAL INTEGRITY RESEARCH FOR ENGINES AND AIRFRAMES
JON: 2307N101
77 January 1 - 83 March 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Objective: To resolve theoretical questions and develop damage tolerance and life analysis methods which can be used to satisfy the requirements of MIL-STD-1530A for advanced composite and metallic airframe structures. The specific objectives in the composites area are:

(a) develop an understanding of the fatigue damage accumulation process, develop a fatigue model that takes into account all pertinent variables, and verify the fatigue model using experimental data; use the experimental data to assess the effect of fabrication variability and determine the effect of the percentage of zero degree plies on fatigue behavior;

(b) perform a critical state-of-the-art assessment of the literature on damage tolerance and effect-of-defects on the behavior of composite materials;

(c) develop a progressive-ply-failure finite element procedure for predicting the damage accumulation process in composite materials;

(d) develop analytical models to determine the ultimate strength and fatigue life of composite hat stiffeners under compressive loading;

(e) evaluate the effects of periodic underloads/overloads on the fatigue behavior of fiber-reinforced metal-matrix composites;

(f) conduct an indepth examination of current acoustical fatigue test techniques; and

(h) verify the nonlinear single mode analytical results with structural response of simple rectangular graphite-epoxy panels excited by broadband acoustic excitation at both low and high levels.

ACOUSTIC FATIGUE DESIGN OF ADVANCED STRUCTURES
JON: 24010146
82 February 3 - 85 February 2

Project Engineer: Howard F. Wolfe

Air Force Wright Aeronautical Laboratories
AFWAL/FIBED
Wright-Patterson AFB, Ohio 45433
(513) 255-5753 Autovon 785-5753

Objective: Develop methods for the prediction of the acoustical fatigue life of graphite-epoxy skin stringer structures. The investigation will include adhesive bonded and co-cured skin stringer beams and acoustic panels to be tested on a vibration shaker and in a progressive wave tube to obtain dynamic properties and fatigue life.

AEROELASTICITY RESEARCH
JON: 24010239
79 September 3 - 83 March 30

Project Engineer: Maxwell Blair
Air Force Wright Aeronautical Laboratories
AFWAL/FIBRC
Wright-Patterson AFB, Ohio 45433
(513) 255-7384 Autovon 785-7384

Objective: Aeroelastic research in several areas is conducted under this work unit. Three principle areas of research are aeroelastic tailoring, active flutter suppression and unsteady aerodynamics. In the area of aeroelastic tailoring the main emphasis has been on Forward Swept Wing research. This research includes wind tunnel testing of composite structures for flutter and divergence, combined active divergence/flutter suppression analysis of tailored structures, and full scale analysis of the tailored X-29A FSW Demonstrator for aeroelastic stability. Future wind tunnel tests are planned to investigate the effect of stores on divergence and flutter of FSW configurations.

ANALYSIS AND OPTIMIZATION OF AEROSPACE STRUCTURES
JON: 24010244
80 March 10 - 33 March 30

Project Engineer: Dr. V. B. Venkayya
Air Force Wright Aeronautical Laboratories
AFWAL/FIBR
Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Objective: The key to the successful design of lighter and more reliable airframe structures is the ability to accurately predict structural response and to make rapid sensitivity analyses with parametric changes. The sensitivity analysis in turn is the important element in the evolution of dependable and cost effective structures. The objective of the effort is to develop computational tools for rapid analysis and optimization of metallic and composite aerospace structures.

STRUCTURAL TESTING OF COMPOSITE PANELS
JON: 24010246
80 April 28 - 83 June 30

Project Engineer: Dr. N. S. Khot
Air Force Wright Aeronautical Laboratories
AFWAL/FIBR
Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Objective: Develop experimental methods and to conduct tests to determine the buckling and postbuckling strength of stiffened and unstiffened composite panels.

COMPRESSIVE TEST FIXTURE EVALUATION
JON: 24010344
79 January 1 - 82 December 1

Project Engineer: Rick Rolfes
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCC
Wright-Patterson AFB, Ohio 45433
(513) 255-6658 Autovon 785-6658

Objective: To evaluate various compression test fixtures currently in use be industry, together with an in-house prototype design. Efforts will focus on (a) elimination of the predominate brooming and buckling failure modes associated with present test fixtures, (b) 0 degreee compressive strengths analogous to 0 degree tensile strengths, and (c) a reduction in costs of test specimen fabrication.

REAPIR OF GRAPHITE/EPOXY COMPOSITES
JON: 24010344
82 September 1 - 84 September 1

Project Engineer: Forrest Sandow
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCB
Wright-Patterson AFB, Ohio 45433
(513) 255-2582 Autovon 785-2582

Objective: At present the development of repair procedures for graphite-epoxy composites is generally baseo on the testing of simulated repairs which are large and relatively costly to produce and evaluate. This problem should be overcome by use of elemental joint specimens representing a section through the repaired region. Detailed studies of the stress-strain behavior of these specimens, under the appropriate variables, will provide basic information for design of repairs.

REPAIR OF V378A COMPOSITES
JON: 24010344
82 September 1 - 84 September 1

Project Engineer: Forrest Sandow
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCB
Wright-Patterson AFB, Ohio 45433
(513) 255-2542 Autovon 785-2582

Objective: To fabricate a series of composite panels using V378A polyimide material for the matrix material. Panels will be cut into coupons for testing before ballistic impact, after ballistic damage, and after repair.

HYDRODYNAMIC RAM ASSESSMENT OF INTEGRAL SKIN/SPAR DESIGNS
JON: 24010349
80 March 24 - 82 December 30

Project Engineer: S. D. Thompson
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Objective: To study the effect of hydrodynamic ram caused by ballistic penetration on advanced composite structures and to evaluate the relative susceptibility of several integral composite skin/spar concepts. This will provide designers with information necessary to allow transition of composites technology to operational aircraft.

ASSESSMENT OF CORROSION CONTROL PROTECTIVE COATINGS
JON: 24010350
80 April 28 - 85 May 1

Project Engineer: S. D. Thompson
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Objective: To determine the susceptibility of graphite/epoxy-aluminum joints to corrosion when protective coatings, that have undergone fatigue loading, are used. The knowledge gained will be used to determine if present corrosion control systems actually prevent corrosion and if not, how they could be modified to prevent corrosion from occurring.

COMPOSITE IMPACT STUDY
JON: 22510115
80 September 1 - 84 September 30

Project Engineer: James M. Remar
Air Force Wright Aeronautical Laboratories
AFWAL/FIES
Wright-Patterson AFB, Ohio 45433
(513) 255-6302 Autovon 785-6302

Objective: To investigate characteristics of advanced filamentary composites impacted by single fragment projectiles. This program will develop basic core of data to predict penetration characteristics useful in preliminary design analysis.

GRANTS

DURABILITY OF REPEATEDLY BUCKLED PANELS
Grant AFOSR 81-0016 JON: 2307N114
80 May 12 - 83 September 30

Project Engineer: Dr. V. B. Venkayya
Air Force Wright Aeronautical Laboratories
AFWAL/FIBRA
Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Principal Investigator: Dr. Josef Singer
Department of Aeronautical Engineering
Technion - Israel Institute of Technology
Haifa, 32000
ISRAEL

Objective: Metallic and composite shear panels in aircraft structures are generally designed to operate in the post-buckled range to reduce structural weight. The objective of this effort is to study the durability of repeatedly buckled panels and to provide guidelines for the design of flat and curved shear panels with various types of stiffener configurations.

CONTRACTS

TEST SYSTEM FOR CONDUCTING BIAXIAL TESTS OF COMPOSITE LAMINATES
Contract F33615-77-C-3014 JON: 2307N103
77 September 19 - 82 September 20

Project Engineer: T. N. Bernstein
Air Force Aeronautical Laboratories
AFWAL/FIBRA

Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Principal Investigator: Scott W. Schramm
IIT Research Institute
10 West 35th Street
Chicago, Illinois 60616
(312) 567-4000

Objective: To develop, design and fabricate a biaxial test machine capable of applying, withour constrains, in-plane loads, singly and in any combination, to laminated tubular composite specimens.

A STUDY OF THE BUCKLING, POST-BUCKLING BEHAVIOR AND VIBRATION OF LAMINATED COMPOSITE PLATES
Contract F33615-81-K-3203 JON: 2307N115
80 November 20 - 83 November 20

Project Engineer: Dr. N. S. Khot
Air Force Wright Aeronautical Laboratories
AFWAL/FIBRA
Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Principal Investigator: Professor Arthur Leissa
Department of Engineering Mechanics
Boyd Laboratory
Ohio State University
155 Wset Woodruff Avenue
Columbus, Ohio 43210

Objective: To prepare a monograph summarizing the state of the art in buckling, post-buckling and vibration behavior of laminated composite plates.

FATIGUE DAMAGE-STRENGTH RELATIONSHIPS IN COMPOSITE MATERIALS
Contract F33615-81-K-3225 JON: 2307N117
80 December 12 - 83 September 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronatical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Professor K. L. Reifsnider
Engineering Science & Mechanics Department
Virginia Polytechnic Institute & State University
Blacksburg, Virginia 24061
(703) 961-5316

Objective: Develop an understanding og the initiation of fiber fractures from matrix cracks and delaminations in resin-matrix composites.

ENVIRONMENTAL TRACKING OF F-15 HORIZONTAL STABILATOR
Contract F33615-79-C-3210 JON: 24010132
79 June 15 - 83 October 1

Project Engineer: Carl L. Rupert
Air Force Wright Aeronautical Laboratories
AFWAL/FIBED
Wright-Patterson AFB, Ohio 45433
(513) 255-5753 Autovon 785-5753

Principal Investigator: Thomas V. Hinkle
McDonnell Douglas Corporation
P. O. Box 516
St. Louis, Missouri 63166
(314) 232-3356

Objective: To evaluate the effects of additional exposure to a service environment on the F-15 boron-epoxy stabilator.

FATIGUE/IMPACT STUDIES IN LAMINATED COMPOSITES
Contract F33615-80-K-3243 JON: 24010152
80 May 12 - 83 December 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Dr. V. Sarma Avva
Mechanical Engineering Department
North Carolina A&T State University
Greensboro, North Carolina 27411
(919) 379-7620

Objective: To systematically document the fatigue induced damage accumulation process in impact damaged structural composite laminates.

DESIGN METHODOLOGY AND LIFE ANALYSIS OF POSTBUCKLED METAL AND COMPOSITE PANELS
Contract F33615-81-C-3208 JON: 24010154
80 June 23 - 84 August 30

Project Engineer: Capt. M. L. Becker
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC

Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Dr. Ben Agarwal
Northrop Corporation
Structural Mechanics Research
3901 West Broadway
Hawthorne, California 90250
(213) 970-5075

Objective: Develop analytical techniques and design procedures for metal and composite aircraft structures operating in the postbuckled range.

DAMAGE ACCUMULATION IN COMPOSITES
Contract F33615-81-C-3226 JON: 24010157
80 August 18 - 84 December 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: David A. Ulman
Structures & Design Department
General Dynamics Corporation
P. O. Box 748
Fort Worth, Texas 76101
(817) 732-4811 ext 4179

Objective: Develop a state-of-damage based procedure for predicting the life of composite structures subjected to spectrum fatigue loading.

DESIGN VERIFICATION FOR OPTIMIZED PANELS
Contract F33615-81-C-3222 JON: 24010248
81 September 15 - 84 July 15

Project Engineer: Dr. N. S. Khot
Air Force Wright Aeronautical Laboratories
AFWAL/FIBRA
Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Principal Investigator: Mr. Bo Almroth
Lockheed Palo Alto Research Laboratory
Bldg 255
3251 Hanover
Palo Alto, California 94304
(415) 858-4027

Objective: To experimentally investigate the behavior of optimized composite stiffened panels.

BOLTED JOINTS IN COMPOSITE STRUCTURES: DESIGN, ANALYSIS AND VERIFICATION
Contract F33615-82-C- JON: 24010255
82 September 15 - 86 July 31

Project Engineer: Dr. V. B. Venkayya
Air Force Wright Aeronautical Laboratories
AFWAL/FIBRA
Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Principal Investigator: To be announced

Objective: To develop reliable analytical and experimental methods for strength and life prediction of multi-fastener joints in full-scale composite structures. The end products will be a new or revised computer program for analysis and a design guide for representative aircraft joints.

DOD/NASA ADVANCED COMPOSITES DESIGN GUIDE
Contract F33615-78-C-3203 JON: 24010324
78 March 1 - 82 November 1

Project Engineer: B. White
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Principal Investigator: G. Howard Arvin
Rockwell International Corporation
LA Aircraft Division
5701 W. Imperial Highway
Los Angeles, California 90009
(213) 670-9151 ext 1666

Objective: To develop a new, updated version of the "Advanced Composites Design Guide." The new version will incorporate new data and analysis techniques. The guide will be reorganized and condensed to make it a more useful document to designers.

SURVIVABLE COMPOSITE FUEL TANK STRUCTURES
Contract F33615-82-C-3212 JON: 24010357
82 August 2 - 85 November 30

Project Engineer: S. D. Thompson

Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Principal Investigator: Dr. M. J. Jacobson
Northrop Corporation
Aircraft Services Division
One Northrop Avenue
Hawthorne, California 90250
(213) 970-2000

Objective: Develop guidelines for the design of composite integral fuel tanks capable of surviving hostile environments created by non-detonating projectiles and warhead fragments.

COMPOSITE WING/FUSELAGE PROGRAM
Contract F33615-79-C-3203 JON: 69CW0152
79 July 1 - 84 July 30

Project Engineer: James L. Mullineaux
Air Force Wright Aeronautical Laboratories
AFWAL/FIBAC
Wright-Patterson AFB, Ohio 45433
(513) 255-6639 Autovon 785-6639

Principal Investigator: J. J. Eves, Program Manager
Northrop Corporation/Aircraft Division
One Northrop Avenue
Hawthorne, California 90250

Objective: To develop structural design technology and durability qualification methodology for application of advanced composites to wing and fuselage primary structures of Mach 2 class fighter aircraft. Secondary efforts within the program will verify low cost fabrication methods, develop quality assurance techniques and evaluate the effects of defects in composites.

AIR FORCE OFFICE OF SCIENTIFIC RESEARCH

INHOUSE

NONE

CONTRACTS

BOUNDARY ELEMENTS FOR DEBOND STRESS ANALYSIS
82 March 01 - 83 February 28

Project Engineer: Capt David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Colin Atkinson
Dept of Mathematics
Imperial College of Science and Technology
London SW7 2BZ England

Objective: To develop a boundary integral equation method valid for short crack initiation at the fiber-matrix interface in composite materials.

FRACTURE BEHAVIOR OF BORON ALUMINUM COMPOSITES
79 April 01 - 83 October 14

Project Engineer: Capt David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Jonathan Awerbuch
Dept of Mechanical Engr and Mechanics
Drexel University
Philadelphia, PA 19104
(215) 895-2291

Objective: To provide insight into the fracture mechanisms in boron aluminum composites at room and elevated temperatures through a comprehensive experimental program and correlation of test data with analytical predictions.

IMPROVED CERAMIC FRACTURE BEHAVIOR FOR HIGH TEMPERATURE TURBINE APPLICATIONS
82 April 01 - 82 September 30

Project Engineer: Capt David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Mr Kent W Buesking
Materials Science Corporation
Blue Bell Office Campus, Merion Towle House
Blue Bell, PA 19422
(215) 542-8400

Objective: To identify the failure modes of fiber-reinforced ceramics and establish the theoretical basis for the development of analytical models capable of predicting these modes.

DAMAGE ESTIMATION IN CARBON FIBRE REINFORCED EPOXY AND ITS INFLUENCE ON RESIDUAL PROPERTIES
82 June 15 - 83 June 14

Project Engineer: Capt David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr A R Bunsell
Ecole Nationale Superieure des Mines de Paris
Centre des Materiaux
BP 87
91003 EVRY cedex
France

Objective: To investigate the failure of fibers and the subsequent accumulation of damage in unidirectional carbon fiber reinforced plastics (cfrp) by using the acoustic amission technique, and to extend a recently developed and verified theory of damage accumulation to unidirectional cfrp subjected to cyclic loading.

ANALYSIS OF DAMAGE PROCESSES IN FIBROUS COMPOSITE LAMINATES
82 September 01 - 83 August 31

Project Engineer: Capt David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr George J Dvorak
Dept of Civil Engineering
University of Utah
Salt Lake City, UT 84112
(801) 581-6931

Objective: To conduct a theoretical study of damage accumulation in unnotched fibrous composite laminates caused by distributed internal cracking in individual layers as well as delamination cracks between layers, under monotonic or cyclic mechanical and thermal loads.

THREE-DIMENSIONAL ANISOTROPIC STRESS CONCENTRATIONS
81 December 01 - 82 November 30

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr R A Eubanks
Dept of Civil Engineering
University of Illinois
Champaign, IL 61820
(217) 333-6946

Objective: To develop rigorous analytical methods for three-dimensional stress concentrations in transversely isotropic materials such as advanced composites or other reinforced or layered materials.

FRACTURE, FATIGUE, DYNAMICS, AND AEROELASTICITY OF COMPOSITE STRUCTURES
82 January 01 - 82 December 31

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr James W Mar
Dept of Aeronautics and Astronautics
Massachusetts Institute of Technology
Cambridge, MA 02139
(617) 253-2426

Objective: To evaluate parameters in semi-empirically derived fracture relationships using an extensive experimental data base on fatigue and fracture of composite laminates, to investigate frequency and modal behavior of cantilever and clamped-clamped plates of unbalanced laminate construction with emphasis on assessing the nonlinear behavior due to large deformations and multi-axis response coupling, and to investigate aeroelastic flutter and divergence of unbalanced laminates.

NONLINEAR DYNAMIC RESPONSE OF COMPOSITE ROTOR BLADES
82 September 01 - 83 August 31

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Ozden Ochoa
Dept of Mechanical Engineering
Texas A&M Research Foundation
College Station, TX 77843
(713) 845-2022

Objective: To develop nonlinear displacement and damage models suitable for predicting the structural dynamic response of composite rotor blades to impact and other transient excitations.

NONLINEAR TRANSIENT ANALYSIS OF LAYERED COMPOSITE PLATES AND SHELLS
81 April 01 - 83 June 15

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator Dr J N Reddy
Dept of Engineering Science & Mechanics
Virginia Polytechnic Inst & State University
Blacksburg, VA 24061
(703) 961-6744

Objective: To evaluate the stability and convergence characteristics of penalty-finite elements applied to the dynamic analysis (e.g. low velocity impact) of composite plates and shells, and to evolve a transient analysis capability with greatly improved accuracy, numerical stability and computational efficiency.

BEHAVIOR OF ADVANCED AND COMPOSITE STRUCTURES
82 January 01 - 82 December 31

Project Engineer: Capt David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-04937

Principal Investigator: Dr Lawrence W Rehfield
Dept of Aerospace Engineering
Georgia Institute of Technology
Atlanta, GA 30332
(404) 894-3067

Objective: To develop new theories for the analysis of bending behavior of laminated beams and plates, to experimentally investigate delamination of composite laminates under compressive loading, and to experimentally evaluate the damage tolerance of continuous filament composite isogrid structure.

RESEARCH ON COMPOSITE MATERIALS FOR STRUCTURAL DESIGN
82 January 01 - 83 December 31

Project Engineer: Capt David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Richard A Schapery
Dept of Civil Engineering
Texas A&M University
College Station, TX 77843
(713) 845-7512

Objective: To experimentally identify, study in detail, and model analytically the basic mechanisms of structural response of resin matrix composite materials including studies of micro- and macro-mechanisms of fracture, effects of transient temperature and moisture content, behavior and structure of water in polymers, toughening mechanisms in resins, and theoretical models for deformation and fracture behavior.

NONLINEAR LARGE DEFORMATION BEHAVIOR OF COMPOSITE CLYINDRICAL SHELLS
81 June 30 - 83 June 29

Project Engineer: Dr Anthony K Amos
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr George J Simitses
Dept of Engineering Science & Mechanics
Georgia Institute of Technology
Atlanta, GA 30332
(404) 894-2770

Objective: To develop nonlinear solution methodology for the response characteristics of stiffened laminated cylindrical shells, including pre-limit point and post-limit point behavior, and to use tne methodology to study nonlinear phenomena in such shell structures.

SPACE ENVIRONMENTAL EFFECTS ON POLYMER MATRIX COMPOSITES
78 September 01 - 82 August 31

Project Engineer: Capt David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Rodney C Tennyson
Institute for Aerospace Studies
University of Toronto
Downsview, Ontario, Canada M3H 5T6
(416) 667-7710

Objective: To identify the influence of simulated and in-situ space environmental conditions on the mechanical characteristics of advanced composite materials in real time, and to evaluate accelerated environmental exposure techniques.

INTERLAMINAR AND INTRALAMINAR FRACTURE GROWTH IN COMPOSITE MATERIALS
79 September 01 - 83 September 30

Project Engineer: Capt David A Glasgow
AFOSR/NA
Bolling AFB, DC 20332
(202) 767-4937

Principal Investigator: Dr Albert S D Wang
Dept of Mechanical Engineering
Drexel University
Philadelphia, PA 19104
(215) 895-2297

Objective: To develop qualitative understanding of and analytical/computational prediction capability for fracture initiation and propagation processes in composite laminates.

NASA LANGLEY RESEARCH CENTER

INHOUSE

EFFECT OF FOIL TOUGHENING ON IMPACT RESISTANCE OF LAMINATES
81 May 1 - 82 November 30

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2338 FTS 928-2338

Objective: To determine the effect on impact resistance of partial interlaminar separations between layers of a laminate. Perforated mylar foil produces the partial separations.

MECHANICS OF LOW-VELOCITY IMPACT
81 June 1 - 83 June 30

Project Engineer: Dr. Wolf Elber
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2093 FTS 928-2093

Objective: From quasi-static deformation analysis, determine the criteria for low-velocity impact damage; establish threshold levels for impact damage. Develop fracture mechanics analyses for delamination growth and membrane failure.

ASSESSING THE ROLE OF SHOCK WAVES IN IMPACT DAMAGE
81 April 1 - 83 March 31

Project Engineer: Dr. Joseph S. Heyman
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3418 FTS 928-3418

Objective: To examine impact damage reduction to composites through acoustic impedance matching techniques and to assess the role shock waves play in low-velocity impact-induced material degradation.

SHOCK WAVE SPECTRAL ENERGY ANALYSIS
81 July 1 - 83 June 30

Project Engineer: Dr. William P. Winfree
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3418 FTS 928-3418

Objective: To develop systems for the study of the energy frequency spectra of impact-induced shock waves in graphite/epoxy composites and related materials to determine damage mechanisms.

TOUGHNESS TEST METHODOLOGY
80 October 1 - 83 September 30

Project Engineer: Dr. Norman J. Johnston
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3041 FTS 928-3041

Objective: To investigate, develop (if necessary), and select appropriate test methods for screening the impact resistance and fracture toughness properties of neat polymers and composites. Methodology will help guide programs to synthesize new toughened matrix resins.

FRACTURE OF LAMINATED COUPONS
78 October 1 - 83 September 30

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2338 FTS 928-2338

Objective: To develop a methodology to predict residual strengths of damaged composite laminates using, as starting points, lamina properties or possibly the properties of the fibers and matrix. To determine the parameters that lead to tough composites.

DAMAGE TOLERANT COMPOSITE STRUCTURES
74 June 1 - 83 May 31

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2338 FTS 928-2338

Objective: To measure the ability of buffer strips and bonded stringers to increase the residual tension strength of damaged panels, and to develop an analysis to predict residual strength in terms of panel configuration and damage size.

EFFECT OF ELEVATED TEMPERATURE ON LARGE GRAPHITE/POLYIMIDE BUFFER STRIP PANELS
81 February 11 - 83 May 31

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2338 928-2338

Objective: To experimentally determine the effect of elevated temperature on the fracture behavior of large graphite/polyimide buffer strip panels with various size buffer strips.

EFFECT OF MOISTURE AND ELEVATED TEMPERATURE ON GRAPHITE/EPOXY BUFFER STRIP PANELS
80 November 1 - 83 May 31

Project Engineer: C. A. Bigelow
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3191 FTS 928-3191

Objective: To experimentally determine the effect of moisture and elevated temperature on the fatigue life of graphite/epoxy buffer strip panels.

WOVEN COMPOSITE BUFFER STRIP PANELS
81 January 1 - 83 May 31

Project Engineer: John M. Kennedy
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3191 FTS 928-3191

Objective: To demonstrate that buffer strip panels built with woven cloth have the crack-arresting capability of panels built with conventional prepreg tape. Damaged panels will be tested in shear and tension.

STRESS ANALYSIS OF LOADED HOLES
82 March 1 - 83 March 1

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3012 FTS 928-3012

Objective: To calculate stresses near loaded holes in finite-size laminates with tensile and compressive applied loads.

FAILURE ANALYSIS OF LOADED HOLES
81 April 1 - 84 April 1

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3012 FTS 928-3012

Objective: To develop the basic understanding needed for an analytical procedure to predict bolt hole failure under combined bearing and bypass loads.

ADHESIVE DEBOND CHARACTERIZATION
76 October 1 - 86 September 30

Project Engineers: Dr. W. S. Johnson
R. A. Everett, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2715 FTS 928-2715

Objective: To verify that identical specimens manufactured at different facilities using the same adhesive/adherent (7075 Al/FM-73) bonding techniques behave in a similar manner when subjected to cyclic loading. To develop an approach to calculate cyclic debond threshold and rate such that the cyclic behavior of the bondline can be predicted for any geometry (using finite elements) for a given adhesive/adherent system. To expand from metal-to-metal to composite-to-composite bonds and to examine temperature, moisture, and spectrum loading effects.

STRESS ANALYSIS OF ADHESIVE BONDS
80 October 1 - 84 September 30

Project Engineers: R. A. Everett, Jr.
J. D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2715 FTS 928-2715

Objective: To review currently available finite-element routines and their applicability to the adhesive bondline stress analysis. To modify available model or develop a new model to assess $\mathcal{G}_I$ and $\mathcal{G}_{II}$ at debond front, and to incorporate into model material and geometric nonlinear behavior.

FAILURE MODES OF ADHESIVELY BONDED COMPOSITE JOINTS
81 June 1 - 83 September 30

Project Engineers: Dr. S. Mall
Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2715 FTS 928-2715

Objective: To conduct experimental tests to determine the failure modes and mechanisms of adhesively bonded composite joints. To assess secondary bonding versus co-curing in graphite/epoxy and Kevlar/epoxy joints. Correlate debond growth rates with strain-energy-release rates. Establish design guideline for adhesively bonded composite joints.

REALISTIC ADHESIVELY BONDED JOINT ELEMENT
81 October 1 - 83 September 30

Project Engineer: R. A. Everett, Jr.
Mail Stop 188E
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2715 FTS 928-2715

Objective: To manufacture several variations of a simple adhesively bonded wing splice joint under contract (metal-to-composite specimens). To determine fatigue and fracture failure modes for a "realistic" aircraft adhesively bonded structure. These joints will consist of titanium wing ribs embedded in graphite/epoxy wing skins.

FATIGUE AND FRACTURE BEHAVIOR OF THICK LAMINATES
81 October 1 - 83 September 30

Project Engineers: Edward P. Phillips
C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3192 FTS 928-3192

Objective: To identify potential fatigue and fracture problems associated with scale-up of graphite/epoxy laminates to thicknesses of about 100 plies. This study will consist mostly of tests of thick laminates containing through-thickness holes and slits.

PREDICTION OF FATIGUE LIFE OF NOTCHED COMPOSITE LAMINATES
73 June 1 - 85 September 30

Project Engineers: Dr. T. Kevin O'Brien
John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Objective: To develop a method to design fatigue resistant composite laminates. The method addresses three areas: failure mechanisms are identified; analyses to predict inplane and interlaminar damage growth are developed; and inplane and interlaminar data bases are developed to evaluate the methodology.

THE EFFECTS OF REALISTIC FLIGHT ENVIRONMENTS ON FATIGUE OF COMPOSITE MATERIALS
72 June 1 - 84 May 31

Project Engineer: John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Objective: To determine the effects of realistic environments on the fatigue behavior of composite materials. Flight environments of conventional and supersonic aircraft transports are being investigated. Tests are either accelerated or conducted in real time. Temperatures and load spectra are simulated for transport environments.

PREDICTION OF INSTABILITY-RELATED DELAMINATION GROWTH
79 January 2 - 83 December 31

Project Engineer: John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Objective: To predict rate of instability-related delamination growth. Approximate stress analyses will be developed based on understanding gained from rigorous analyses. Experiments will be performed to obtain a data base for use by the analysis in making predictions and for verifying and improving the analysis.

PREDICTION OF STIFFNESS LOSS, RESIDUAL STRENGTH, AND FATIGUE LIFE OF UNNOTCHED LAMINATES
80 June 1 - 83 October 31

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Objective: To predict the stiffness loss, residual strength, and fatigue life of realistic unnotched laminates using baseline data from simple laminates.

FLIGHT SERVICE EVALUATION OF COMPOSITE COMPONENTS ON COMMERCIAL AND MILITARY AIRCRAFT
72 March 1 - 90 December 31

Project Engineer: H. Benson Dexter
Mail Stop 188A
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2869 FTS 928-2869

Objective: To evaluate the long-term durability of composite components installed on commercial and military transport and helicopter aircraft. Over 200 components constructed of boron, graphite, and Kevlar composites will be evaluated after extended service. Components include graphite/epoxy rudders, spoilers, tail rotors, vertical stabilizers, Kevlar/epoxy fairings, doors and ramp skins, boron-reinforced aluminum center wing boxes and tail cone, and boron/aluminum aft pylon skins. Note: Over 2.5 million total component flight hours have been accumulated since initiation of flight service in 1972. Composite components on L-1011, B-737, and DC-10 aircraft have accumulated over 22,000 flight hours each. Excellent in-service performance and maintenance experience have been achieved with the composite components.

POSTBUCKLING RESPONSE OF COMPOSITE MATERIAL SUBJECTED TO SHEAR LOADING
79 July 1 - 85 June 30

Project Engineer: Gary L. Farley
Mail Stop 188A
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2850 FTS 928-2850

Objective: To determine the postbuckling strength of Kevlar and Kevlar-graphite/epoxy composites under static shear and spectrum fatigue loading. This study will establish a basis for demonstrating the use of thin composite laminates beyond the point of initial shear instability. A shear fixture has been developed that virtually eliminates the adverse stresses in the corners of the shear panel.

THE ENERGY ABSORPTION OF COMPOSITE CRASHWORTHY STRUCTURE
80 August 1 - 85 December 31

Project Engineer: Gary L. Farley
Mail Stop 188A
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2850 FTS 928-2850

Objective: To determine the energy absorption characteristics of glass, Kevlar, and graphite/epoxy composites and to develop the analytical capability to predict the energy absorption characteristics of new composite materials. Tube specimens are being subjected to static and dynamic crushing tests. Over 30 combinations of materials and ply orientations have been tested. The research is focused on development of the capability to design efficient crashworthy composite structures for rotorcraft.

THE EVALUATION OF GRAPHITE/POLYIMIDE HONEYCOMB SANDWICH PANELS
79 June 15 - 83 March 31

Project Engineer: Jane A. Hagaman
Mail Stop 364
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3666 FTS 928-3666

Objective: To evaluate the shear behavior of an optimized sandwich panel at room and elevated temperatures using a diagonal tension test method, and to correlate the behavior with analytical predictions.

PRELIMINARY BOLTED JOINT DATA
78 July 1 - 81 October 31

Project Engineer: Gregory R. Wichorek
Mail Stop 188A
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2848 FTS 928-2848

Objective: To determine bolted joint strength and failure modes for advanced graphite/polyimide laminates from 116K and 589K, as well as the effect of joint geometry and temperature on joint strength and failure mode. Note: The results of the experimental program are reported in NASA TP-2015, "Experimental Data on Single-Bolt Joints in Quasi-Isotropic Graphite/Polyimide Laminates." Joint strength in net tension, bearing, and shear-out at 116K, 297K, and 589K is given for the Celion 6000/PMR-15 and Celion 6000/LARC-160 laminates.

DEVELOPMENT OF PRECISION ALIGNMENT FIXTURE FOR TENSILE TESTING
78 September 1 - 83 March 31

Project Engineer: Michael C. Lightfoot
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2143 FTS 928-2143

Objective: To determine the effect of precision alignment on the mean and variance of the tensile strength of composite materials.

EFFECTS OF THERMAL CYCLING ON DIMENSIONAL STABILITY OF GRAPHITE/EPOXY COMPOSITES
81 October 1 - 84 September 30

Project Engineer: Dr. Stephen S. Tompkins
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2143 FTS 928-2143

Objective: To determine the effects of thermal cycling from 117K to 400K on dimensional stability of graphite/epoxy composites.

RADIATION EFFECTS ON MATERIALS FOR STRUCTURAL COMPOSITES
79 July 1 - 84 June 30

Project Engineer: Dr. Edward R. Long, Jr.
Mail Stop 396
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3892 FTS 928-3892

Objective: To determine and correlate the effects of particulate radiation exposure on the properties and chemical structure of materials for structural composites and to develop procedures for accelerated laboratory simulation of long-term missions in a space radiation environment.

DIMENSIONAL STABILITY OF METAL-MATRIX COMPOSITES IN THE SPACE ENVIRONMENT
82 October 1 - 85 September 30

Project Engineer: Dr. Stephen S. Tompkins
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2143 FTS 928-2143

Objective: To determine and predict the dimensional changes induced by long-time exposure to the space environment.

EFFECT OF MICROCRACKING ON THE DIMENSIONAL STABILITY OF COMPOSITES
80 October 1 - 84 September 30

Project Engineer: David E. Bowles
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2143 FTS 928-2143

Objective: To develop analytical methods to predict the effect of microcracking on the dimensional stability of graphite/resin composites and correlate with experimental data.

POSTBUCKLING AND CRIPPLING OF COMPRESSION-LOADED COMPOSITE STRUCTURAL COMPONENTS
79 March 1 - 83 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2552 FTS 928-2552

Objective: To study the postbuckling and crippling of compression-loaded composite components and to determine the limitations of postbuckling design concepts to structural applications.

DESIGN TECHNOLOGY FOR STIFFENED CURVED COMPOSITE PANELS
79 October 1 - 83 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2552 FTS 928-2552

Objective: To develop verified design technology for generic advanced-composite stiffened curved panels.

COMPRESSION STRENGTH OF COMPOSITE LAMINATES WITH CUTOUTS
77 October 1 - 83 September 30

Project Engineer: Mark Shuart
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2813 FTS 928-2813

Objective: To study the effects of cutouts on the compression strength of composite structural components and to identify the failure modes that govern the behavior of compression-loaded components with cutouts.

POSTBUCKLING OF FLAT STIFFENED GRAPHITE/EPOXY SHEAR WEBS
81 July 1 - 83 September 30

Project Engineer: Marshall Rouse
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-4585 FTS 928-4585

Objective: To study the postbuckling response and failure characteristics of flat stiffened graphite/epoxy shear webs.

CURVED GRAPHITE/EPOXY PANELS SUBJECTED TO INTERNAL PRESSURE
80 October 1 - 83 September 30

Project Engineer: Richard L. Boitnott
Mail Stop 190
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3714 FTS 928-3714

Objective: To study the effects of internal pressure on the nonlinear response and failure characteristics of curved graphite/epoxy panels.

POSTBUCKLING ANALYSIS OF GRAPHITE/EPOXY LAMINATES
80 October 1 - 83 September 30

Project Engineer: Dr. Manuel Stein
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2813 FTS 928-2813

Objective: To develop accurate analyses for the postbuckling response of graphite/epoxy laminates and to determine the parameters that govern postbuckling behavior.

STRUCTURAL PANEL ANALYSIS AND SIZING CODE FOR STIFFENED PANELS
79 October 1 - 83 September 30

Project Engineers: Dr. Melvin S. Anderson
Dr. W. Jefferson Stroud
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3054 FTS 928-3054

Objective: To develop an accurate analysis and structural optimization capability for stiffened composite panels subjected to inplane tension, compression, shear, normal pressure, and thermal loads.

COMPRESSION STRENGTH OF COMPOSITE LAMINATES WITH LOW-VELOCITY IMPACT DAMAGE
76 October 1 - 83 September 30

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3524 FTS 928-3524

Objective: To study the effects of low-velocity impact damage on the compression strength of composite structural components and to identify the failure modes that govern the behavior of compression-loaded components subjected to low-velocity impact damage.

DAMAGE TOLERANT DESIGN TECHNOLOGY FOR COMPRESSION-LOADED COMPOSITE STRUCTURAL COMPONENTS
78 October 1 - 83 September 30

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3524 FTS 928-3524

Objective: To develop structural design concepts for containing and resisting damage in compression-loaded composite structural components.

CONTRACTS

INCREMENTAL ANALYSIS OF IMPACT DAMAGE
NAS1-15888
79 August 3 - 83 March 30

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2338 FTS 928-2338

Principal Investigator: Edward A. Humphreys
Materials Sciences Corporation
Gwynedd Plaza II
Bethlehem Pike
Spring House, Pennsylvania 19477
(215) 542-8400

Objective: To develop an incremental damage analysis that predicts the extent of fiber breaks and matrix delaminations as a projectile transfers energy to a laminate in discrete steps. At each step, failure criteria determine the advance of damage and thus establish the configuration for the next increment of deformation.

TRANSIENT STRAINS DUE TO IMPACT
NAS1-16763
81 August 20 - 82 December 31

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2338 FTS 928-2338

Principal Investigator: Dr. I. M. Daniel
IIT Research Institute
10 West 35th Street
Chicago, IL 60616
(312) 542-4402

Objective: To characterize impact damage in graphite/epoxy composite laminates and correlate it with transient strain and deformation history during impact. Plate and beam specimens containing embedded strain gages will be impacted with projectiles of various radii at two velocities.

IMPACT CONTACT STRESS ANALYSIS
NAG-1-222
81 November 1 - 83 October 31

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2338 FTS 928-2338

Principal Investigator: Dr. C. T. Sun
School of Aeronautics and Astronautics
Purdue University
West Lafayette, Indiana 47907
(317) 749-2527

Objective: To integrate the contact behavior and dynamic structural response to solve impact problems involving laminates under initial stress. With the aid of the previously-developed contact law, the dynamic response of the laminate will be modeled by finite elements. Impact damage will be investigated experimentally and correlated with the results of the analysis.

QUANTITATIVE RECONSTRUCTIVE ULTRASONIC IMAGING
NCCI-50
81 April 1 - 84 March 31

Project Engineer: Dr. Joseph S. Heyman
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3418 FTS 928-3418

Principal Investigator: Dr. Gary Brandenburger
Virginia Associated Research Campus
College of William and Mary
12070 Jefferson Avenue
Newport News, Virginia 23606
(804) 877-9231

Objective: To develop a state-of-the-art ultrasonic reconstructive imaging system for quantitative materials characterization.

QUANTITATIVE PHYSICAL ANALYSIS OF IMPACT DAMAGE
NSG-1601
80 March 1 - 84 February 28

Project Engineer: Dr. Joseph S. Heyman
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3418 FTS 928-3418

Principal Investigator: Professor James G. Miller
Laboratory for Ultrasonics
Physics Department
Washington University
St. Louis, Missouri 63130
(314) 889-6229

Objective: To improve nondestructive acoustic/ultrasonic techniques for quantitative characterization of defects in composite materials and to investigate new quantitative measurement phenomena applicable to graphite/epoxy.

NONEQUILIBRIUM MATERIAL EFFECTS ON THE BEHAVIOR OF POLYMERIC COMPOSITE MATRICES AND THEIR RELATED COMPOSITES
NAG-1-78
80 July 1 - 82 September 30

Project Engineer: Dr. Norman J. Johnston
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3041 FTS 928-3041

Principal Investigator: Dr. Garth L. Wilkes
Department of Chemical Engineering
Virginia Polytechnic Institute & State University
Blacksburg, Virginia 24061
(703) 961-5498

Objective: Following their processing as composite matrices, polymeric resins such as epoxies and polyimides initially are in a nonequilibrium state. The contractor will measure mechanical and sorption property changes of the resins as, over several thousand hours, they approach an equilibrium state. A determination will be made whether, and to what extent, resin property changes are reflected in carbon fiber composite property changes.

APPLICATION OF NONLINEAR IRREVERSIBLE THERMODYNAMICS TO COMPOSITE MATERIALS
NAS1-16301
80 August 18 - 82 September 30

Project Engineer: Dr. Norman J. Johnston
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3041 FTS 928-3041

Principal Investigator: Dr. Paul H. Lindenmeyer
Boeing Aerospace Company
P. O. Box 3999
Seattle, Washington 98124
(206) 237-5650

Objective: To apply nonlinear irreversible thermodynamics to the fracture mechanics of composite materials; namely, to determine how a composite will respond to a changing environment when the flow of energy (i.e., the power) becomes sufficiently great so that the system responds by the formation of dissipative rather than equilibrium structures or defects.

GRAPHITE FIBER SURFACE TREATMENT
NAS1-15869
80 September 26 - 82 September 30

Project Engineer: Dr. Terry L. St. Clair
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3041 FTS 928-3041

Principal Investigator: Dr. James T. Paul, Jr.
Hercules Incorporated
Research Center
Wilmington, Delaware 19899
(302) 995-3000

Objective: To determine how various sizings and coatings on graphite fibers affect the impact tolerance of composite panels made with the modified fibers. To determine the effect of such modifications on the mechanical and thermo-oxidative properties of laminates.

AD-A130 750 PROCEEDINGS OF THE ANNUAL MECHANICS OF COMPOSITES REVIEW (8TH) HELD AT WR..(U) AIR FORCE WRIGHT AERONAUTICAL LABS WRIGHT-PATTERSON AFB OH L A WILSON 5/5

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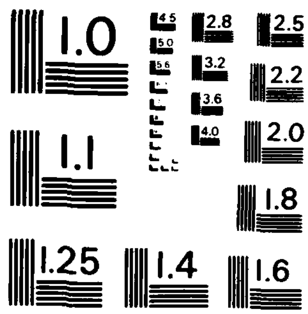

MICROCOPY RESOLUTION TEST CHART

NATIONAL BUREAU OF STANDARDS-1963-A

FIBER/MATRIX LOAD INTERACTIONS
NAS1-15749
80 September 19 - 82 August 31

Project Engineer: Dr. Stephen S. Tompkins
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2143 FTS 928-2143

Principal Investigator: Dr. Paul McMahon
Celanese Research Company
86 Morris Avenue
Summit, New Jersey 07901
(201) 522-7500, ext. 425

Objective: To characterize and model the interactions between fibers and resin within graphite/epoxy composite materials. To determine the effects of the interface properties on composite laminate properties and correlate the properties of the interface with lamina and laminate properties. Note: Results are reported in NASA CR-3607.

FRACTURE AND CRACK GROWTH IN ORTHOTROPIC LAMINATES
NSG-1606
79 July 1 - 83 May 31

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2338 FTS 928-2338

Principal Investigator: Dr. Jonathan Awerbuch
Department of Mechanical Engineering
Drexel University
Philadelphia, Pennsylvania 19104
(215) 895-2291

Objective: To explore the fracture characteristics of graphite/polyimide composites at elevated temperatures using laminates with slits.

FRACTURE AND CRACK GROWTH IN ORTHOTROPIC LAMINATES
NSG-1297
74 October 16 - 83 October 15

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2338 FTS 928-2338

Principal Investigator: Dr. James G. Goree
Department of Mechanical Engineering
Clemson University
Clemson, South Carolina 29631
(803) 656-3291

Objective: To develop analyses that predict strength of buffer strip panels using models that treat the fiber and matrix as discrete elements.

THREE-DIMENSIONAL STRESS ANALYSIS OF FASTENER HOLES
NCCI-36
80 October 1 - 84 September 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3012 FTS 928-3012

Principal Investigator: Dr. I. S. Raju
Mail Stop 188E
Joint Institute for Advancement of Flight Sciences
George Washington University at NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3178 FTS 928-3178

Objective: To compute 3-D stresses near unloaded and loaded fastener holes in laminated composites. These stresses are needed to predict the delamination onset in bolted joints.

THE VISCOELASTIC CHARACTERIZATION AND LIFETIME PREDICTION OF STRUCTURAL ADHESIVES
NAG-1-227
81 October 1 - 83 September 30

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2715 FTS 928-2715

Principal Investigator: Dr. H. F. Brinson
Department of Engineering Science and Mechanics
Virginia Polytechnic Institute & State University
Blacksburg, Virginia 24061
(703) 961-6627

Objective: To develop a procedure to predict the failures of adhesive joints where service life must span 10 to 20 years using, as a basis, analytical projections or extrapolations from short-time test data.

MATERIAL CHARACTERIZATION OF STRUCTURAL ADHESIVES IN THE LAP-SHEAR MODE
NAG-1-284
82 June 1 - 83 May 31

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2715 FTS 928-2715

Principal Investigator: Dr. Erol Sancaktar
Mechanical & Industrial Engineering Department
Clarkson College of Technology
Potsdam, New York 13676
(315) 268-2308

Objective: A general method for characterizing structural adhesives in the bonded lap-shear mode is proposed. Two approaches in the form of semi-empirical and theoretical approaches will be used.

EVALUATION OF THE DURABILITY AND DAMAGE TOLERANCE OF CTOL COMPOSITE STRUCTURES
NAS1-15107
77 October 12 - 83 July 1

Project Engineer: Edward P. Phillips
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3192 FTS 928-3192

Principal Investigator: Ray Horton
Boeing Commercial Airplane Company
P. O. Box 3707
Seattle, Washington 98124
(206) 241-3443

Objective: To perform selected analysis, fabrication, and testing tasks in the general area of durability and damage tolerance of graphite/epoxy composites, laminates, and structures. Current tasks involve design and fabrication of damage tolerance test specimens--unstiffened panels containing glass or Kevlar buffer strips and stiffened panels without buffer strips.

A STUDY OF STIFFNESS, RESIDUAL STRENGTH, AND FATIGUE LIFE RELATIONSHIPS FOR COMPOSITE LAMINATES
NAS1-16406
80 October 1 - 83 August 15

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Principal Investigators: Dr. James T. Ryder
Lockheed-California Company
Burbank, California 91520
(213) 847-6121, ext. 291

Dr. Frank W. Crossman
Lockheed Research Laboratory
Palo Alto, California 94304
(415) 858-4034

Objective: To develop quantitative relationships between laminate stiffness, residual strength, and fatigue life for unnotched laminates.

FATIGUE DAMAGE IN NOTCHED COMPOSITE LAMINATES UNDER TENSION-TENSION CYCLIC LOADS
NAG-1-232
82 January 1 - 83 January 1

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Structures Laboratory, USARTL (AVRADCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Principal Investigator: Dr. Wayne W. Stinchcomb
Department of Engineering Science & Mechanics
Virginia Polytechnic Institute & State University
Blacksburg, Virginia 24061
(703) 961-5316

Objective: To determine life-limiting fatigue damage mechanisms in graphite/epoxy laminates containing open holes and subjected to tension fatigue loading.

QUANTITATIVE STUDY OF INSTABILITY-RELATED DELAMINATION GROWTH
NAS1-16727
81 July 13 - 83 January 13

Project Engineer: John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3011 FTS 928-3011

Principal Investigator: Dr. R. L. Ramkumar
Dept. 3852/82
Northrop Corporation
Aircraft Division
One Northrop Avenue
Hawthorne, California 90250
(213) 970-5075

Objective: To predict the rate of instability-related delamination growth. Simple tests will be performed to quantify the relationship between delamination growth rate and Mode I and Mode II strain-energy-release rates.

DESIGN ALLOWABLES CHARACTERIZATION OF C6000/LARC-160 GRAPHITE/POLYIMIDE
NAS1-15183
80 October 1 - 82 October 15

Project Engineer: Gregory R. Wichorek
Mail Stop 188A
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2848 FTS 928-2848

Principal Investigator: H. Q. Norris
Rockwell International Corporation
Space Division
Seal Beach, California 90740
(231) 594-3289

Objective: To experimentally determine mechanical properties of graphite/polyimide laminates for use in designing aerospace structures for service at temperatures from 117K (-250°F) to 589K (600°F).

DESIGN ALLOWABLES CHARACTERIZATION OF C6000/PMR-15 GRAPHITE/POLYIMIDE
NAS1-15644
80 October 1 - 82 August 1

Project Engineer: Gregory R. Wichorek
Mail Stop 188A
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2848 FTS 928-2848

Principal Investigator: D. E. Skoumal
Boeing Aerospace Company
P. O. Box 3999
Seattle, Washington 99124
(206) 773-8016

Objective: To experimentally determine mechanical properties of graphite/polyimide laminates for use in designing aerospace structures for service at temperatures from 117K (-250°F) to 589K (600°F). Note: Results are reported in NASA CR-165840, "Design Allowables Test Program, Celion 3000/PMR-15 and Celion 6000/PMR-15, Graphite/Polyimide Composites," by J. B. Cushman and S. F. McCleskey. Test results show a consistent material performance over the temperature range and environmental conditions evaluated. Material strengths and stiffnesses were of sufficient magnitude to demonstrate Celion 3000/PMR-15 and Celion 6000/PMR-15 as viable materials for use in structural applications at temperatures up to 589K (600°F).

LSST HOOP/COLUMN ANTENNA: MATERIALS TASK
NAS1-15763
82 July 20 - 83 March 31

Project Engineer: Dr. Darrel R. Tenney
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2143 FTS 928-2143

Principal Investigator: Marvin Sullivan
Harris Corporation
P. O. Box 37
Melbourne, Florida 32901
(305) 727-5813

Objective: To develop tension stabilizing cables with a high degree of dimensional stability for use on a 100-meter diameter space deployable antenna; to develop lightweight, thermally stable composite materials for structural members and joints.

EFFECTS OF SPECIMEN VARIABILITY AND MATERIAL DEFECTS ON THERMAL EXPANSION OF GRAPHITE/EPOXY COMPOSITES
NCCI-15
80 October 1 - 83 June 30

Project Engineer: Dr. Stephen S. Tompkins
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2143 FTS 928-2143

Principal Investigator: Dr. M. W. Hyer
Department of Engineering Science & Mechanics
Virginia Polytechnic Institute & State University
Blacksburg, Virginia 20461
(703) 961-5905

Objective: To determine the effects of variability between specimens and material defects formed during fabrication on the thermal expansion of composite materials.

EFFECTS OF HIGH-ENERGY RADIATION ON THE MECHANICAL PROPERTIES OF GRAPHITE FIBER REINFORCED EPOXY RESINS
NSG-1562
79 October 1 - 82 December 31

Project Engineer: Dr. Edward R. Long, Jr.
Mail Stop 396
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3892 FTS 928-3892

Principal Investigators: Dr. Jasper D. Memory
Dr. Raymond E. Fornes
Departments of Physics and Textiles
North Carolina State University
Raleigh, North Carolina 27650
(919) 737-2503/737-3231

Objective: To investigate the effects of high-energy radiation on graphite fiber composites by study of composite curing effects, radiation exposure rates, mechanical fracture surfaces, and electron spin resonance properties.

ENVIRONMENTAL EXPOSURE EFFECTS ON COMPOSITE MATERIALS FOR COMMERCIAL AIRCRAFT
NAS1-15148
77 November 1 - 88 November 30

Project Engineer: Dr. Ronald K. Clark
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2143 FTS 928-2143

Principal Investigator: Daniel J. Hoffman
Boeing Commercial Airplane Company
P. O. Box 3707
Seattle, Washington 98124
(206) 241-3443

Objective: To provide technology in the areas of characterization methods and environmental effects on graphite/epoxy composite materials, including development of accelerated test and analysis methods to predict long-term performance of advanced resin-matrix composite materials within 20 percent of real-time aircraft service exposure results.

TIME-TEMPERATURE-STRESS CAPABILITIES OF COMPOSITE MATERIALS FOR ADVANCED SUPERSONIC TECHNOLOGY APPLICATIONS
NAS1-12308
73 June 1 - 84 September 30

Project Engineer: Bland A. Stein
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3041 FTS 928-3041

Principal Investigator: J. F. Haskins
Mail Zone 43-6320
General Dynamics
P. O. Box 80847
San Diego, California 92138
(714) 891-8900, ext. 2088

Objective: To establish the time-temperature-stress characteristics and capabilities of five classes of high-temperature composite materials (graphite/epoxy, boron/epoxy, graphite/polyimide, boron/polyimide, and boron/aluminum) subjected to a simulated supersonic cruise flight environment for up to 50,000 hours. Note: Phase I of this contract has been documented in NASA CR-159267.

EFFECTS OF STRESS CONCENTRATIONS IN COMPOSITE STRUCTURES
NSG-1483
78 January 15 - 83 January 15

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2552 FTS 928-2552

Principal Investigators: Dr. Wolfgang G. Knauss
Dr. Charles D. Babcock
California Institute of Technology
Pasadena, California 91125
(213) 795-6811, ext. 1524/1528

Objective: To study the effects of low-speed impact damage in composite structural components using high-speed motion pictures and to develop an analytical procedure for the propagation of the resulting impact damage.

ADVANCED COMPOSITE STRUCTURAL DESIGN TECHNOLOGY FOR COMMERCIAL TRANSPORT AIRCRAFT
NAS1-15949
79 September 24 - 84 September 24

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2552 FTS 928-2552

Principal Investigator: John N. Dickson
Lockheed-Georgia Company
86 South Cobb Drive
Marietta, Georgia 30063
(404) 424-3085

Objective: To design, analyze, fabricate, and test generic advanced-composite structural components for transport aircraft applications in order to develop verified design technology.

STRUCTURAL OPTIMIZATION FOR IMPROVED DAMAGE TOLERANCE
NAG-1-168
81 September 1 - 83 October 15

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-2552 FTS 928-2552

Principal Investigator: Dr. Raphael T. Huftka
Virginia Polytechnic Institute & State University
Blacksburg, Virginia 24061
(703) 961-6611

Objective: To develop a structural optimization procedure for composite wing boxes that includes the influence of damage-tolerance considerations in the design process.

STRUCTURAL TEST SPECIMENS USING FIBER-REINFORCED COMPOSITE MATERIALS
NAS1-12675
73 September 6 - 83 March 6

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3524 FTS 928-3524

Principal Investigator: Cliff Kam
Douglas Aircraft Company
3855 Lakewood Blvd.
Long Beach, California 90846
(213) 593-5332

Objective: To design, fabricate, and test composite compression components for structural applications; to develop fabrication procedures for stiffened panels; and to evaluate damage tolerant materials.

EVALUATION OF THE DURABILITY AND DAMAGE TOLERANCE OF COMPOSITE STRUCTURES SUITABLE FOR COMMERCIAL TRANSPORT AIRCRAFT
NAS1-15107
77 October 1 - 83 September 30

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3524 FTS 928-3524

Principal Investigator: John E. McCarty
Boeing Commercial Airplane Company
P. O. Box 3707
Seattle, Washington 98124
(206) 433-1430

Objective: To design, fabricate, and test generic composite structural components for commercial aircraft applications that are durable and damage tolerant.

COMPRESSION FAILURE MECHANISMS OF COMPOSITE STRUCTURES
NAG-1-295
82 September 1 - 83 August 31

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 827-3524 FTS 928-3524

Principal Investigator: Dr. H. Thomas Hahn
Washington University
St. Louis, Missouri 63130
(314) 889-6052

Objective: To study the effects of material properties on microbuckling and shear crippling failure modes.

NAVAL AIR SYSTEMS COMMAND
WASHINGTON, D.C. 20361

INHOUSE

FATIGUE OF COMPOSITES UNDER COMPLEX LOADS
79 October - 83 September

Project Engineer: Dr. P. W. Mast
Naval Research Laboratory
Washington, D.C. 20375
(202) 767-2165 Autovon 297-2165

Objective: Develop a capability for predicting the structural response and initiation of failure in composite laminates under complex cyclic loading.

DETERIORATION OF LAMINATING RESINS
81 October 1 - 82 September 30

Project Engineer: Dr. J. Augl
Naval Surface Weapons Center
White Oak, Silver Spring, MD 20910
(204) 394-2262 Autovon 290-2261

Objective: To investigate the mechanisms of composite degradation under storage and service environments and to develop quantitative analytical procedures to predict such degradations and to verify these predictions experimentally.

HIGH PERFORMANCE COMPOSITES & ADHESIVES FOR NAVAL AIRCRAFT
81 October 1 - 82 September 30

Project Engineer: Dr. C. Poranski
Naval Research Laboratory
Washington, DC 20375
(202) 767-2336

Objective: To provide guidance for selection ot high performance polymers for adhesive and composite systems.

CONTRACTS

DELAMINATION FAILURE CRITERIA FOR COMPOSITE STRUCTURES
80 August - 83 September

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, DC 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Dr. R. Wilkins
General Dynamics
Fort Worth, TX 76101
(817) 732-4811 Ext. 4631

Objective: Conduct experimental studies to develop a delamination failure criteria and analysis methods to predict debonding in composite structures.

DELAMINATION IN COMPOSITE STEPPED LAP JOINTS
80 August - 83 June

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, DC 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Dr. M. M. Ratwani
Northrop Corporation
Hawthorne, CA 90250
(213) 970-5285

Objective: Conduct analytical and experimental studies of delamination in a laminated composite metallic stepped lap joint configuration.

THERMOPLASTIC MATRIX COMPOSITES
N00019-81-C-0345
81 July 22 - 82 September 30

Project Engineer: Max Stander
Naval Air Systems Command
Washington, DC 20361
(202) 692-6025 Autovon 222-6025

Principal Investigator: Mr. E. House
Boeing Aerospace Corporation
Seattle, WA 98124
(206) 655-3081

Objective: To evaluate the engineering properties of various thermoplastic resins and known reinforcements for aircraft structural application.

FATIGUE INDUCED DAMAGE IN COMPOSITE LAMINATES
78 October - 83 April

Project Engineer: Dr. Y. Rajapakse
Office of Naval Research
Washington, DC 22217
(202) 696-4307 Autovon 226-4307
and
Dr. D. R. Mulville
Naval Air Systems Command
Washington, DC 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Dr. Z. Hashin
University of Pennsylvania
Philadelphia, PA 19104

Objective: Develop analytical models to describe the response of composite laminates to cyclic loading, include problems of reduction of elastic moduli due to initiation of microcracking.

ACCEPTANCE CRITERIA FOR GRAPHITE/EPOXY STRUCTURES
79 July - 82 March

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, DC 20361
(202) 692-2515 Autovon 222-2515

Principal Investigator: Mr. R. Riley
McDonnell Aircraft Company
St. Louis, MO 63166
(314) 232-0232

Objective: Conduct an experimental investigation to determine the effects of various void levels in graphite/epoxy composites on structural response under combined compression and shear loading.

FATIGUE LIFE AND RESIDUAL STRENGTH OF COMPOSITE STRUCTURES
81 September - 83 September

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, DC 20361
(202) 692-2515 Autovon 222-2515

Principal Investigators: Dr. J. Yang and
Dr. D. Jones
The George Washington University
Washington, DC 20052
(202) 676-6929

Develop statistical models to describe fatigue life and residual strength of composite structures.

HOLES AND FASTENERS FOR ADVANCED COMPOSITES
N00019-81-C-0387
81 October 28 - 82 July 30

Project Engineer: Max Stander
Naval Air Systems Command
Washington, DC 20361
(202) 692-6025 Autovon 222-6025

Principal Investigator: H. Turner
McDonnell Douglas Corporation
St. Louis, MO 63166
(314) 232-9501

Objective: To develop optimum hole preparation and fastener installation techniques.

COMPRESSION FATIGUE OF COMPOSITES
N00019-82-C-0156 (Northrop)
N00019-82-C-0063 (U. Wyo)
82 May

Project Engineer: Max Stander
Naval Air Systems Command
Washington, DC 20361
(202) 692-6025 Autovon 222-6025

Principal Investigators: Mr. R. L. Ramkumar
Northrop Corporation
3901 W. Broadway
Hawthorne, CA 90250
(213) 970-5075

Mr. Don Adams
University of Wyoming
Laramie, WY 82071
(307) 766-2371

Objective: To investigate the compression fatigue properties of graphite fiber epoxies, particularly under moist conditions.

METALLIC COATINGS FOR ADVANCED COMPOSITES
N00019-80-C-0059
80 March

Project Engineer: Max Stander
Naval Air Systems Command
Washington, DC 20361
(202) 692-6025 Autovon 222-6025

Principal Investigator: Mr. C. Staebler
Grumman Aerospace Corporation
Bethpage, L.I., NY 11714
(516) 575-2244

Objective: To explore and evaluate the trade-offs associated with the use of metallic coatings on graphite epoxy composites.

NAVAL AIR DEVELOPMENT CENTER
WARMINSTER, PA 18974

INHOUSE

COMPOSITE IMPACT RESISTANCE
74 March - Continuing

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
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Objective: Ascertain the impact response of generic composite structural elements and identify the physical mechanisms associated with impact damage and the critical parameters governing impact response.

HYBRID COMPOSITE FRACTURE CHARACTERIZATION
80 September - 84 September

Project Engineer: Lee W. Gause
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Objective: Characterize the strength, mechanical properties, and failure characteristics of woven and intimately mixed hybrid composite laminates.

CONTRACTS

DESIGN OF HIGHLY LOADED COMPOSITE JOINTS
AND ATTACHNENTS FOR TAIL STRUCTURES
N62269-82-C-0239
82 February - 84 July

Project Engineer: Ramon Garcia
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Principal Investigator: S. J. Kong
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(213) 970-3442

Objective: To develop composite designs which will permit the use of metal to composite bolted root attachments in aircraft tail structures as an alternative to high-load transfer adhesive bonded titanium step joints. To improve damage tolerance, survivability and repairability over current composite designs. Structural efficiency, manufacturing feasibility and quality assurance requirements will be determined.

DESIGN OF HIGHLY LOADED COMPOSITE JOINTS
AND ATTACHMENTS FOR WING STRUCTURES
N62269-82-C-0238
82 February - 84 July

Project Engineer: Ramon Garcia
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Principal Investigator: S. Garbo
McDonnell Aircraft Co.
P.O. Box 516
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(314) 233-8626

Objective: To develop composite designs which will permit the use of metal to composite bolted root attachments in aircraft wing structures as an alternative to high-load transfer adhesive bonded titanium step joints. Strain concentration around fastener holes, fatigue and environmental affects, damage tolerance and repairability for each concept will be determined.

MAGNAWEAVE TEST SPECIMENS
N62269-81- 0274
81 June - 82 September

Project Engineer: Lee W. Gause
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Principal Investigator: Dr. R. A. Florentine
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26 S. Wakefield Road
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(215) 539-7280

Objective: Provide a variety of multi-dimensional woven composite test specimens to be used for static, impact and fatigue strength characterization.

EFFECTS OF LAYUP AND LOADING FREQUENCY ON FATIGUE LIFE OF GRAPHITE/EPOXY
N62269-81-C-0258
81 September - 82 September

Project Engineer: Lee W. Gause
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Principal Investigator: C. R. Saff
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Objective: The effect of cyclic frequency on fatigue life of layups used in aircraft structure is not well characterized. The objective of this program is to determine the sensitivity of these layups to load frequency and to develop analysis techniques that can identify particularly sensitive layups so that their behavior can be considered in the design and analysis of aircraft structure.

QUASI 3-DIMENSIONAL FINITE ELEMENT ANALYSIS OF DELAMINATION GROWTH IN COMPOSITES
N62269-82-C-0250
82 April - 83 October

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-2867 Autovon 441-2867

Principal Investigator: Dr. A. S. D. Wang
Drexel University
Philadelphia, PA 19104
(215) 895-2297

Objective: Linear elastic fracture mechanics will be generalized to the more complicated, two-dimensional delamination process by providing a convenient computational scheme to accurately determine the three-dimensional stress field surrounding a delamination crack and a general delamination growth criterion developed.

POLYMER MATRIX FATIGUE PROPERTIES
N62269-80-C-0278
80 September - 82 September

Project Engineer: Lee W. Gause
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ACSTD/6043
Warminster, PA 18974
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Principal Investigator: Dr. D. F. Adams
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Objective: Characterize and compare the fatigue properties of various matrix materials and correlate the resin fatigue properties to composite laminate fatigue behavior.

SUPPRESSION OF DELAMINATION IN COMPOSITES BY THICKNESS DIRECTION REINFORCEMENT
N62269-82-C-0248
82 June - 83 December

Project Engineer: Lee W. Gause
Naval Air Development Center
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Warminster, PA 18974
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Principal Investigator: Dr. C. T. Sun
Purdue University
West Lafayette, Indiana 47907
(317) 494-5130

Objective: Improve the damage tolerance and durability of laminated composite structure by providing thickness-direction reinforcement to constrain the growth of delamination damage.

DEVELOPMENT OF HIGH STRAIN COMPOSITE WING
N62269-81-C-0727
81 September - 84 March

Project Engineer: Mark Libeskind
Naval Air Development Center
ACSTD/60433
Warminster, PA 18974
(215) 441-2866 Autovon 441-2866

Principal Investigator: J. Bruno
Grumman Aerospace Corp.
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Objective: Design and evaluate an advanced composite wing which operates at significantly higher strain levels than current composite wings resulting in significant weight savings. Emphasis will be placed upon damage tolerance, survivability, durability and repairability.

APPENDIX C

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DAYTON, OHIO
5-7 October 1982

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